DOUGLAS DC-3

MAINTENANCE MANUAL

DOUGLAS

AOL
ALL OPERATOR LETTER

PRODUCT SUPPORT

BOEING

April 16, 1998
L30-98-025/AOL/ACH
3-30-00
4-30-00

AOL 3-016
AOL 4-009

To: All DC-3 and DC-4 Operators

Subject: AIRWORTHINESS DIRECTIVE FOR OPERATING IN SEVERE ICING CONDITIONS

Applicable To: All DC-3 and DC-4 Airplanes

Reference: Federal Aviation Administration (FAA) Airworthiness Directive (AD) Docket No. 97-NM-173-AD; Amendment 39-10347; AD 98-04-35, effective March 25, 1998

REASON

THE AIRPLANE FLIGHT MANUAL HAS BEEN REVISED FOR OPERATING IN SEVERE ICING CONDITIONS.

The reference AD has been issued by the FAA. Sections entitled, "Addresses", "For Further Information Contact", and "Supplemental Information", are not included in this transmittal.

Operators requiring a complete copy of the reference AD should contact their regulatory agency or Boeing Commercial Airplane Group.

QUOTE:

SUMMARY: This amendment adopts a new airworthiness directive (AD), applicable to all McDonnell Douglas Model DC-3 and DC-4 series airplanes, that requires revising the Airplane Flight Manual (AFM) to specify procedures that would prohibit flight in severe icing conditions (as determined by certain visual cues), limit or prohibit the use of various flight control devices while in severe icing conditions, and provide the flight crew with recognition cues for, and procedures for exiting from, severe icing conditions.

This amendment is prompted by results of a review of the requirements for certification of the airplane inicing conditions, new information on the icing environment, and icing data provided

currently to the flight crews. The actions specified by this AD are intended to minimize the potential hazards associated with operating the airplane in severe icing conditions by providing more clearly defined procedures and limitations associated with such conditions.

EFFECTIVE DATE: March 25, 1998.

UNQUOTE:

QUOTE:

Comments

Interested persons have been afforded an opportunity to participate in the making of this amendment. Due consideration has been given to the following comments received.
In addition to the proposed rule described previously, in September 1997, the FAA issued 24 other similar proposals that address the subject unsafe condition on various airplane models (see below for a listing of all 24 proposed rules). These 24 proposals also were published in the Federal Register on September 16, 1997. This final rule contains the FAA's responses to all relevant public comments received for each of these proposed rules.

Docket No.	Manufacturer/airplane model	Federal Register Citation
97-CE-49-AD	Aerospace Technologies of Australia, Models N22B and N24A	62 FR 48520
97-CE-50-AD	Harbin Aircraft Mfg., Corporation Model Y12IV	62 FR 48513
97-CE-51-AD	Partenavia Costruzioni Aeronauticas, S.p.A., Models P68, AP68TP 300, AP68TP 600	62 FR 48524
97-CE-52-AD	Industrie Aeronautiche e Meccaniche Rinaldo Piaggio S.p.A., Model P-180	62 FR 48502
97-CE-53-AD	Pilatus Aircraft Ltd., Models PC-12 and PC-12/45	62 FR 48499
97-CE-54-AD	Pilatus Britten-Norman Ltd., Models BN-2A, BN-2B, and BN-2T	62 FR 48538
97-CE-55-A D	SOCATA-Groupe Aerospatiale, Model	

	TBM-700	62 FR 48506
97-CE-56-AD	Aerostar Aircraft Corporation, Models PA-60-600, -601,-601P, -602P, and -700P	62 FR 48481
97-CE-57-AD	Twin Commander Aircraft Corporation, Models 500, -500-A, -500-B,-500-S, -500-U, -520, -560, -560-A, -560-E, -560-F,-680, -680-E, -680FL(P), -680T, -680V, -680W, -681,-685, -690, -690A, -690B, -690C, -690D, -695, -695A, -695B, and -720	62 FR 48549
97-CE-58-AD	Raytheon Aircraft Company (formerly known as Beech Aircraft Corporation), Models E55, E55A, 58, 58A, 58P, 58PA, 58TC, 58TCA, 60 series, 65-B80 series, 65-B-90 series, 90 series, F90 series, 100 series, 300 series, and B300 series	62 FR 48517
97-CE-59-AD	Raytheon Aircraft Company (formerly known as Beech Aircraft Corporation), Model 2000	62 FR 48531
97-CE-60-AD	The New Piper Aircraft, Inc., Models PA-46 -310P and PA-46-350P	62 FR 48542
97-CE-61-AD	The New Piper Aircraft, Inc., Models PA-23, PA-23-160, PA-23-235, PA-23-250, PA-E23-250, PA-30, PA-39, PA-40, PA-31, PA-31-300, PA-31-325, PA-31-350, PA-34-200, PA-34-200T, PA-34-220T, PA-42, PA-42-720, PA-42-1000	62 FR 52294
97-CE-62-AD	Cessna Aircraft Company, Models P210N, T210N, P210R, and 337 series	62 FR 48535
97-CE-63-AD	Cessna Aircraft Company, Models T303, 310R, T310R, 335, 340A, 402B, 402C, 404, F406, 414, 414A,421B, 421C, 425, and 441	62 FR 48528
97-CE-64-AD	SIAI-Marchetti S.r.I. (Augusta),	

	Models SF600 and SF600A	62 FR 48510
97-NM-170-AD	Cessna Aircraft Company, Models 500, 501, 550, 551, and 560 series	62 FR 48560
97-NM-171-AD	Sabreliner Corporation, Models 40, 60, 70, and 80 series	62 FR 48556
97-NM-172-AD	Gulfstream Aerospace, Model G-159 series	62 FR 48563
97-NM-173-AD	McDonnell Douglas, Models DC-3 and DC-4 series	62 FR 48553
97-NM-174-AD	Mitsubishi Heavy Industries, Models YS-11 and YS-11A series	62 FR 48567
97-NM-175-AD	Frakes Aviation, Models G-73 (Mallard) and G-73T series	62 FR 48577
97-NM-176-AD	Lockheed, Models L-14 and L-18 series	62 FR 48574
97-NM-177-AD	Fairchild, Models F27 and FH227 series	62 FR 48570

Unquote:

Quote:

Sec. 39.13 (Amended)

2. Section 39.13 is amended by adding the following new airworthiness directive:

98-04-35 McDonnell Douglas: Amendment 39-10347. Docket 97-NM-173-AD.

Applicability: All Model DC-3 and DC-4 series airplanes, certificated in any category.

Note 1: This AD applies to each airplane identified in the preceding applicability provision, regardless of whether it has been modified, altered, or repaired in the area subject to the requirements of this AD. For airplanes that have been modified, altered, or repaired so that the performance of the requirements of this AD is affected, the owner/operator must request approval for an alternative method of compliance in accordance with paragraph (b) of this AD. The request should include an assessment of the effect of the modification, alteration, or repair on the unsafe condition addressed by this AD; and, if the unsafe condition has not been eliminated, the request should include specific proposed actions to address it.

Compliance: Required as indicated, unless accomplished previously. To minimize the potential hazards associated with operating the airplane in severe icing conditions by providing more clearly defined procedures and limitations associated with such conditions, accomplish the following:

(a) Within 30 days after the effective date of this AD, accomplish the requirements of paragraphs (a)(1) and (a)(2) of this AD.

Note 2: Operators should initiate action to notify and ensure that flight crewmembers are apprised of this change.

(1) Revise the FAA-approved Airplane Flight Manual (AFM) by incorporating the following into the Limitations Section of the AFM. This may be accomplished by inserting a copy of this AD in the AFM.

Warning

Severe icing may result from environmental conditions outside of those for which the airplane is certificated. Flight in freezing rain, freezing drizzle, or mixed icing conditions (super cooled liquid water and ice crystals) may result in ice build-up on protected surfaces exceeding the capability of the ice protection system, or may result in ice forming aft of the protected surfaces. This ice may not be shed using the ice protection systems, and may seriously degrade the performance and controllability of the airplane.

** During flight, severe icing conditions that exceed those for which the airplane is certificated shall be determined by the following visual cues. If one or more of these visual cues exists, immediately request priority handling from Air Traffic Control to facilitate a route or an altitude change to exit the icing conditions.

--Unusually extensive ice accumulation on the airframe and windshield in areas not normally observed to collect ice.
--Accumulation of ice on the upper surface of the wing aft of the protected area.
--Accumulation of ice on the engine nacelles and propeller spinners farther aft than normally observed.

** Since the autopilot, when installed and operating, may mask tactile cues that indicate adverse changes in handling characteristics, use of the autopilot is prohibited when any of the visual cues specified above exist, or when unusual lateral trim requirements or autopilot trim warnings are encountered while the airplane is in icing conditions.
** All wing icing inspection lights must be operative prior to flight into known or forecast icing conditions at night. NOTE: This supersedes any relief provided by the Master Minimum Equipment List (MMEL).
(2) Revise the FAA-approved AFM by incorporating the following into the Normal Procedures Section of the AFM. This may be accomplished by inserting a copy of this AD in the AFM.

The Following Weather Condition May Be Conducive to Severe In-Flight Icing

** Visible rain at temperatures below 0 degrees Celsius ambient air temperature.
** Droplets that splash or splatter on impact at temperatures below 0 degrees Celsius ambient air temperature.

"Procedures for Exiting the Severe Icing Environment

These procedures are applicable to all flight phases from takeoff to landing. Monitor the ambient air temperature. While severe icing may form at temperatures as cold as -18 degrees Celsius, increased vigilance is warranted at temperatures around freezing with visible moisture present. If the visual cues specified in the Limitations Section of the AFM for identifying severe icing conditions are observed, accomplish the following:
** Immediately request priority handling from Air Traffic Control to facilitate a route or an altitude change to exit the severe icing conditions in order to avoid extended exposure to flight conditions more severe than those for which the airplane has been certificated.
** Avoid abrupt and excessive maneuvering that may exacerbate control difficulties.
** Do not engage the autopilot.
** If the autopilot is engaged, hold the control wheel firmly and disengage the autopilot.
** If an unusual roll response or uncommanded roll control movement is observed, reduce the angle-of-attack.
** Do not extend flaps when holding in icing conditions. Operation with flaps extended can result in a reduced wing angle-of-attack, with the possibility of ice forming on the upper surface further aft on the wing than normal, possibly aft of the protected area.
** If the flaps are extended, do not retract them until the airframe is clear of ice.
** Report these weather conditions to Air Traffic Control."
(b) An alternative method of compliance or adjustment of the compliance time that provides an acceptable level of safety may be used if approved by the Manager, Los Angeles Aircraft Certification Office (ACO), FAA, Transport Airplane Directorate. The request shall be forwarded through an appropriate FAA Operations Inspector, who may add comments and then send it to the Manager, Los Angeles ACO.

Note 3: Information concerning the existence of approved alternative methods of compliance with this AD, if any, may be obtained from the Los Angeles ACO.

(c) Special flight permits may be issued in accordance with sections 21.197 and 21.199 of the Federal Aviation Regulations (14 CFR 21.197 and 21.199) to operate the airplane to a location where the requirements of this AD can be accomplished.

(d) This amendment becomes effective on March 25, 1998.

Issued in Renton, Washington, on February 6, 1998.

Darrell M. Pederson,

Acting Manager, Transport Airplane Directorate, Aircraft Certification Service.

Unquote:

Should additional information be required, please submit your inquiries through your local Field Service Representative or to Boeing Commercial Airplane Group - Douglas Products Division, ATTN: Technical and Fleet Support – Propulsion, Environmental, and Interiors Systems, P. O. Box 1771, Long Beach, California 90801; SITA: TOAMD7X, ARINC: LAXMDCR, Telex 674357, FAX: (562) 593-7710, or call (562) 593-7268.

THIS AOL IS FOR ADVISORY PURPOSES ONLY. NO FAA APPROVAL IS REQUIRED.

Fred C. Haas

Fred C. Haas
Director
Technical AND Fleet Suport
Service Engineering
Customer Division

ACH:mp
(NAA)

BOEING

March 5, 1998
L30-017-98/AOL
3-112-0
4-112-0
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AOL 3-015
AOL 4-008
AOL 6-011
AOL 7-008
AOL 8-1248
AOL 9-2564
AOL 90-083
AOL 10-2214
AOL 11-464

To: All DC-3, DC-4, DC-6, DC-7, DC-8, DC-9, C-9, MD-80, MD-90, DC-10, KC-10, KDC-10, AND MD-11 OPERATORS

Subject: MANUFACTURE OF FEDERAL AVIATION ADMINISTRATION (FAA)-APPROVED REPLACEMENT PARTS

Applicable To: All DC-3, DC-4, DC-6, DC-7, DC-8, DC-9, C-9, MD-80, MD-90, DC-10, KC-10A, KDC-10, and MD-11 Airplanes

Reference:
(a) Federal Aviation Regulation (FAR) Part 21, Subpart K - Approval of Materials, Parts, Processes, and Appliances
(b) FAA Advisory Circular No. AC 20-62D, Eligibility, Quality, and Identification of Aeronautical Replacement Parts
(c) FAA Advisory Circular No. AC 21-29A, Detecting and Reporting Suspected Unapproved Parts

REASON

SEVERAL OPERATORS HAVE REQUESTED CLARIFICATION ON CERTAIN TERMINOLOGY USED BY BOEING DOUGLAS PRODUCTS DIVISION (DPD) WHEN TRANSMITTING TYPE DESIGN DATA TO OWNERS, OPERATORS, AND LICENSEES, AND REGULATORY REQUIREMENTS FOR AIRWORTHINESS OF PARTS MANUFACTURED BY THEM.

Boeing may provide type design data to its airline customers and to licensees to support the fleet when replacement parts are not available directly from Boeing in a timely manner.

The process of providing type design data to Boeing's airline customers and licensees has been known as Authority To Manufacture (ATM). The ATM process does not imply that Boeing approves any parts manufactured by airline customers or licensees. Boeing may grant authorization to owners, operators, and licensees to use Boeing type design data, but Boeing does not, and cannot, grant authority to manufacture aircraft replacement parts. **The United States (US) Federal Aviation Administration (FAA) grants authority to manufacture aircraft replacement parts for US type certificated aircraft (reference [a]).** Reference (a) also exempts owners and operators from the requirements to produce modification or replacement parts pursuant to FAA Parts Manufacturer Approval (PMA) when maintaining or altering the owner's or operator's own type certificated aircraft (ref. FAR section 211.303[b][2]).

Sections 91.403, 121.363, and 125.243 of the FAR place the responsibility for airworthiness of the type certificated product (aircraft, aircraft engines, propellers) on the owner or operator. Airworthiness for United States operators is defined in Public Law 103-272 as conforming to the (approved) type certificate, and being in a condition for safe operation.

Boeing Provided Type Design Data:

Type design data, as an element of the type certificate, is FAA-Approved. Boeing's type certificates are elements of Boeing's FAA-Approved Production Certificate (PC). Boeing produces parts in accordance with PC requirements (reference FAR Part 21, Subpart G - Production Certificates). Mere possession of Boeing type design data, including detail drawings, specifications, and other data, does not imply that Boeing approves the recipient as a supplier pursuant to Boeing's PC. The manufacture of parts using Boeing type design data, unless accomplished expressly in accordance with the requirements of Boeing's PC, is to be considered outside of Boeing's control and responsibility. Boeing assumes no responsibility for the quality, or the airworthiness of owner, operator, or licensee manufactured parts.

Parts Marking:

Boeing identifies parts in accordance with the provisions of its PC. FAR section 43.9 requires a maintenance record entry of the work performed, which only implies owner, operator, and licensee produced parts be identified as such. Marking requirements for parts for which a replacement time, inspection interval, or related procedure is specified in the Airworthiness Limitations section of the manufacturer's maintenance manual or Instructions for Continued Airworthiness are cited in FAR section 45.14. There is no specific regulatory requirement to mark owner, operator, or licensee manufactured parts as **not** manufactured by an FAA production approval holder. Once owner or operator or licensee produced parts are installed on the aircraft, they are in the stream of commerce, and if marked as if produced under an FAA-Approved production system (that is, with part numbers identical to those used in the FAA-Approved production system), might possibly be considered fraudulent (reference [a]), section 43.12). The commission by any person of an act prohibited under Section 43.12(a) is a basis for, among other things, suspending or revoking the applicable airman, or operator certificate held by that person.

Owner or operator manufactured parts, for those owners or operators holding Boeing Purchase Agreement (contract) are required, by the terms of that agreement, to be marked with a part number different from the original Boeing part number to identify that the parts so produced are the product of the owner or operator, and not of Boeing.

Should additional information be required, please submit your inquiries through your local Field Service Representative or to Boeing Commercial Airplane Group, ATTN: Contract Data Management, P. O. Box 1771, Long Beach, California 90801; SITA: TOAMD7X, ARINC: LAXMDCR, Telex 674357, FAX: (562) 593-2001, or call (562) 496-9621.

THIS AOL IS FOR ADVISORY PURPOSES ONLY. NO FAA APPROVAL REQUIRED.

F. C. Haas

F. C. Haas
Director
Technical and Fleet Support
Service Engineering
Customer Division

MEF:cr
(NAA)

Handbook

Maintenance Instructions

NAVY MODELS
R4D-8, R4D-8Z
AIRCRAFT

THIS PUBLICATION SUPERSEDES AN 01-40NK-2
DATED 15 MAY 1952 REVISED 15 DECEMBER 1952

NOTE: A COPY OF AN 02A-35GK-2, SERVICE INSTRUCTIONS FOR MODEL R-1820-80 ENGINES, SHOULD BE FILED WITH SECTION V OF THIS HANDBOOK

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE
AND THE CHIEF OF THE BUREAU OF AERONAUTICS

15 April 1953

LIST OF REVISED PAGES ISSUED

INSERT LATEST REVISED PAGES. DESTROY SUPERSEDED PAGES.

NOTE: The portion of the text affected by the current revision is indicated by a vertical line in the outer margins of the page.

* The asterisk indicates pages revised, added or deleted by the current revision.

ADDITIONAL COPIES OF THIS PUBLICATION MAY BE OBTAINED AS FOLLOWS: BuAer

USAF ACTIVITIES.—In accordance with Technical Order No. 00-5-2.

NAVY ACTIVITIES.—Submit request to nearest supply point listed below, using form NavAer-140: NASD, Philadephia, Pa.; NAS, Alameda, Calif.; NAS, Jacksonville, Fla.; NAS, Norfolk, Va.; NAS, San Diego, Calif.; NAS, Seattle, Wash.; ASD, NSC, Guam.

For listing of available material and details of distribution see Naval Aeronautics Publications Index NavAer 00-500.

TABLE OF CONTENTS

INTRODUCTION

This publication covers maintenance instructions for the R4D-8 aircraft manufactured by the Douglas Aircraft Company, Inc., Santa Monica, California, under contract NOa(s) 51-012. This Maintenance Handbook contains essential information required for field maintenance and service of the complete airframe, as well as for engines, accessories, and instruments installed in the aircraft.

Instructions included in this handbook are applicable to R4D-8 aircraft as delivered. For additional information concerning the R4D-8 aircraft, consult the following publications:

Handbook of Flight Operating Instructions ....AN 01-40NK-1
Handbook of Structural Repair Instructions ....AN 01-40NK-3
Illustrated Parts Catalog ....AN 01-40NK-4
Handbook of Inspection Requirements ....AN 01-40NK-6
Handbook of Weight and Balance ....AN 01-1-40
Handbook of Cargo Loading Instructions ....NavAer 01-40NK-509

The following practices have been applied in preparation of the text for this handbook:

Right and left, clockwise and counterclockwise, upper and lower, and similar references apply to the aircraft as viewed from the rear of the aircraft center line, looking forward.

Pagination throughout this handbook is continuous with information contained herein divided into sections numbered consecutively in capital Roman numerals. The sections, in turn, are divided into paragraphs numbered consecutively with each section utilizing Arabic numbers separated by a dash. The number preceding the dash indicates the section (converted from Roman to Arabic) with the number following the dash indicating the paragraph within the section. As an example, "5-4" would be the fourth paragraph of Section V.

Illustrations are identified by figure number and title positioned beneath each illustration. Figures are numbered consecutively within each section, similar to the manner of paragraph numbering. For example, "4-15" represents the fifteenth illustration in Section IV.

The Table of Contents lists the major divisions of the handbook together with related page number and paragraph sequence. In addition, a Table of Contents precedes each section to identify the material contained therein.

The Alphabetical Index lists both the text and illustration coverage of this handbook by subject matter arranged in alphabetical order, together with the related page number. For ease in reference, illustrations have been listed in italics.

The aircraft described in this handbook may be identified by the following Effectivity Coding Chart:

Aircraft No.	*BuAer Serial No.*	*Aircraft No.*	*BuAer Serial No*
A	17175	29	17160
B	39097	30	99857
C	12412	31	17123
D	17103	32	17216
1	17273	33	17122
2	17111	34	17127
3	12422	35	50772
4	17177	36	39096
5	17116	37	99853
6	50762	38	17098
7	17171	39	17165
8	17124	40	17156
9	17150	41	17241
10	12420	42	39061
11	39109	43	17153
12	12445	44	17179
13	39087	45	17166
14	12419	46	12410
15	50823	47	50833
16	17255	48	17169
17	12438	49	12440
18	50804	50	17194
19	50835	51	12435
20	50821	52	39080
21	17219	53	39084
22	50834	54	50801
23	39104	55	17196
24	39071	56	50812
25	12443	57	17281
26	12439	58	39070
27	50838	59	50796
28	17102	60	99845

Aircraft No.	*BuAer Serial No.*	*Aircraft No.*	*BuAer Serial No.*	*Aircraft No.*	*BuAer Serial No.*	*Aircraft No.*	*BuAer Serial No.*
61	50826	70	17287	79	17092	88	17258
62	17152	71	17154	80	39072	89	17211
63	17190	72	17248	81	17149	90	17242
64	50782	73	17182	82	17188	91	12437
65	17284	74	39064	83	17140	92	50808
66	17158	75	17253	84	50786	93	12431
67	17108	76	17119	85	17270	94	12425
68	12428	77	17191	86	50836	95	39081
69	17097	78	50784	87	12441	96	50780

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For listing of available material and details of distribution see Naval Aeronautics Publications Index NavAer 00-500.

SECTION I

GENERAL INFORMATION

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1103

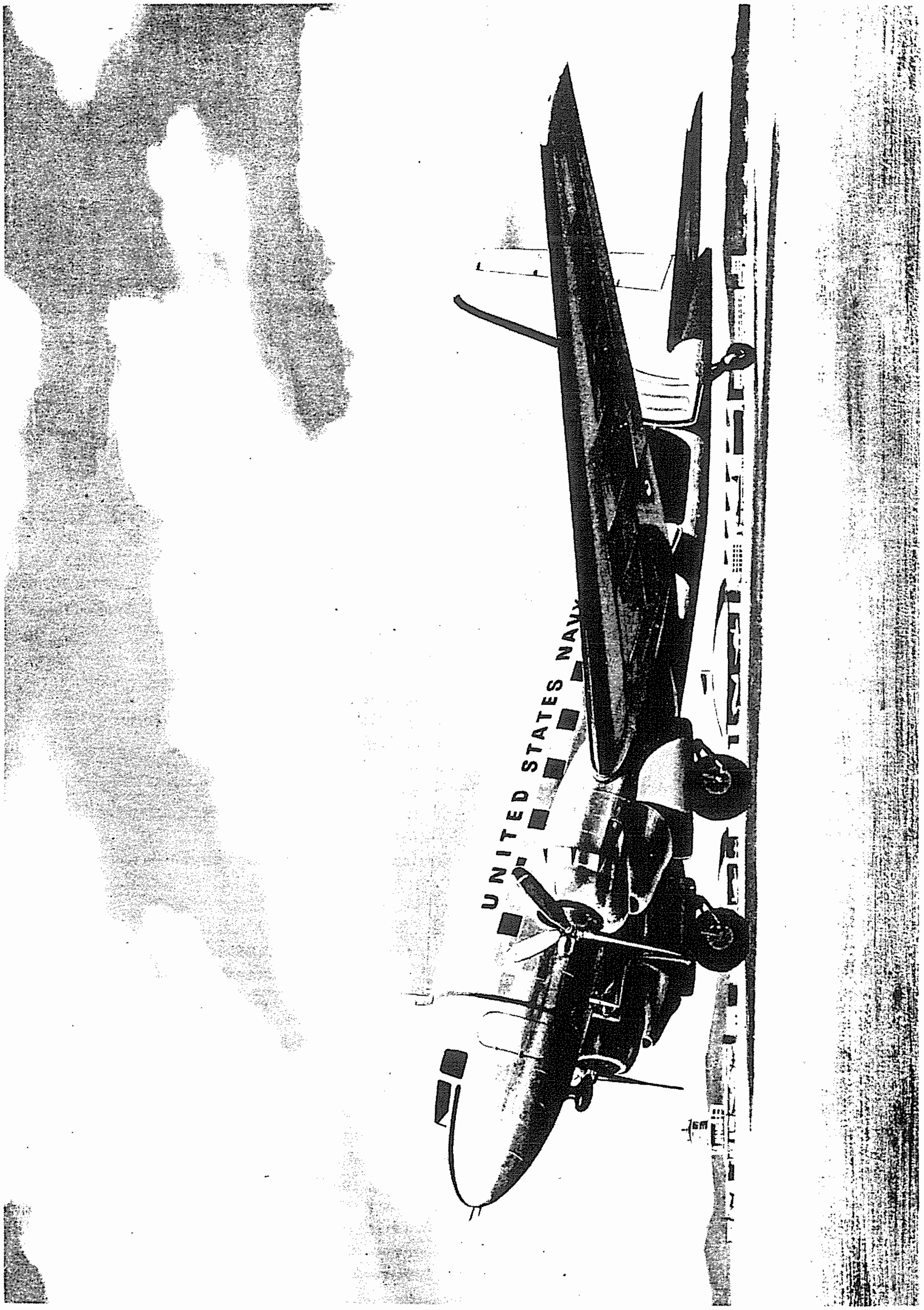

Figure 1-1. The Aircraft

SECTION I

GENERAL INFORMATION

1-1. GENERAL DESCRIPTION.

1-2. The R4D-8 and R4D-8Z aircraft are low-wing monoplanes with full cantilever wings and tail surfaces, semi-monocoque fuselage, and retractable alighting gear. Aircraft 1 through 96 are designed for use as diversified personnel, cargo, or ambulance transports; aircraft A through D are designed for use as staff transports. The aircraft are powered by two R-1820-80 engines which are equipped with Hamilton Standard hydromatic propellers. For graphic presentations of the aircraft and the aircraft stations, see figures 1-1 through 1-7.

1-3. In aircraft 1 through 96, accommodations are provided for a regular crew of three: pilot, co-pilot, and radio operator. Folding troop benches are installed along both sides of the main cargo compartment to accommodate 35 troops. Thirty collapsible high-density personnel seats can be installed in place of the folding-type troop benches. Litters can be installed as an alternate to the folding benches or personnel seats, to accommodate 27 ambulance patients *(see figure 1-8).*

1-4. In aircraft A through D, the interior arrangement, forward of station 97, is identical to aircraft 1 through 96. Aft of station 97, a table, desk, buffet, and a lavatory are installed for staff use. On aircraft A and B, 4 divans and 4 reclining swivel chairs are also installed, aft of station 97; on aircraft C and D, 2 divans and 8 reclining swivel chairs are installed, aft of station 97. Two bunks are installed in the aft part of the fuselage, between stations 538 and 623, for the use of 2 crew members: for interior arrangement, see figure 2-40.

1-5. The weight empty of the aircraft is approximately 20,400 pounds. The design gross weight is 31,000 pounds, and the overload gross weight is 36,800 pounds. For further information, refer to the applicable handbook of weight and balance instructions.

1-6. AIRCRAFT BALANCE.

1-7. The forward limit of the cg travel, with the alighting gear retracted, is 7 per cent of the MAC and the aft limit is 36 per cent of the MAC. The forward limit of the cg travel, with the alighting gear extended, is 10 per cent of the MAC and the aft limit is 37 per cent of the MAC.

1-8. AIRCRAFT DIMENSIONS.

1-9. DESCRIPTION.

Span .... 90 feet 0.3 inch
Length (overall) .... 67 feet 8.5 inches
Height .... 18 feet 3 inches

1-10. WING GROUP.

Span:
- Ailerons (each side) .... 277.0 inches
- Flaps (each side) .... 261.5 inches

Areas:
- Wings (gross, including ailerons) .... 967.1 square feet
- Ailerons (total, aft of hinge line) .... 71.7 square feet
- Flaps (total, both sides) .... 92.5 square feet
- Aileron trim tab, geared (each side) .... 2.7 square feet

Chords:
- Wing:
  - Chord at root (theoretical) .... 170.0 inches
  - Chord at tip (theoretical) .... 57.0 inches
- Flap:
  - Chord .... 25.5 inches
- Aileron:
  - Chord (average aft of hinge line, per cent chord line) .... 15.6 degrees

Wing taper ratio (outer panel) 0.347

Wing incidence at zero lift line:
- Root .... +4 degrees
- Horizontal station 142.0 .... +4 degrees
- Tip .... +0.2 degree

Wing dihedral, outer panel (reference plane) .... 6.8 degrees

Wing sweepback, outer panel (25 per cent chord line) .... 15.6 degrees

Wing aspect ratio 8.38

Wing MAC .... 139.7 inches

1-11. HORIZONTAL TAIL SURFACES.

Span .... 38 feet 0 inches

Areas:
- Horizontal surface (gross) .... 289.5 square feet
- Horizontal stabilizer (total less movable surface area) .... 158.5 square feet
- Elevator (total, aft of hinge line) .... 95.0 square feet
- Control tab (both sides) .... 7.8 square feet
- Trim tab (both sides) .... 9.4 square feet

Chords:
- Root (theoretical) .... 131.0 inches
- Tip (theoretical) .... 52.5 inches

Dihedral .... 0 degrees

Sweepback (25 per cent chord line) .... 7.5 degrees

Aspect ratio 4.99

Horizontal stabilizer incidence of zero lift line .... 0 degrees

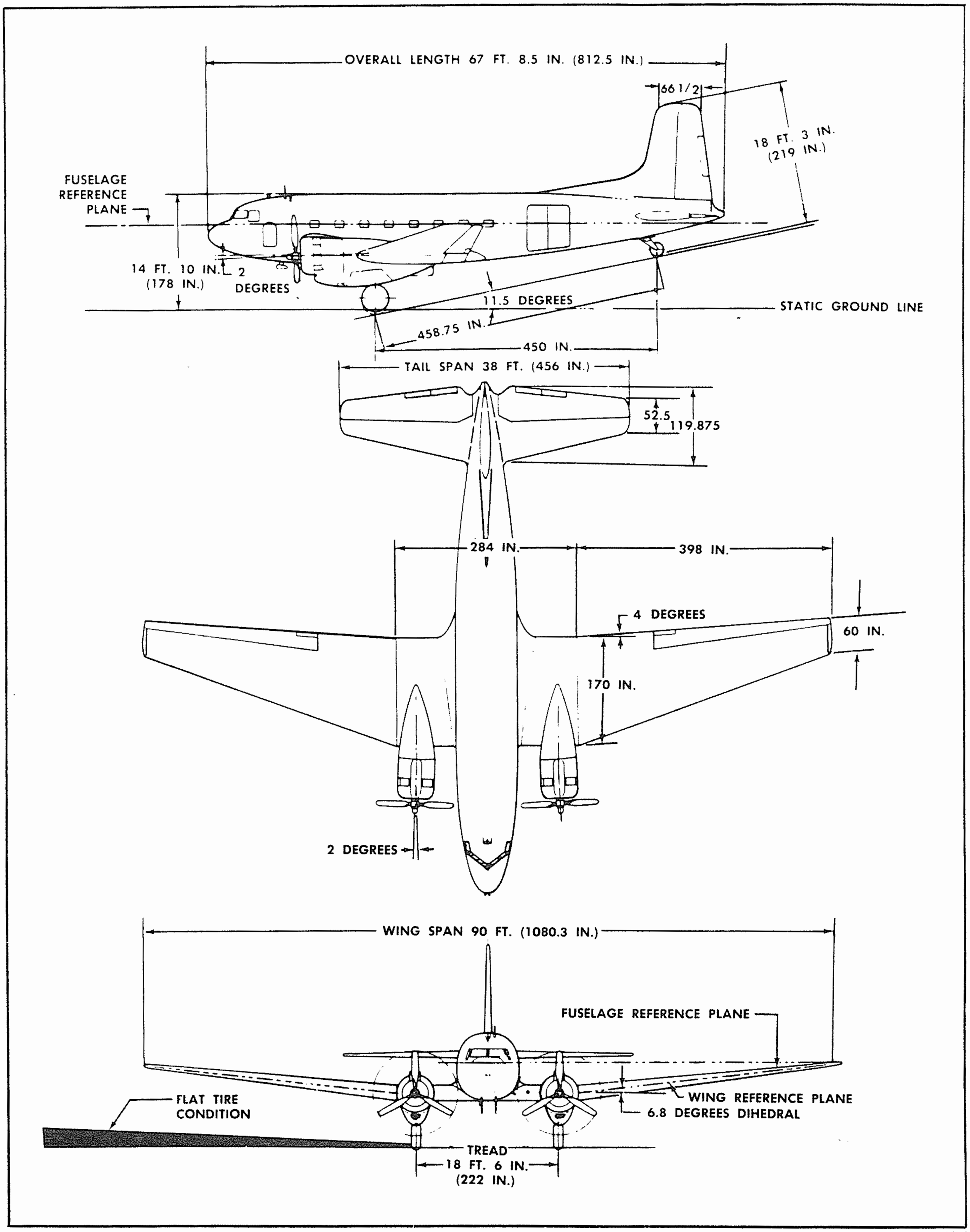

Figure 1-2. Three-View of Aircraft

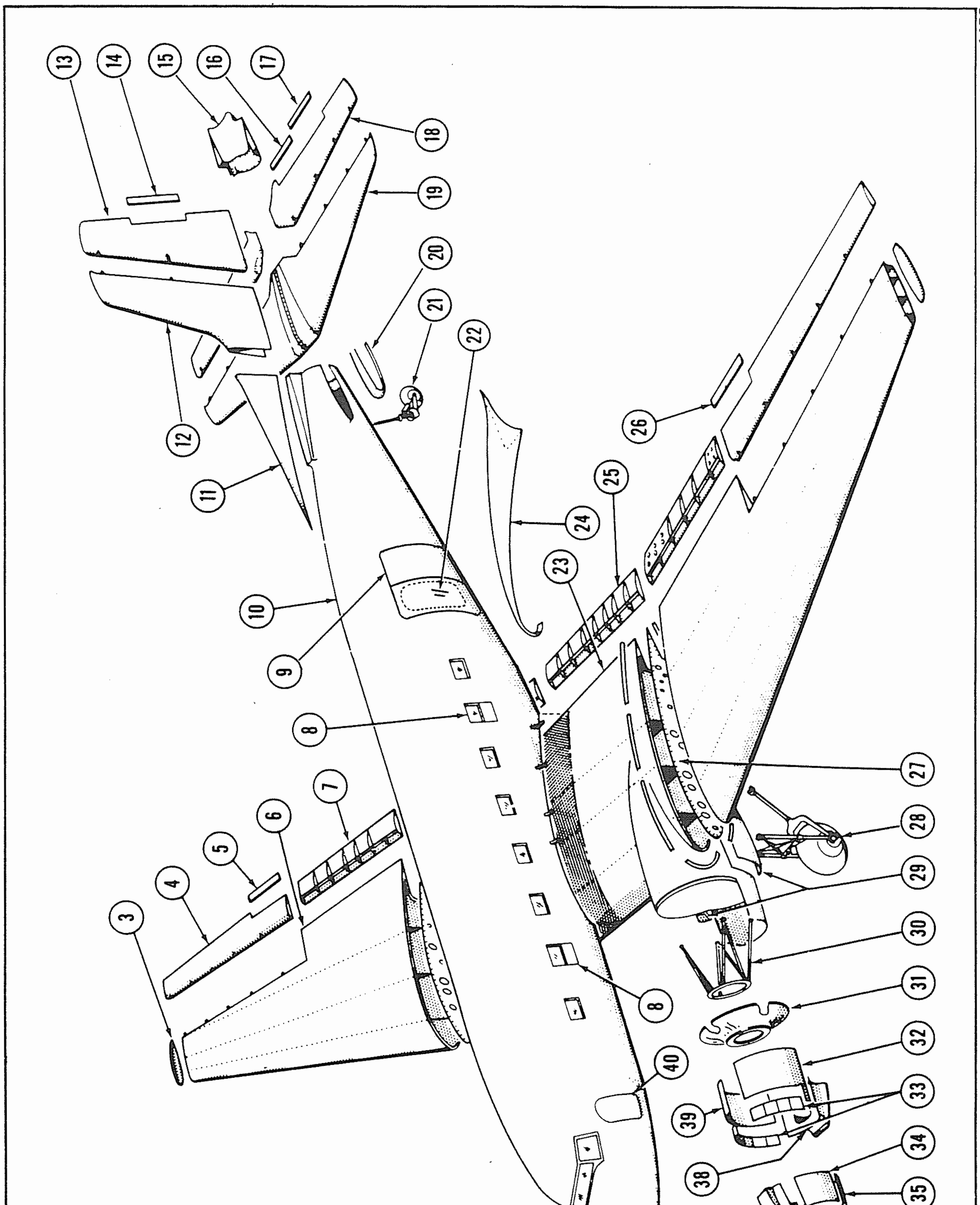

Figure 1-3. Exploded View of Aircraft

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z AIRCRAFT

SECTION I GENERAL INFORMATION

THIS SECTION SUPERSEDES SECTION I OF AN 01-40NK-2
DATED 15 MAY 1952 REVISED 1 NOVEMBER 1952

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE
AND THE CHIEF OF THE BUREAU OF AERONAUTICS

15 April 1953

Key to Figure 1-3

1. Nose Cover—Upper Section
2. Nose Cover—Lower Section
3. Outer Wing Tip
4. Aileron
5. Aileron Trim Tab
6. Outer Wing Panel
7. Outer Wing Flap
8. Auxiliary Exit (4)
9. Main Cargo Door (Aircraft 1 through 96)
10. Fuselage
11. Dorsal Fin
12. Vertical Stabilizer
13. Rudder
14. Rudder Tab
15. Tail Fairing Assembly
16. Elevator Spring Control Tab
17. Elevator Trim Tab
18. Elevator
19. Horizontal Stabilizer
20. Horizontal Stabilizer Fairing
21. Tail Wheel Assembly
22. Main Cabin Door (Aircraft A through D)
23. Center Wing
24. Wing-to-Fuselage Fillet
25. Center Wing Flap
26. Aileron Spring Control Tab
27. False Rib
28. Main Gear
29. Main Gear Doors
30. Engine Mount
31. Inner Ring
32. Accessory Cowling Outboard Section
33. Cowl Flaps
34. Anti-Drag Ring Outboard Segment
35. Anti-Drag Ring Bottom Segment
36. Propeller
37. Anti-Drag Ring Inboard Segment
38. Accessory Cowling Bottom Section
39. Accessory Cowling Inboard Section
40. Service Door

1-12. VERTICAL TAIL SURFACES.

Span ....174.0 inches
Areas:
Vertical surface (total exposed excluding dorsal fin) ....103.8 square feet
Fin (total exposed, excluding dorsal fin) .... 53.9 square feet
Rudder (aft of hinge line).................. 36.8 square feet
Trim tab, geared .... 4.0 square feet
Chords:
Root (theoretical) ....127.6 inches
Tip (theoretical) .... 67.6 inches
Sweepback ....7.5 degrees

1-13. FUSELAGE.

Height (maximum cross section)........................ 106 inches
Width (maximum cross section).........................97.3 inches

1-14. LEADING PARTICULARS.

1-15. ALIGHTING GEAR.

Main gear—standard:
Type..........................Hydraulically actuated, retractable, dual-shock strut
Tread ....18 feet 6 inches
Wheel base ....37 feet 9 inches
Shock strut....................Oleopneumatic, Douglas 2390420
Fluid required MIL-O-5606
Wheels................17.00 x 16, Type III, Goodyear 9540547
Tires..................................17.00 x 16, 12-ply rating, nylon Goodrich or equivalent
Tire pressure for maximum take-off gross weight .... 60 psi
Brakes...Goodyear, single-disc, spot
Tail wheel gear:
Type............................Hydraulically actuated, retractable
Shock strut....................Oleopneumatic, Douglas 3367199
Fluid required MIL-O-5606

Tail wheel gear *(Continued)*:
Wheel.............9.00 x 6, General 204-A-204M-1, Firestone XSO 200 FM, Goodrich B-3-648, Bendix 52058
Tire..9.00 x 6, 10-ply rating, nylon; Goodrich or equivalent
Tire pressure ....70 psi

1-16. ENGINES.

Number 2
Type.................Wright-Cyclone R-1820-80, 9-cylinder, air-cooled radial, single-speed supercharger
Propeller-to-engine gear ratio................................0.5625:1
Engine supercharger gear ratio....................................7.21:1
Fuel specification ....MIL-F-5572, Grade 100/130
Fuel specification, alternate permissible............MIL-F-5572, Grade 115/145
Oil specification ....MIL-O-6082, Grade 1100 (summer and winter)

1-17. PROPELLERS.

Type ....Hamilton Standard Hydromatic
Hub 23E50
Blade type 6615A-0
Diameter..............................11 feet 6¼ inches (± ¼ inch)
Number of blades 3
Propeller settings at station 42.0
Low pitch ....18 degrees
High pitch ....None
Feather ....88 degrees

1-18. TANK CAPACITIES.

Fuel tanks:
Front (2, left and right)........202 US. (168.19 Imp.) gallons each
Rear (2, left and right)........200 US. (166.52 Imp.) gallons each
Outer wing (5, left and right)............411 US. (342.20 Imp.) gallons each set
Total.................1626 US. (1353.81 Imp.) gallons

(Continued on Page 9)

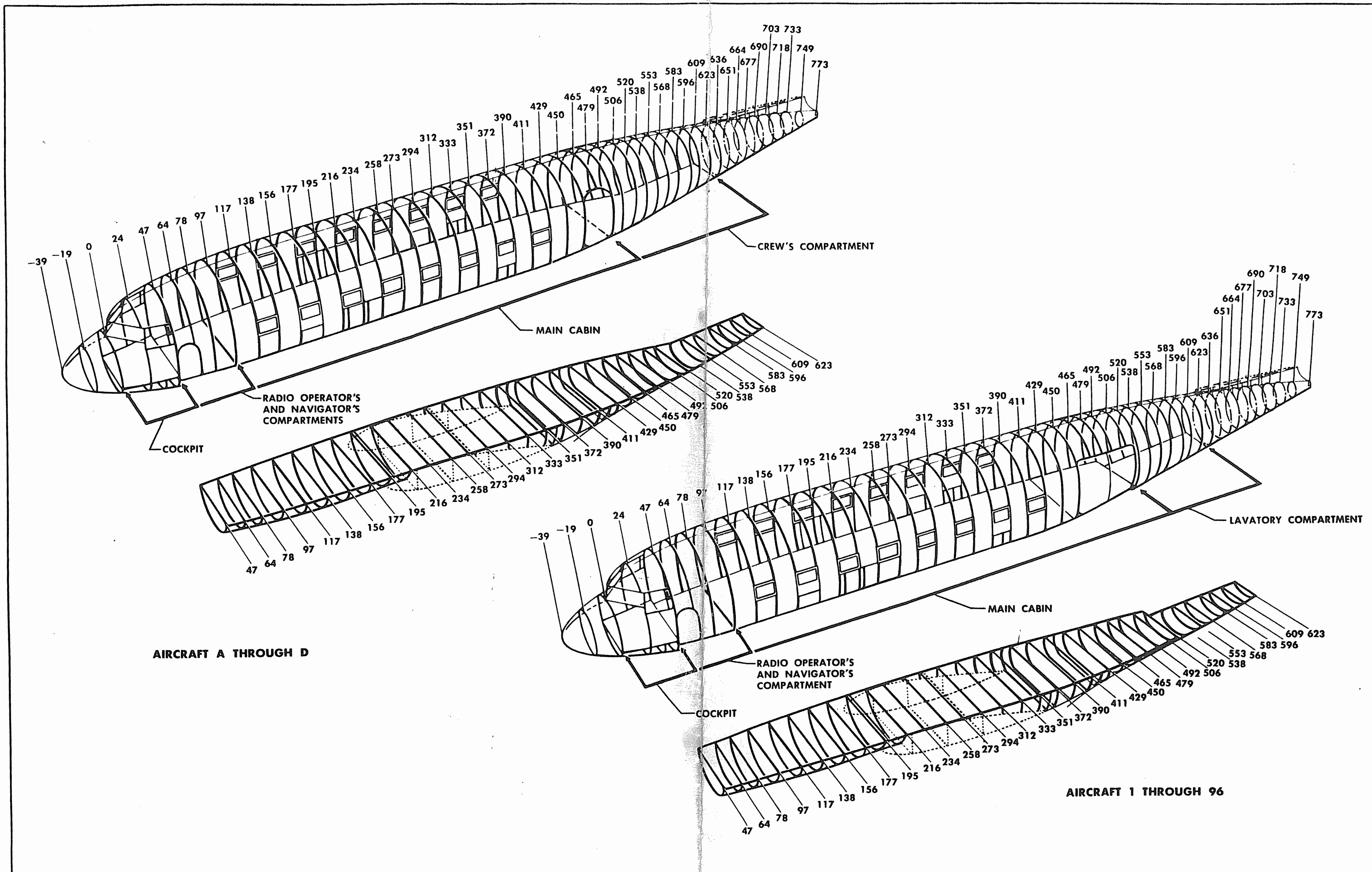

Figure 1-4. Fuselage Stations

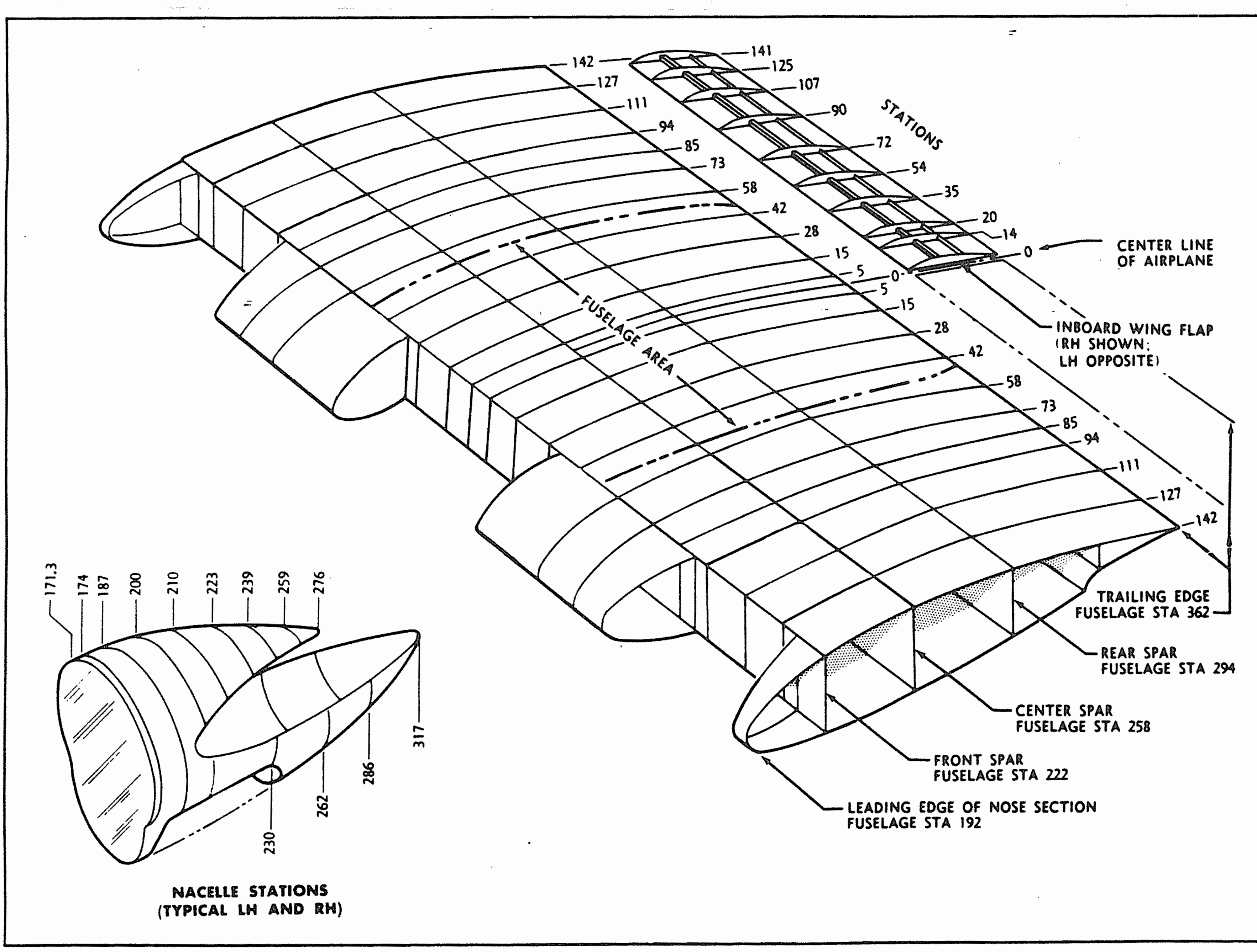

Figure 1-5. Wing Center Section and Nacelle Stations

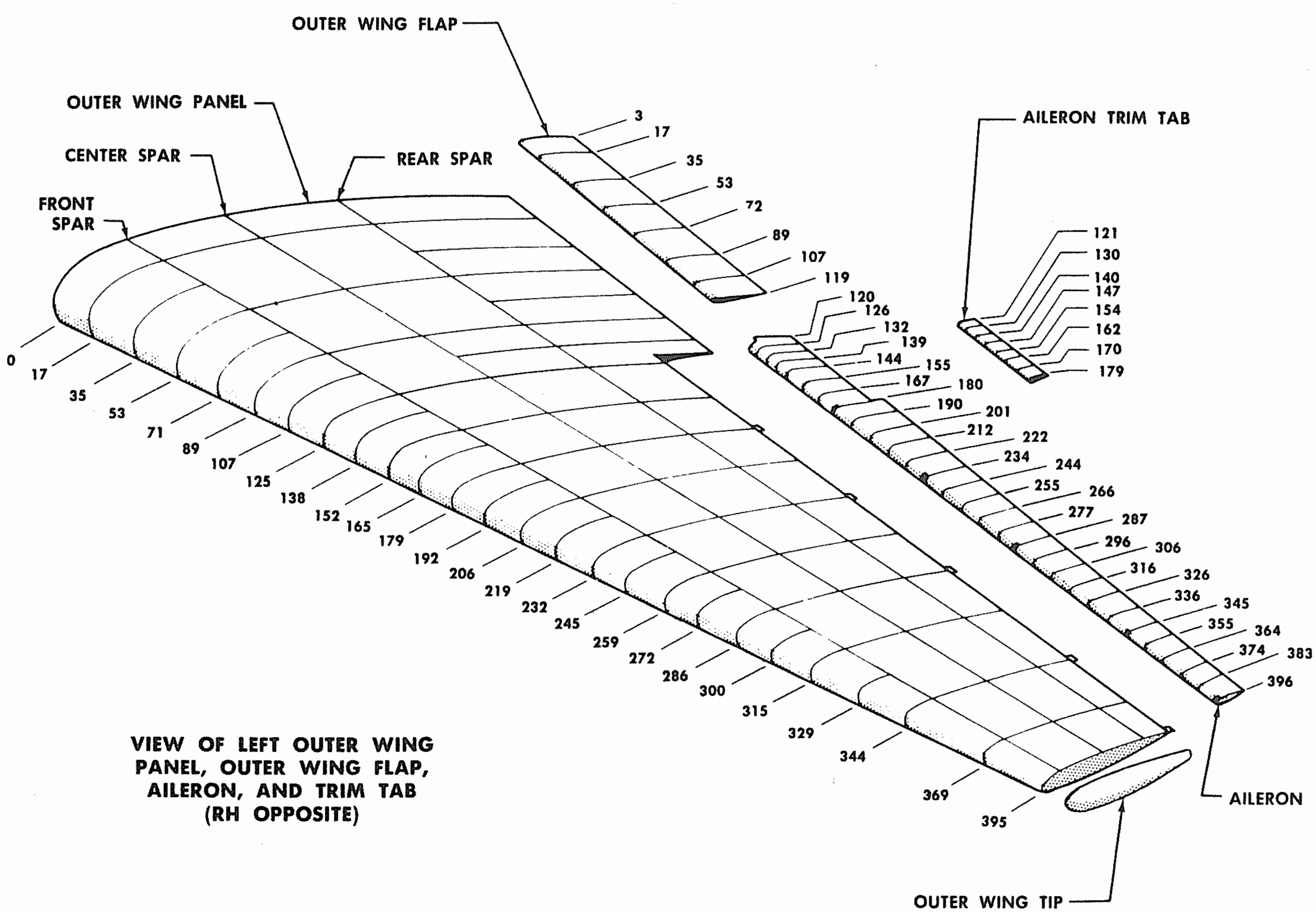

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Figure 1-6. Wing Outer Panel Stations

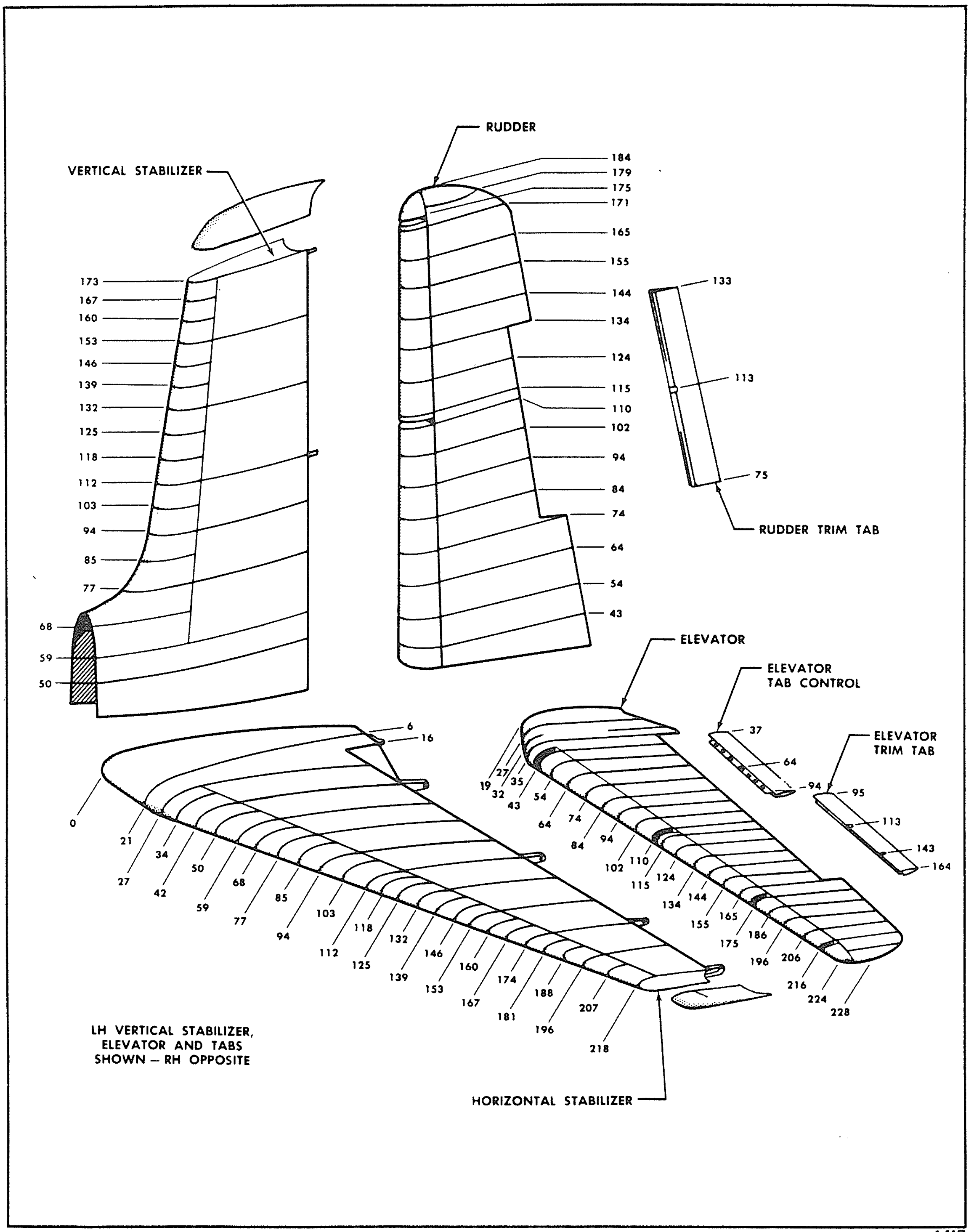


Figure 1-7. Empennage Stations

(Continued from Page 4)

Oil tanks:
- Nacelle (1 for each engine)....27¾ US. (23.11 Imp.) gallons each
- Total ....55½ US. (46.21 Imp.) gallons

Hydraulic fluid:
- Fluid ....Specification MIL-O-5606
- Total system capacity ....6 US. (5.0 Imp.) gallons
- Reservoir capacity ....2.3 US. (1.91 Imp.) gallons
- Reserve ....3.2 US. (2.67 Imp.) gallons

Anti-icing alcohol tank:
- Fluid.............................Isopropyl alcohol conforming to Specification MIL-F-5566
- Tank capacity ....20 US. (16.65 Imp.) gallons

Water tank:
- Wash basin supply (aircraft 1 through 96) ....3 US. (2.5 Imp.) gallons
- Lavatory and buffet supply (aircraft A through D) 2 tanks ....7 US. (5.83 Imp.) gallons each
- Total ....14 US. (11.66 Imp.) gallons

1-19. ACCESS AND INSPECTION PROVISIONS.

1-20. INTERIOR ACCESS DOORS. *(See figure 1-9.)* Access doors and cover plates are provided in the interior structure of the aircraft to facilitate inspection, maintenance, and repair of controls, system components, and structural sections.

1-21. EXTERIOR ACCESS DOORS. *(See figure 1-9.)* Access doors and cover plates are provided in the exterior surfaces of the wing, nacelle, fuselage and empennage to facilitate inspection, maintenance, and general repair.

1-22. DRAIN AND VENT HOLES. *(See figure 1-10.)* Draining provisions are installed throughout the wing and fuselage structures for servicing the various fluid systems. Drain holes are provided in the flight control surfaces to permit the escape of condensation. Vent holes are provided for those systems requiring a supply of air for proper operation.

1-23. GROUND OPERATION.

1-24. ENGINE OPERATION.

1-25. COWL FLAPS. The cowl flaps must be fully open, regardless of outside air temperatures, during all ground operations, including starting, warm-up, ground testing, and taxiing.

1-26. PRIMING. The priming system functions as an aid in starting the engines and, when used properly, reduces the hazard of flooding. To avoid overloading a warm engine, 15°C (59°F) or more, it is advisable to flick the primer switch intermittently. The actual amount of priming necessary will depend upon the engine installation, weather conditions, and air temperatures. Do not prime until the starter is actually cranking the engine.

1-27. EXCESSIVE PRIMING. Excessive priming will load the cylinders with raw fuel, making starting difficult and tending to wash the oil film from the cylinder walls with the consequent danger of cylinder scoring or piston seizure. If the engine is allowed to stand for a day or more following unsuccessful attempts to start the engine, and if the cylinder surfaces are not protected by a fresh application of oil, rusting of piston rings and cylinder walls will frequently occur. Continuous use of the primer may cause the fuel to drain back through the intake pipes of primed cylinders and collect in the intake pipes of bottom cylinders without indication of fuel leakage from the supercharger drain valve. The drain valve will not indicate fuel collection forward of the supercharger impeller, or in the lower intake pipes, since it drains only from the area of the impeller inlet in the rear supercharger housing.

1-28. INSUFFICIENT PRIMING. Insufficient priming during the starting operation is usually indicated by backfiring, with attendant fire hazards. Cold-weather starting requires more priming and a slightly greater throttle opening than would be required when starting in moderate or warm weather.

1-29. BACKFIRING. When an engine backfires, indicating too lean a mixture, it is recommended that the priming system be used to assist in keeping the engine running until fuel from the carburetor reaches all cylinders. In the event of backfiring, do not shut down, but continue cranking with the starter until backfiring ceases and the engine is running.

1-30. OVER-RICH MIXTURE. An engine receiving too rich a mixture to continue running during a start, can be revived by momentarily moving the mixture control to the IDLE CUT-OFF position. This will lean out the mixture, after which the mixture control should be returned to the AUTO-RICH or FULL-RICH position.

1-31. FUEL BOOSTER PUMPS. The engine must be started with the booster pumps ON. Refer to the applicable handbook of flight operating instructions for recommended use.

1-32. FUEL OVERLOADING. Overloading of a warm engine may be indicated by a discharge of fuel from the drain in the lower part of the supercharger. In this case, do not attempt to start the engine until fuel has ceased to flow from the drain. Pull the propeller through 6 to 8 revolutions (18 to 24 blades) using the starter, in the direction of rotation, with the ignition OFF, the mixture control in IDLE CUT-OFF, and the throttle wide open. Then repeat the starting procedure. The most frequent cause of overloading is failure to return the mixture control to IDLE CUT-OFF position immediately after a false start.

Note

The mixture control must not be moved out of the IDLE CUT-OFF position before the engine starts firing from fuel used in priming.

(Continued on Page 18)

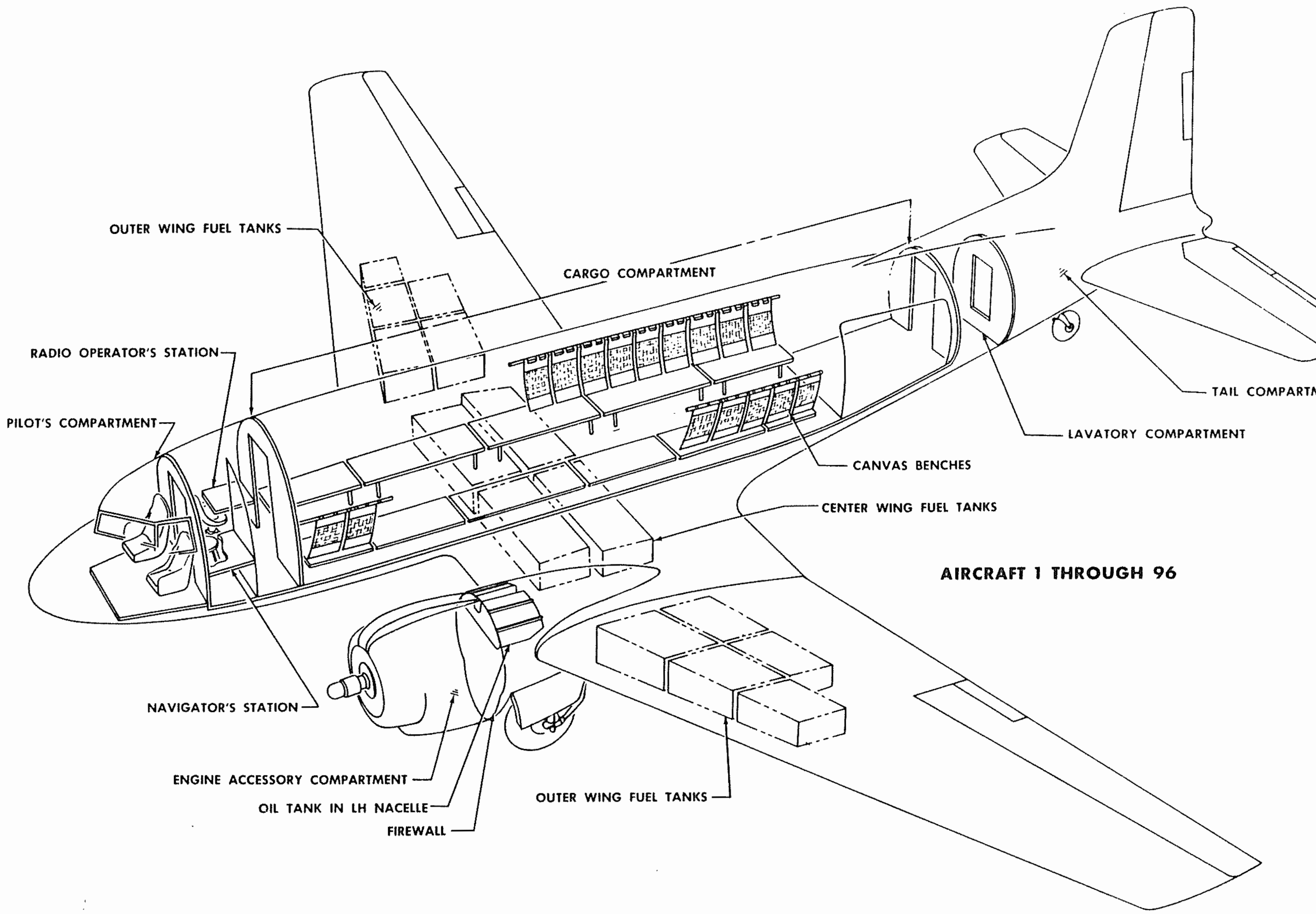


Figure 1-8. Location of Compartments and Areas

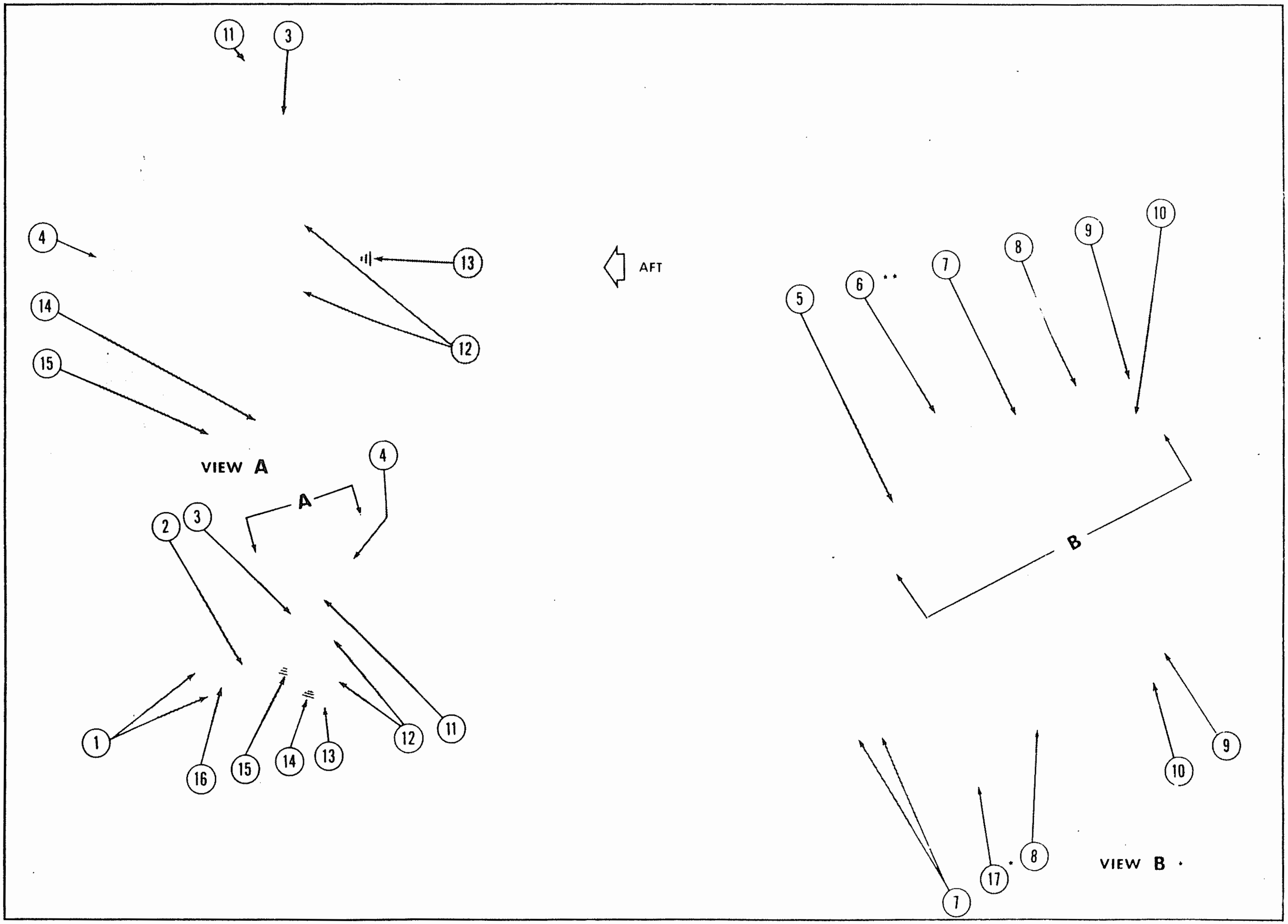


Figure 1-9 (Sheet 1 of 5 Sheets). Access Doors — Interior of Aircraft

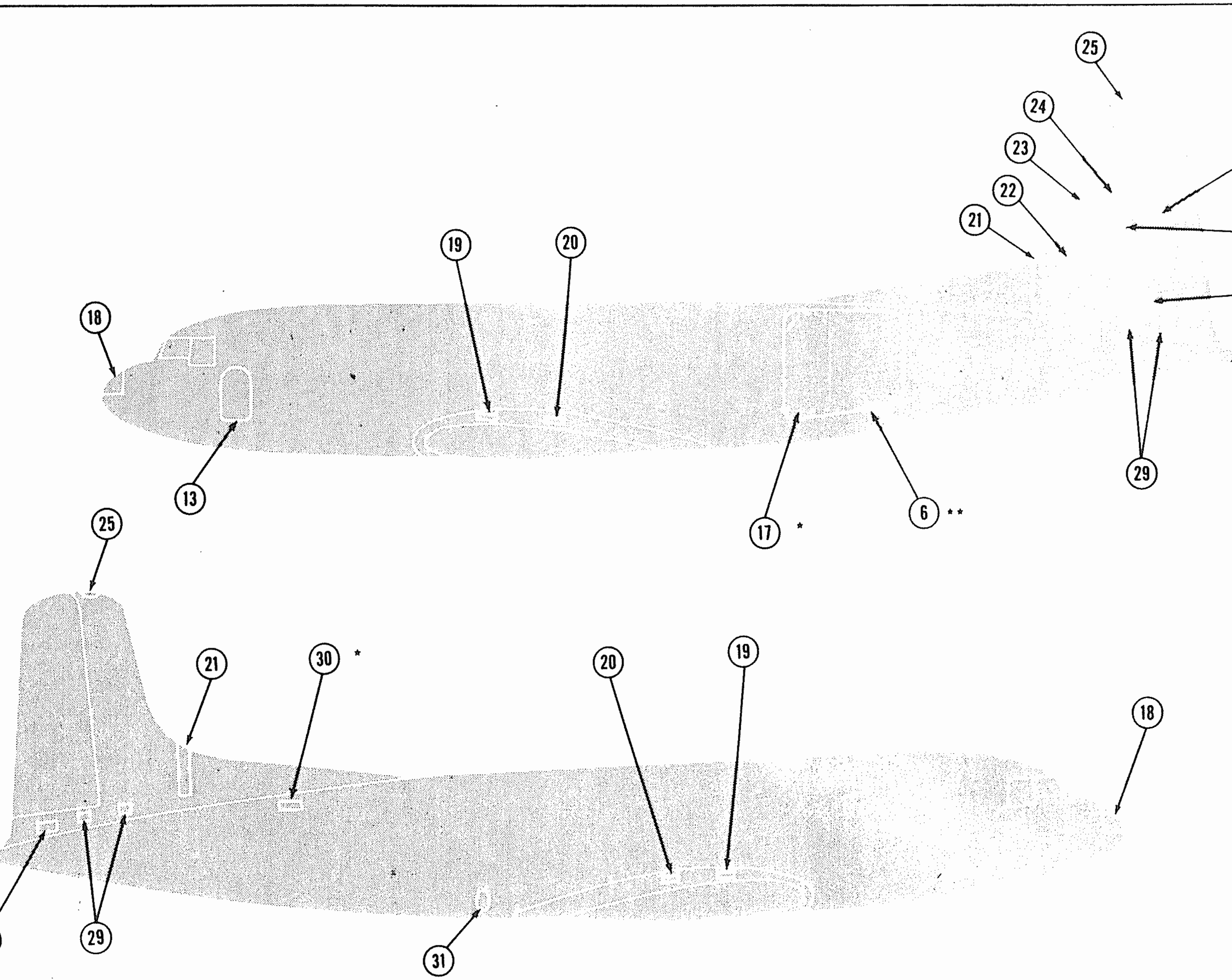


Figure 1-9 (Sheet 2 of 5 Sheets). Access Doors — Fuselage and Empennage, Left and Right Sides

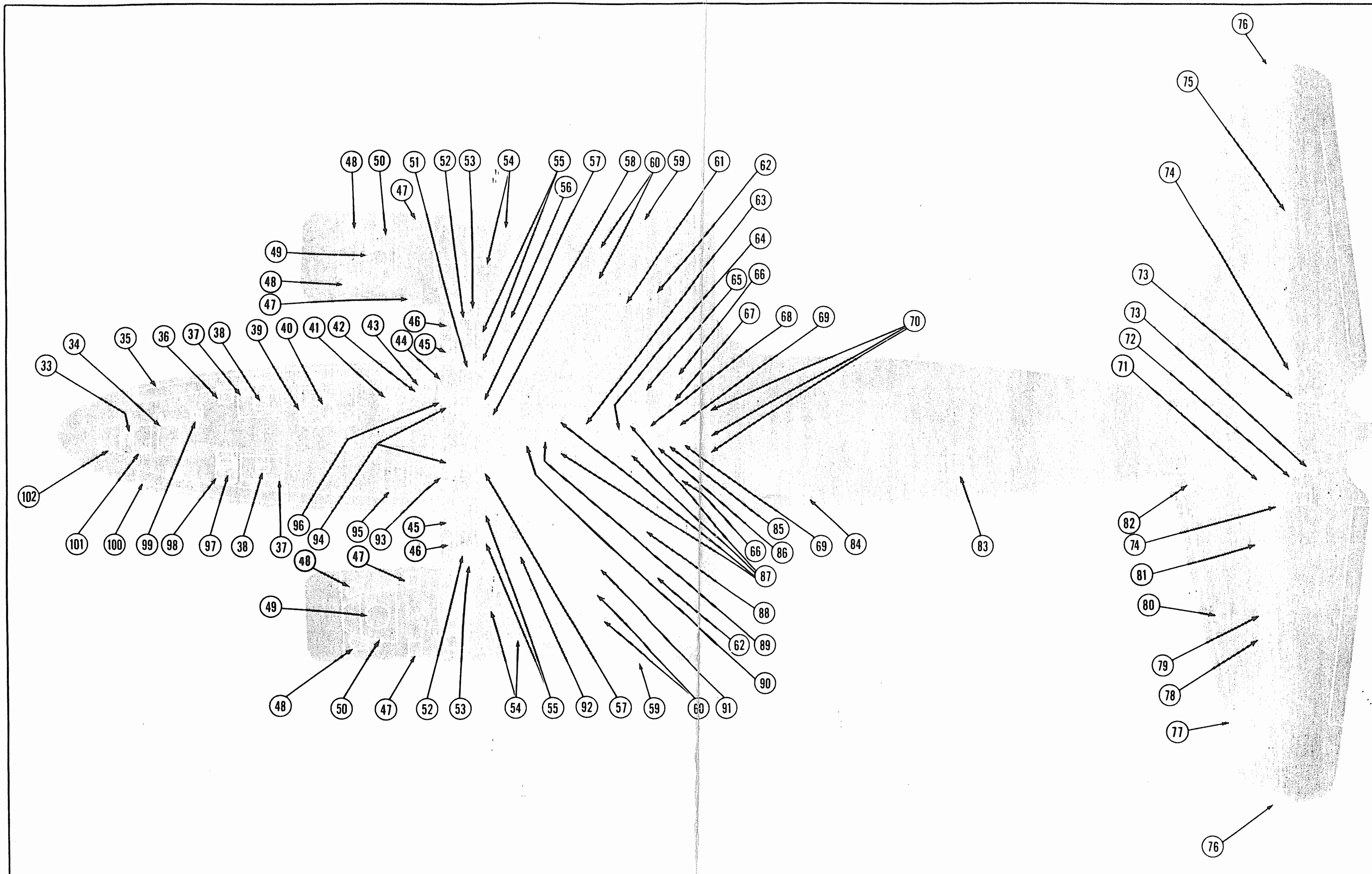


Figure 1-9 (Sheet 3 of 5 Sheets). Access Doors—Bottom View of Aircraft

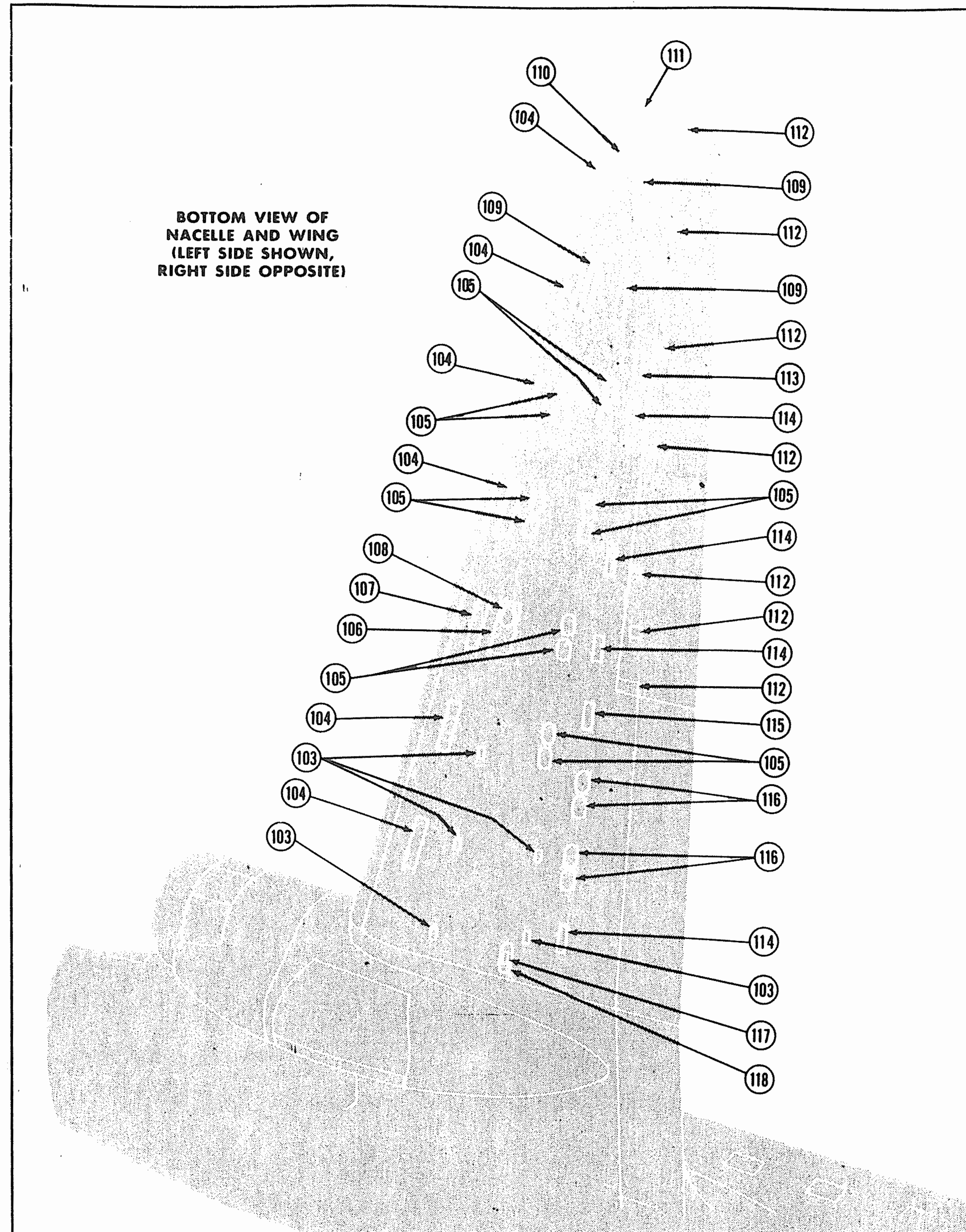


Figure 1-9 (Sheet 4 of 5 Sheets). Access Doors — Bottom View of Left Wing

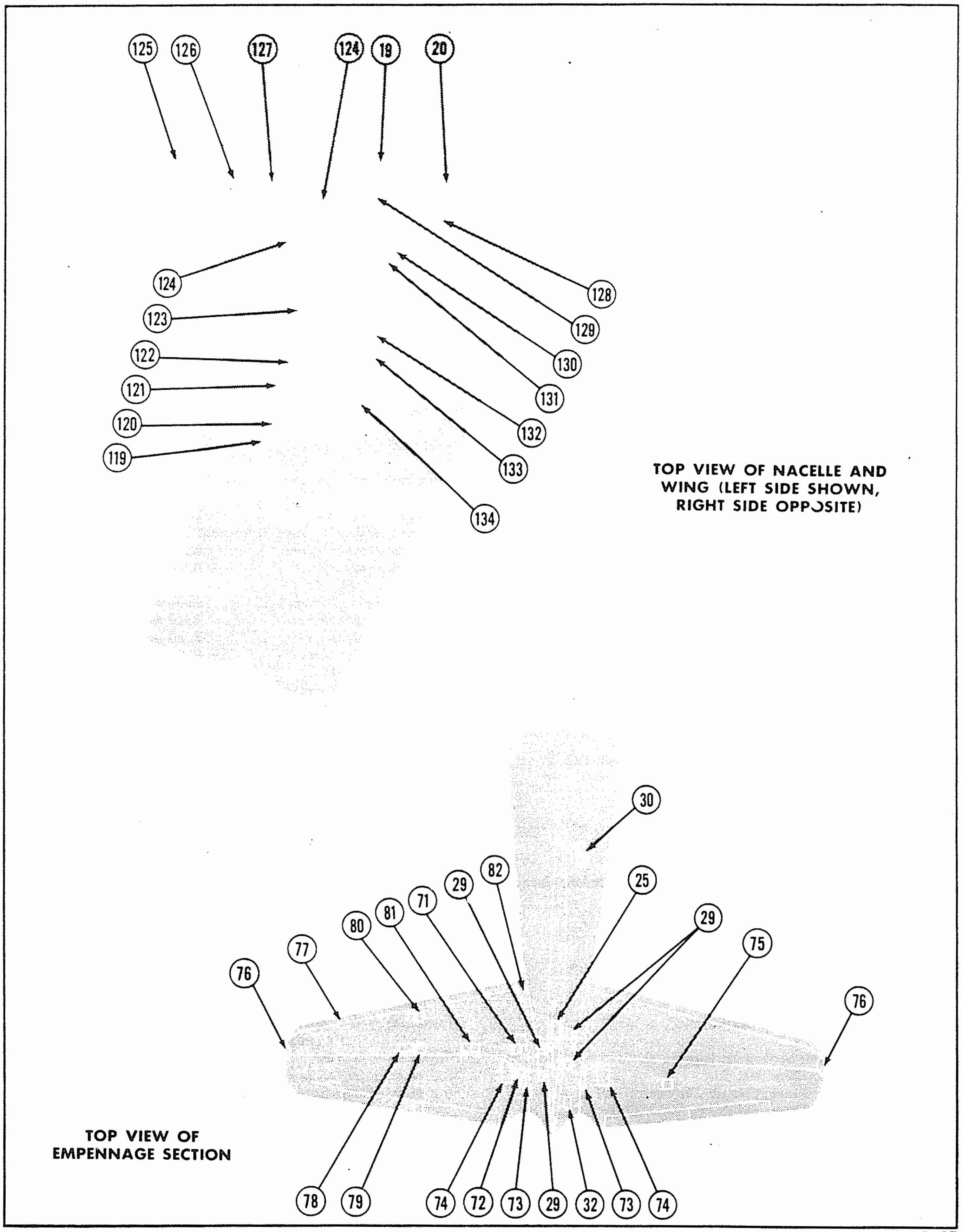


Figure 1-9 (Sheet 5 of 5 Sheets). Access Doors — Top Views of Left Wing and Empennage

Key to Figure 1-9

Access Door No.	Access to	Type of Door	Attaching Parts	Tool Used
1.	Control Surface Cables, Engine Control Cables	Nonstress	Screw	Screw Driver
2.	Fire Control Valves, Emergency Shutoff Valves	Nonstress	Snap	Hand Operated
3.	Voltage Regulators	Nonstress	Camloc Fastener	Screw Driver
4.	Flight Compartment	Nonstress	Handle	Hand Operated
5.	Control Surface Cables	Nonstress	Screw	Reed & Prince Screw Driver
* 6.	Main Cargo Door	Stress	Handle	Hand Operated
7.	Rear Lounge Area	Nonstress	Handle	Hand Operated
8.	Rear Compartment Door	Nonstress	Handle	Hand Operated
9.	Control Surface Cables, Elevator Torque Tube, Horizontal Stabilizer Attach Bolts	Nonstress	Handle	Hand Operated
10.	Control Cables	Nonstress	Camloc Fastener	Screw Driver
11.	Circuit Breakers	Nonstress	Screw	Screw Driver
12.	Main Junction Box (Upper Electrical, Lower Radio)	Nonstress	Screw	Screw Driver
13.	Service Door	Nonstress	Handle	Hand Operated
14.	Inverters	Nonstress	Camloc Fastener	Screw Driver
15.	Control Cables, Hydraulic Piping	Nonstress	Camloc Fastener	Screw Driver
16.	Control Cables, Electrical Wiring	Stress	Screw	Screw Driver
†17.	Main Cabin Door	Stress	Handle	Hand Operated
18.	Rudder Torque Tube, Brake Torque Tube, Instruments	Nonstress	Screw	Reed & Prince Screw Driver
19.	Auxiliary Fuel Tank Liquidometer, Vent Pipes, Fuselage-to-Wing Attach Fittings	Nonstress	Screw	Reed & Prince Screw Driver
20.	Main Fuel Tank Liquidometer, Vent Pipes, Fuselage-to-Wing Attach Fittings	Nonstress	Screw	Reed & Prince Screw Driver
21.	Vertical Stabilizer Attach Bolts, Dorsal Fin Attach Bolts, Surface De-Icer Piping, Electrical Conduit	Nonstress	Screw	Reed & Prince Screw Driver
22.	Flux Gate Compass	Nonstress	Camloc Fastener	Screw Driver
23.	Structure	Nonstress	Screw	Reed & Prince Screw Driver
24.	Rudder Trim Control Cable Drum	Nonstress	Screw	Reed & Prince Screw Driver
25.	Antenna Disconnect, Vertical Stabilizer Tip Attach Bolts	Nonstress	Screw	Reed & Prince Screw Driver
26.	Rudder Hinge and Tab Control Rod	Nonstress	Screw	Reed & Prince Screw Driver
27.	Rudder Trim Control Cable	Nonstress	Screw	Reed & Prince Screw Driver
28.	Rudder-to-Rudder Torque Tube Attach Bolts	Nonstress	Screw	Reed & Prince Screw Driver
29.	Rudder Control Cables, Elevator Control Cables	Nonstress	Snap	Hand Operated
†30.	Fresh Water Panel	Nonstress	Screw	Reed & Prince Screw Driver
31.	Alcohol Tank Filler Neck	Nonstress	Spring Latch	Hand Operated
32.	Elevator Torque Tube	Nonstress	Screw	Screw Driver
33.	Control Surfaces, Engine Controls	Nonstress	Screw	Reed & Prince Screw Driver
34.	Control Surfaces, Engine Controls, CO_2 Panel	Nonstress	Screw	Reed & Prince Screw Driver
35.	Electrical Tunnel	Nonstress	Screw	Reed & Prince Screw Driver
36.	Cockpit Warm-Air Control Rod	Nonstress	Spring Latch	Hand Operated
37.	Battery Elevator	Nonstress	Dzus Fastener	Screw Driver
38.	Battery Compartment	Nonstress	Spring Latch	Hand Operated
39.	Cabin Heater Unit	Nonstress	Spring Latch	Hand Operated
40.	No. 39 Door Handle	Nonstress	Spring Latch	Hand Operated
41.	Automatic Pilot Servo Unit, Control Surface Cables, Engine Control Cables	Nonstress	Spring Latch	Hand Operated
42.	Ground Power Receptacle and Relay	Nonstress	Screw	Reed & Prince Screw Driver
43.	(See Door No. 42)	Nonstress	Spring-Loaded Hinged Door	Hand Operated
44.	Electrical Conduit, Ignition Junction Box	Nonstress	Screw	Reed & Prince Screw Driver
45.	Electrical Conduit, Structure	Stress	Screw	Reed & Prince Screw Driver
46.	Electrical Conduit, Structure	Stress	Screw	Reed & Prince Screw Driver
47.	Engine Accessory Section	Nonstress	Spring-Loaded Hinged Door	Hand Operated
48.	Lower Section Exhaust Stacks	Nonstress	Screw	Reed & Prince Screw Driver
49.	(See Door No. 50)	Nonstress	Screw	Reed & Prince Screw Driver
50.	Oil Cooler Temperature Regulator	Nonstress	Screw	Reed & Prince Screw Driver
51.	Engine Instrument Piping, CO_2 Piping, Hydraulic Piping, Emergency Air Brake Piping	Stress	Screw	Reed & Prince Screw Driver
52.	Engine Instrument Piping, CO_2 Piping, Hydraulic Piping, Emergency Air Brake Piping	Stress	Screw	Reed & Prince Screw Driver

*Aircraft 1 through 96.
†Aircraft A through D.

Key to Figure 1-9 (Continued)

Access Door No.	Access to	Type of Door	Attaching Parts	Tool Used
53.	Engine Instrument Piping, CO_2 Piping, Hydraulic Piping, Emergency Air Brake Piping; Nacelle-to-Wing Disconnects	Stress	Screw	Reed & Prince Screw Driver
54.	Main Alighting Gear Door	Nonstress	Hydraulically Operated	....
55.	Engine Control Cables, Alighting Gear Latch Cable, Surface De-Icer Piping	Nonstress	Spring Latch	Hand Operated
56.	Left Main Fuel Tank	Stress	Screw	Reed & Prince Screw Driver
57.	(See Door No. 58)	Nonstress	Spring Latch	Hand Operated
58.	Center Wing Fuel Selector Valve, Fuel Strainers, Surface De-Icer Distributor Valve, Fuel Piping, Engine Control Cables, Control Surface Cables, Surface De-Icer Piping	Stress	Screw	Reed & Prince Screw Driver
59.	Aileron Control Cables	Stress	Screw	Reed & Prince Screw Driver
60.	Outer Wing Fuel Sump, Outer Wing Fuel Hose, Hydraulic Piping, Emergency Air Brake Piping	Nonstress	Screw	Reed & Prince Screw Driver
61.	Left Auxiliary Fuel Tank	Stress	Screw	Reed & Prince Screw Driver
62.	Aileron Control Cables	Nonstress	Snap	Hand Operated
63.	Fuel Piping, Hydraulic Piping, Alcohol Piping, Surface De-Icer Piping; Control Surface Cables, Elevator Tab Control Cable, Automatic Pilot Cable Drum	Stress	Screw	Reed & Prince Screw Driver
64.	(See Door No. 63)	Stress	Screw	Reed & Prince Screw Driver
65.	Aileron Control Cables	Nonstress	Screw	Reed & Prince Screw Driver
66.	Structure	Nonstress	Screw	Reed & Prince Screw Driver
67.	Tail Wheel Retract Hydraulic Piping, Structure	Nonstress	Screw	Reed & Prince Screw Driver
68.	(See Door No. 86)	Nonstress	Snap	Hand Operated
69.	Structure	Nonstress	Snap	Hand Operated
70.	Wing Flap Actuating Mechanism	Nonstress	Screw	Reed & Prince Screw Driver
71.	Elevator Trim Tab Control Cable	Nonstress	Screw	Reed & Prince Screw Driver
72.	Elevator Attach Bolts	Nonstress	Screw	Reed & Prince Screw Driver
73.	Elevator Spring Control Tab Actuating Rod	Nonstress	Screw	Reed & Prince Screw Driver
74.	Elevator Spring Control Tab Mechanism	Nonstress	Screw	Reed & Prince Screw Driver
75.	Elevator Trim Tab Actuating Crank	Nonstress	Screw	Reed & Prince Screw Driver
76.	Wing Tip Attach Bolts, Elevator Outboard Fitting	Nonstress	Screw	Reed & Prince Screw Driver
77.	Structure	Nonstress	Screw	Reed & Prince Screw Driver
78.	Elevator Trim Tab Cable Drum	Nonstress	Screw	Reed & Prince Screw Driver
79.	Elevator Trim Tab Control Cable	Nonstress	Screw	Reed & Prince Screw Driver
80.	Structure	Nonstress	Screw	Reed & Prince Screw Driver
81.	Elevator Trim Tab Control Cable	Nonstress	Screw	Reed & Prince Screw Driver
82.	Surface De-Icer Piping Disconnects	Nonstress	Screw	Reed & Prince Screw Driver
†83.	Toilet Service Panel	Nonstress	Spring Latch	Hand Operated
84.	Alcohol Tank Filter and Drain	Nonstress	Spring Latch	Hand Operated
85.	Wing Flap Actuating Cylinder, Control Cables	Nonstress	Screw	Reed & Prince Screw Driver
86.	Aileron Cross	Stress	Screw	Reed & Prince Screw Driver
87.	Fuel Tank Sump Drain	Nonstress	Camloc Fastener	Screw Driver
88.	Aileron and Aileron Tab Control Cables	Nonstress	Screw	Reed & Prince Screw Driver
89.	(See Door No. 90)	Stress	Screw	Reed & Prince Screw Driver
90.	Fuel Piping, Hydraulic Piping, Alcohol Piping, Surface De-Icer Piping; Control Surface Cables	Stress	Screw	Reed & Prince Screw Driver
91.	Right Auxiliary Fuel Tank	Stress	Screw	Reed & Prince Screw Driver
92.	Right Main Fuel Tank	Stress	Screw	Reed & Prince Screw Driver
93.	Disconnects for Engine Instrument Piping, CO_2 Piping and Emergency Air Brake Piping	Nonstress	Screw	Reed & Prince Screw Driver
94.	(See Door No. 96)	Nonstress	Spring Latch	Hand Operated
95.	Cabin Heater Control Unit	Nonstress	Spring Latch	Hand Operated
96.	Right and Left Outer Wing Fuel Selector Valve, Right and Left Outer Wing Fuel Booster Pump, Engine Control Cables, Control Surface Cables, Surface De-Icer Oil Separator, Cabin Heater Magnetic Valve	Nonstress	Screw	Reed & Prince Screw Driver

†Aircraft A through D.

Key to Figure 1-9 (Continued)

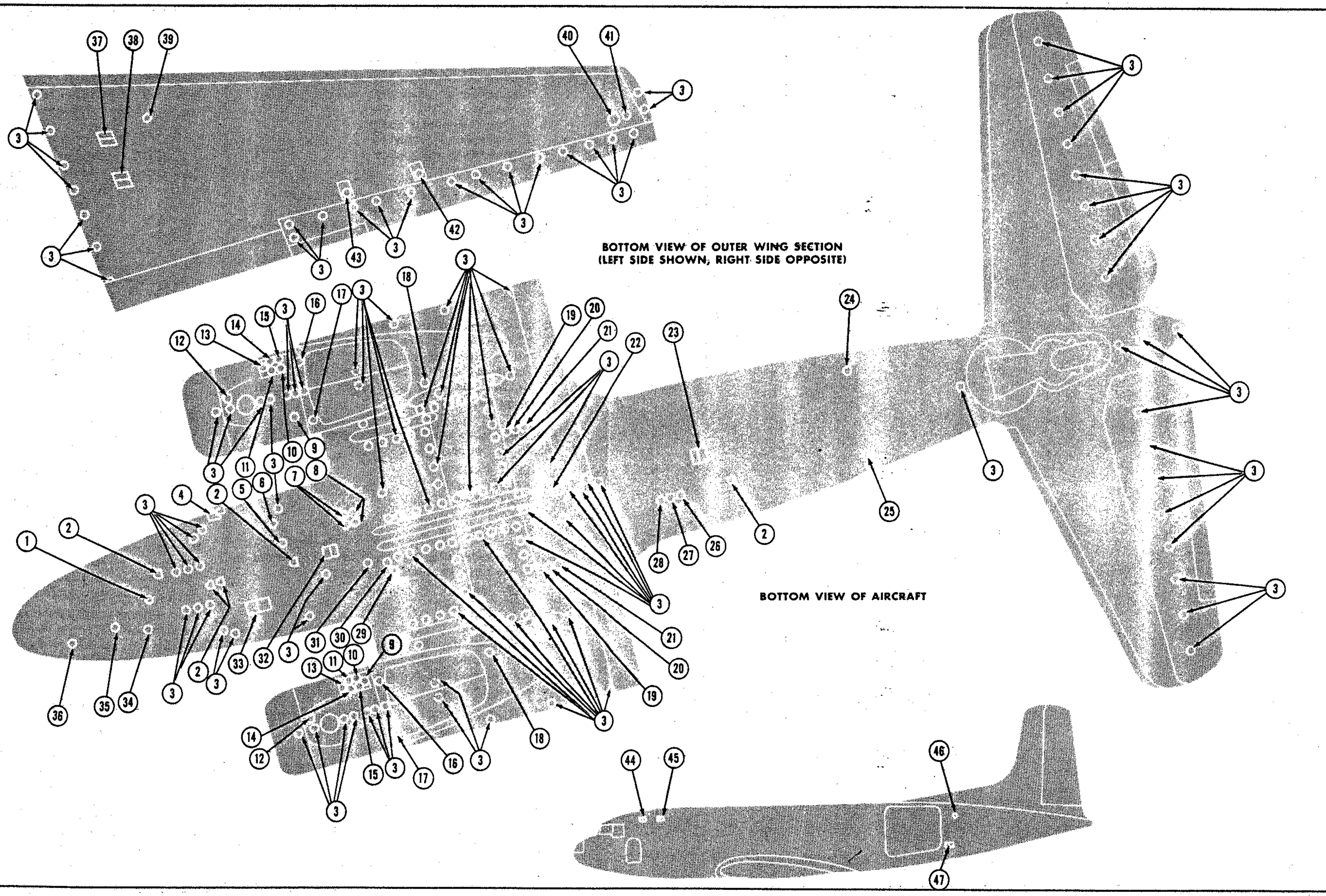
Access Door No.	Access to	Type of Door	Attaching Parts	Tool Used
97.	Ground Blower	Nonstress	Camloc Fastener	Screw Driver
98.	Ground Blower	Nonstress	Camloc Fastener	Screw Driver
99.	Drift Meter Tube	Nonstress	Screw	Reed & Prince Screw Driver
100.	Engine Instrument Piping, Hydraulic Piping, CO_2 Piping, Emergency Air Brake Piping	Nonstress	Screw	Reed & Prince Screw Driver
101.	Control Surface Cables, Engine Control Cables	Nonstress	Screw	Reed & Prince Screw Driver
102.	Brake Valve, Control Surfaces, Heater Ducts	Nonstress	Camloc Fastener	Screw Driver
103.	Outer Wing Fuel Tank Sump Drain	Nonstress	Spring Latch	Hand Operated
104.	Wing Structure	Nonstress	Screw	Reed & Prince Screw Driver
105.	Wing Structure	Stress	Screw	Reed & Prince Screw Driver
106.	Outer Wing Fuel Vent Tank, Wing Structure	Stress	Screw	Reed & Prince Screw Driver
107.	Landing Light Conduit Junction, Wing Structure	Nonstress	Screw	Reed & Prince Screw Driver
108.	(See Door No. 106)	Stress	Screw	Reed & Prince Screw Driver
109.	Wing Flap Actuating Cylinder, Control Surface Cables	Stress	Screw	Reed & Prince Screw Driver
110.	Wing Vent Pipe, Wing Structure	Stress	Screw	Reed & Prince Screw Driver
111.	Wing Tip Attach Bolts, Navigation Light Wiring	Nonstress	Screw	Reed & Prince Screw Driver
112.	Aileron Hinge	Nonstress	Screw	Reed & Prince Screw Driver
113.	Outboard Aileron Bellcrank, Wing Structure	Nonstress	Spring Latch	Hand Operated
114.	Aileron Control Cable, Wing Vent Tube, Structure	Nonstress	Spring Latch	Hand Operated
115.	Inboard Aileron Bellcrank, Wing Structure, Landing Light, Wing Vent Pipe	Nonstress	Spring Latch	Hand Operated
116.	Outer Wing Tank Vent Piping, Aileron Control Cable, Wing Structure	Nonstress	Spring Latch	Hand Operated
117.	(See Door No. 118)	Nonstress	Spring Latch	Hand Operated
118.	Outer Wing Fuel Connection, Outer Wing Fuel Tank Strainer Screen	Stress	Screw	Reed & Prince Screw Driver
119.	Outer Wing Fuel Cell	Stress	Screw	Reed & Prince Screw Driver
120.	Outer Wing Fuel Liquidometer	Stress	Screw	Reed & Prince Screw Driver
121.	Outer Wing Fuel Cell	Stress	Screw	Reed & Prince Screw Driver
122.	Outer Wing Fuel Tank Filler Neck	Nonstress	Spring Latch	Hand Operated
123.	Outer Wing Fuel Cell	Stress	Screw	Reed & Prince Screw Driver
124.	Alighting Gear	Nonstress	Screw	Reed & Prince Screw Driver
125.	Carburetor Airscoop	Nonstress	Screw	Screw Driver
126.	Alighting Gear Actuating Strut Grease Fitting, Oil Tank Hose Fittings	Nonstress	Screw	Reed & Prince Screw Driver
127.	Oil Tank Filler	Nonstress	Dzus Fastener	Screw Driver
128.	Auxiliary Fuel Tank Filler Neck	Nonstress	Camloc Fastener	Screw Driver
129.	Main Fuel Tank Filler	Nonstress	Camloc Fastener	Screw Driver
130.	Alighting Gear Bungee Attachment	Nonstress	Screw	Reed & Prince Screw Driver
131.	Nacelle Vent	Nonstress	Screw	Reed & Prince Screw Driver
132.	Outer Wing Fuel Liquidometer	Stress	Screw	Reed & Prince Screw Driver
133.	Outer Wing Fuel Cell	Stress	Screw	Reed & Prince Screw Driver
134.	Outer Wing Fuel Cell	Stress	Screw	Reed & Prince Screw Driver

(Continued from Page 9)

1-33. Overloading of a cold engine is not necessarily indicated by a discharge of fuel from the supercharger drain, but rather by the presence of liquid fuel, or mist, in the exhaust. A cold engine can be cleared by using the same procedure as that given for a warm engine.

1-34. IDLING. Prolonged idling at low rpm is permissible and will not cause fouling of the spark plugs, providing that the idling fuel-air mixture is at, or near, "best power." If the mixture is too rich, fouling of the plugs will result at low rpm. The rpm for idle mixture adjustment, with cylinder head temperature at 150°C (302°F) minimum during adjustment, should be 700 (± 50) rpm.

1-35. PISTON HYDRAULICKING. If the engine has not been operated for 2 hours or longer, it is essential that the propeller be pulled through at least 4 revolutions (12 blades) by the starter, prior to an engine start, in order to make certain that there is no fuel or oil present in the lower cylinders. Oil and fuel are incompressible and their presence in the lower cylinders might result in a bent rod, a cracked piston, or a damaged crankshaft when the engine is started. If, when the propeller is pulled through, it seizes or binds, do not attempt to force it on around. Remove the spark plugs from the lower cylinders and drain the residual fluid. Then rotate the propeller before again installing the spark plugs.



1. Hydraulic Drip Pan Drain Pipe
2. Fairing Covered External Vapor Vent Holes
3. External Vapor Vent Holes
4. Cabin Heater Exhaust
5. Heater Shroud Drain
6. Combustion Heater Drain
7. Fuel Booster Pump Vent
8. Fuel Booster Pump Drain
9. Engine Breather Pipe
10. Vacuum Pump Drain Pipe
11. Fuel Pump Drain Pipe
12. Motor Vapor Vent
13. Supercharger Drain Pipe
14. Hydraulic Pump Drain Pipe
15. Fuel Pump Vent Pipe
16. Propeller Feathering Pump Drain
17. Oil Tank Scupper Drain Pipe
18. Outer Wing Fuel Cell Sump Can Drain
19. Aft Fuel Tank Vent
20. Forward Fuel Tank Vent
21. Tank Syphon Breaker Pipe
22. Center Wing Trailing Edge Vent
23. Fuselage Belly Vent
24. Toilet Floor Drain Outlet
25. Aft Lavatory Urinal Outlet
26. Alcohol Tank Drain Pipe
27. Alcohol Pump Drain Pipe
28. Alcohol Tank Vent Pipe
29. Surface De-Icer Distributer Valve Vent Pipe
30. Surface De-Icer Distributer Valve Drain Pipe
31. Surface De-Icer Oil Separator Vent Pipe
32. Fuselage Belly Vent
33. Cabin Heater Mixing Valve Air Exhaust
34. Hydraulic Reservoir Drain and Vent
35. Pilot's Relief Tube Outlet
36. Emergency Air Brake Exhaust
37. Outer Wing Fuel Vapor Vent
38. Outer Wing Fuel Vapor Vent
39. Outer Wing Fuel Cell Filler Neck
40. Outer Wing Vent Inlet
41. Outer Wing Vent Inlet
42. Outer Wing Fuel Tank Vent
43. Outer Wing Vent Inlet
44. Master Junction Box Vent
45. Radio Rack Vent
46. Airspeed Static Pipe Vent (Left Vent Shown, Right Vent Opposite)
47. Lavatory Vent (Left Vent Shown, Right Vent Opposite)

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Figure 1-10. Drain and Vent Holes

1-36. FUEL SYSTEM MANAGEMENT. The fuel system consists of the following units:

Center section tanks
Outer wing tanks
Selector valves (center section tanks)
Selector valves (outer wing tanks)
Strainers
Electric fuel booster pumps
Firewall shutoff valves
Engine-driven fuel pumps
Priming system

1-37. The fuel system is basically designed so that each engine is provided with an independent fuel supply system, including its own fuel tanks, although fuel from any of the tanks may be directed to either of the engines. Transfer of fuel between tanks is not possible.

Note

Fuel from the front tanks should be used for all ground operations, if all of the tanks are full; if any tank is not full, fuel from the tanks containing the most fuel should be used. Do not use the outer wing tanks for ground operations.

1-38. The inner wing tank fuel selector dials are divided into five positions as follows:

LEFT FRONT
RIGHT FRONT
LEFT REAR
RIGHT REAR
OFF

1-39. The outer wing tank fuel selector dials are divided into three positions as follows:

LEFT TANK
RIGHT TANK
OFF

1-40. Through the operation of the selector valves in any of the above positions, fuel can be supplied to either or both of the engines from any of the inner or outer wing tanks.

1-41. FUEL BOOSTER PUMP CHECK. Before starting the engines, check the operation of both electric fuel booster pumps as follows:

a. Place the left mixture control in IDLE CUT-OFF position.

b. Place the left fuel tank selector valve in position to select the desired tank.

c. Place the booster pump switch for the left booster pump in the ON position. The fuel pressure indicator for the left engine should read 21 ($\pm\frac{1}{2}$) psi.

d. Repeat steps a, b, and c for the right fuel booster pump.

1-42. FUEL TANK SELECTOR VALVE CHECK. With the engines running, check the selector valves to insure unrestricted flow through all pipes and valves in the system.

1-43. ENGINE STARTING PROCEDURE. For engine starting procedure, see the applicable handbook of flight operating instructions.

1-44. PROPELLER FEATHERING CHECKS.

1-45. PROPELLER FEATHERING PROCEDURE FOR GROUND OPERATION. For complete information on propeller feathering or engine isolation procedures, refer to the applicable handbook of flight operating instructions. During ground check of the propeller feathering system, it is not required to shut down the engine. To make the ground check, momentarily depress the feathering button, located on the pilot's or co-pilot's overhead electrical panels, until a drop of 300 to 400 rpm is attained. Then pull the feathering button out and allow the propeller to return to the original rpm setting.

1-46. GROUND FIRE CONTROL. Ground fire extinguishing equipment should be spotted in the immediate vicinity of the aircraft during ground operations. Use the ground fire extinguishing equipment, if possible, rather than the aircraft's fire extinguishing system. Access holes, covered with spring-loaded doors, are located in the outboard side of each nacelle. Insert the CO_2 ground fire equipment horn by pushing it against the access door. Two hand fire extinguishers are provided, one on the cockpit floor aft of the co-pilot's seat, and one in the cabin area, just forward of the main cargo door. The engine compartments and the combustion heater compartment fire extinguishing systems are controlled from a panel on the floor between the pilot's and co-pilot's seats. For fire extinguishing system operating procedures, see the applicable handbook of flight operating instructions.

1-47. ENGINE SECTION FIRE. Each engine compartment is divided into two areas, or fire zones. Zone 1 is the area forward of the inner ring and Zone 2 is the area between the inner ring and the firewall. A fire detector system is installed in each zone, but only Zone 2 is protected by CO_2. Detectors located in both zones of an engine compartment are connected to a warning light located on the pilot's side of the main instrument panel; this light will illuminate in the event of a fire.

1-48. COMBUSTION HEATER COMPARTMENT FIRE. A fire detector system is provided in the combustion heater compartment. Detectors, which are located in the heater ducting and throughout the compartment, are connected to a warning light on the left overhead electrical panel. The light will be illuminated in the event of a fire.

1-49. FIRE WARNING CIRCUIT TEST SWITCHES. The fire detector circuits can be electrically tested for normal circuit continuity prior to flight by means

of the momentary-contact circuit test switches installed adjacent to the warning lights.

1-50. ENGINE SECTION FIRE WARNING CIRCUIT TEST SWITCH. Two lights and one test switch are installed on the pilot's instrument panel to check the continuity of the two engine circuits. When the test switch is depressed, the warning lights will illuminate and the fire bell will sound. From 2 to 10 seconds may elapse before the lights will illuminate when the test switch is closed.

1-51. HEATER COMPARTMENT FIRE WARNING CIRCUIT TEST SWITCH. One light and one test switch are installed on the overhead electrical panel to check the continuity of the heater compartment circuit. When the test switch is depressed, the light will illuminate and the fire bell will sound. The light will illuminate immediately when the test switch is closed.

1-52. ENGINE RUN-UP. See the applicable handbook of flight operating instructions for specific engine operating procedures.

1-53. GROUND HANDLING.

1-54. HOISTING.

1-55. DESCRIPTION. Provisions are made for hoisting the complete aircraft, or for hoisting component parts of the aircraft, including the fuselage, the outer wing panels, the horizontal stabilizer, the vertical stabilizer, the engine assemblies, and the propellers.

CAUTION

In every case, the capacity of overhead hoisting equipment and the security of suspension must be carefully checked before it is used.

1-56. HOISTING COMPLETE AIRCRAFT. *(See figure 1-11.)* Six permanently installed fittings are provided to lift the complete aircraft, using three separate hoists. One hoist, incorporating a two-cable sling, is attached at each of the nacelles to lift the forward section of the aircraft. A single hoist, incorporating a two-cable sling, lifts the empennage. The forward section sling cables are fastened to two fittings, in each nacelle, which are bolted to the top of each of the main gear strut fittings. The sling cables for the empennage are bolted to the 2 pick-up points at station 623. Access to the four forward fittings, used with the two forward hoists, is gained by removing the two nacelle access hole covers which are installed just above the hoisting fittings, located in each nacelle. Single point wing jacks and a tail stand sometimes prove useful during the hoisting operations.

1-57. HOISTING FUSELAGE. Two hoists must be used to lift the fuselage. The forward hoist is attached to a 2-cable sling which is fastened by clevises and bolts to permanent shackles on each side of the fuselage at station 202, forward of the auxiliary spar of the center wing section. The aft hoist, through a 2-cable sling, attaches to the fittings at station 624.

CAUTION

Before attempting to hoist the fuselage, remove all baggage, furnishings, and equipment from the interior in order to prevent any overloading of the fuselage structure.

1-58. HOISTING OUTER WING PANEL. *(See figure 1-12.)* A single hoist is used in raising the outer wing panel. The hoist is attached to a four-cable sling which, in turn, is connected to the four points of attachment on the outer wing panel, by the use of tee-screws on the ends of the four sling cables. The four points of attachment on the outer wing panel are plugged with flush screws when the sling is not installed. The tee-screws must be tightened snugly into the panel to prevent bending or breaking.

1-59. HOISTING HORIZONTAL STABILIZER. Six to eight men can manually lift the horizontal stabilizer if a suitable sling is not available.

1-60. HOISTING VERTICAL STABILIZER. Four to six men can manually accomplish this operation.

1-61. HOISTING RUDDER. The rudder may be handled by two men during removal or installation.

1-62. HOISTING ENGINE ASSEMBLY. *(See figure 1-13.)* An engine assembly is hoisted by means of a sling attached to the three hoist fittings, one provided on the engine nose section and two on the upper engine mount fittings at the firewall.

1-63. HOISTING PROPELLER. Two slings may be used in lifting the propellers. One is a two-cable sling used in raising a propeller while it is held in a vertical position, and the other is a three-cable sling used in raising a propeller while it is held in a horizontal position.

1-64. JACKING.

1-65. DESCRIPTION. *(See figures 1-14 through 1-18.)* Provisions are made for jacking the entire aircraft, the forward section, the empennage, and for emergency jacking of the main wheel axles. See figures 1-14 and 1-15 for location of jack points.

1-66. JACKING PRECAUTIONS. Make certain that jacks, shoring stands, etc., are of sufficient capacity, both vertically and horizontally, to support the air-

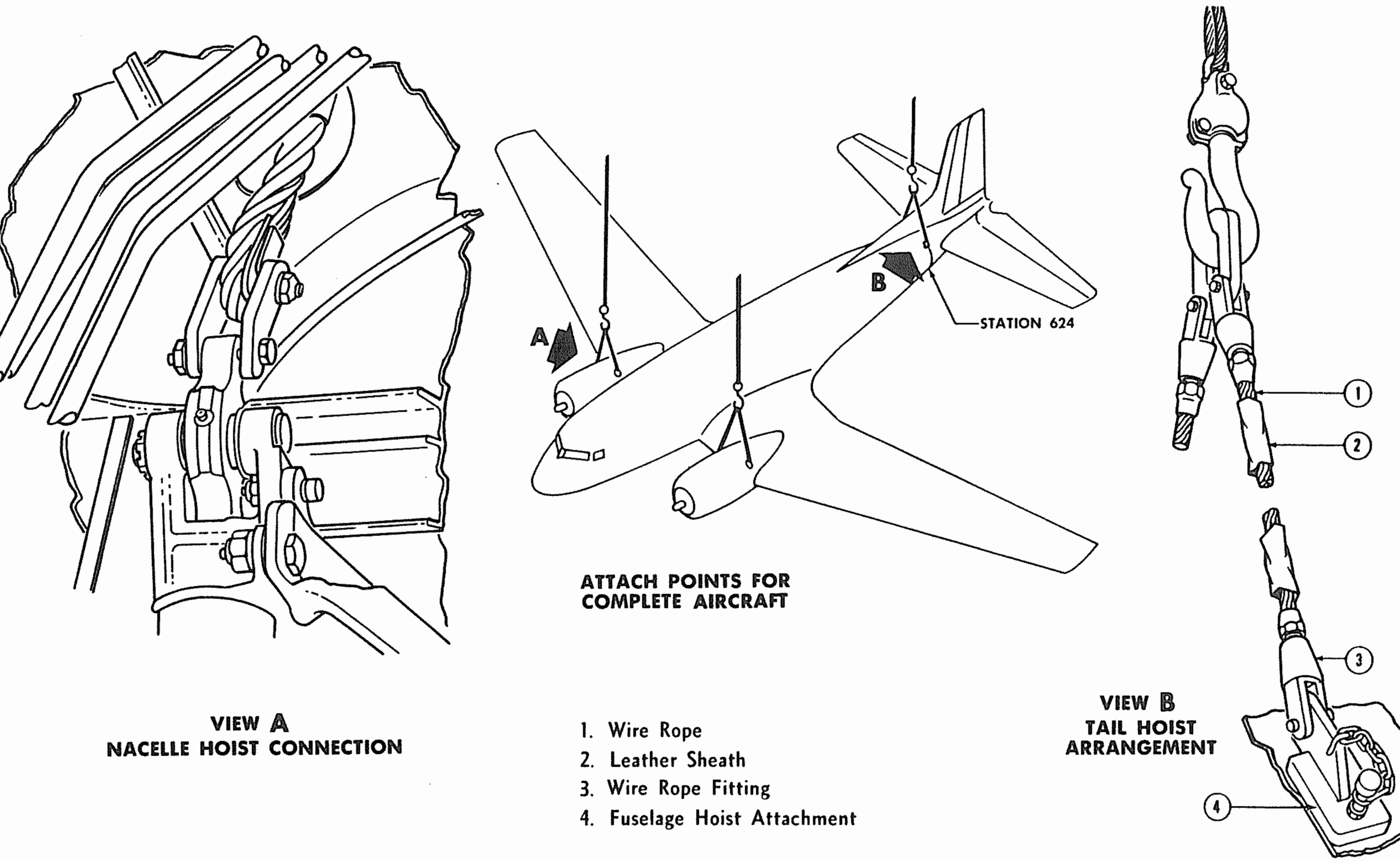

Figure 1-11. Hoisting Complete Aircraft

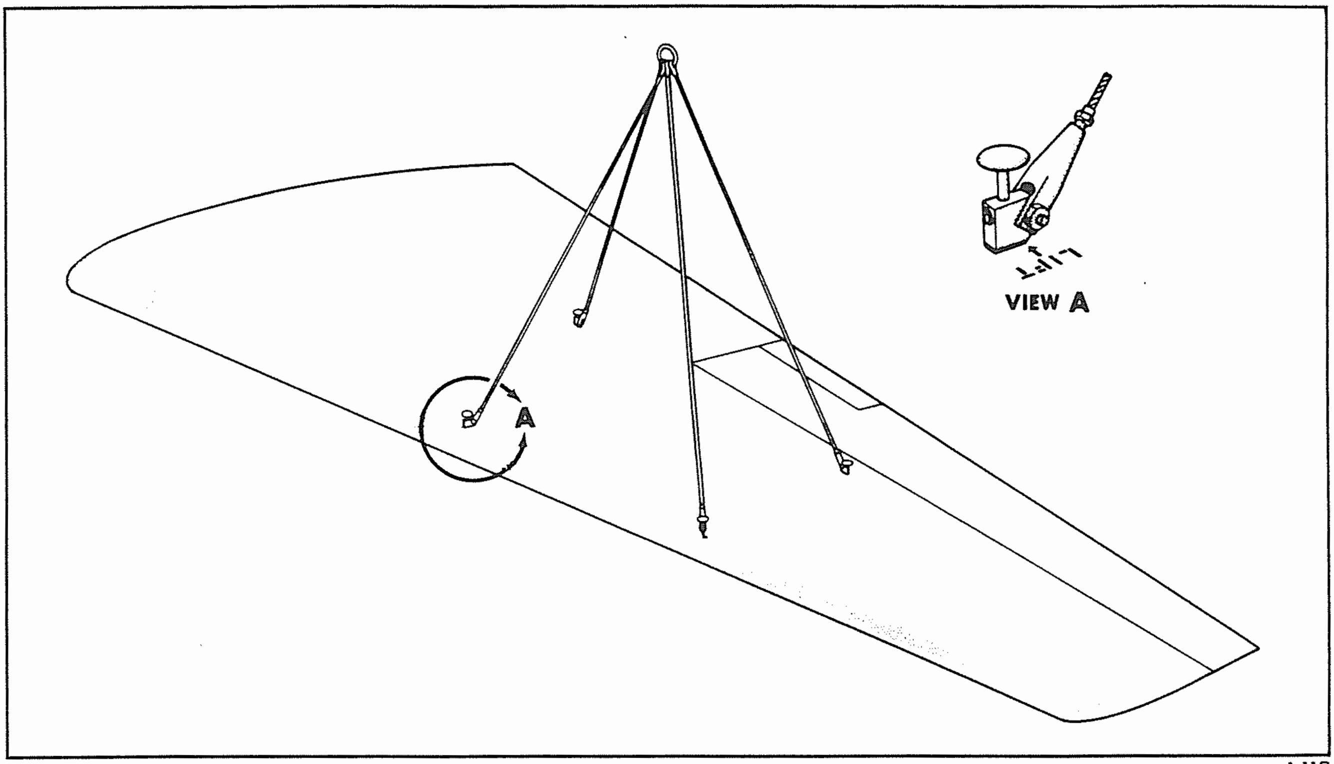

Figure 1-12. Outer Wing Panel Hoist Sling

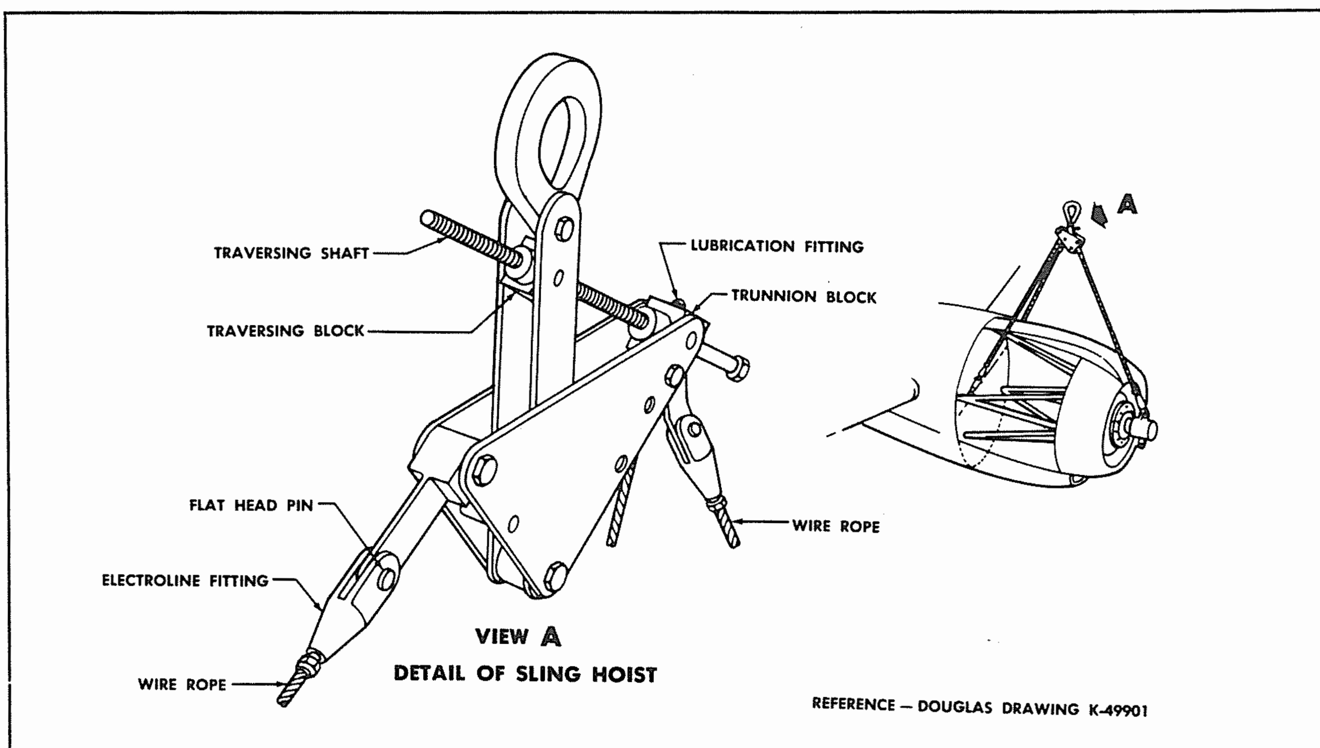

Figure 1-13. Engine Assembly Hoist Sling

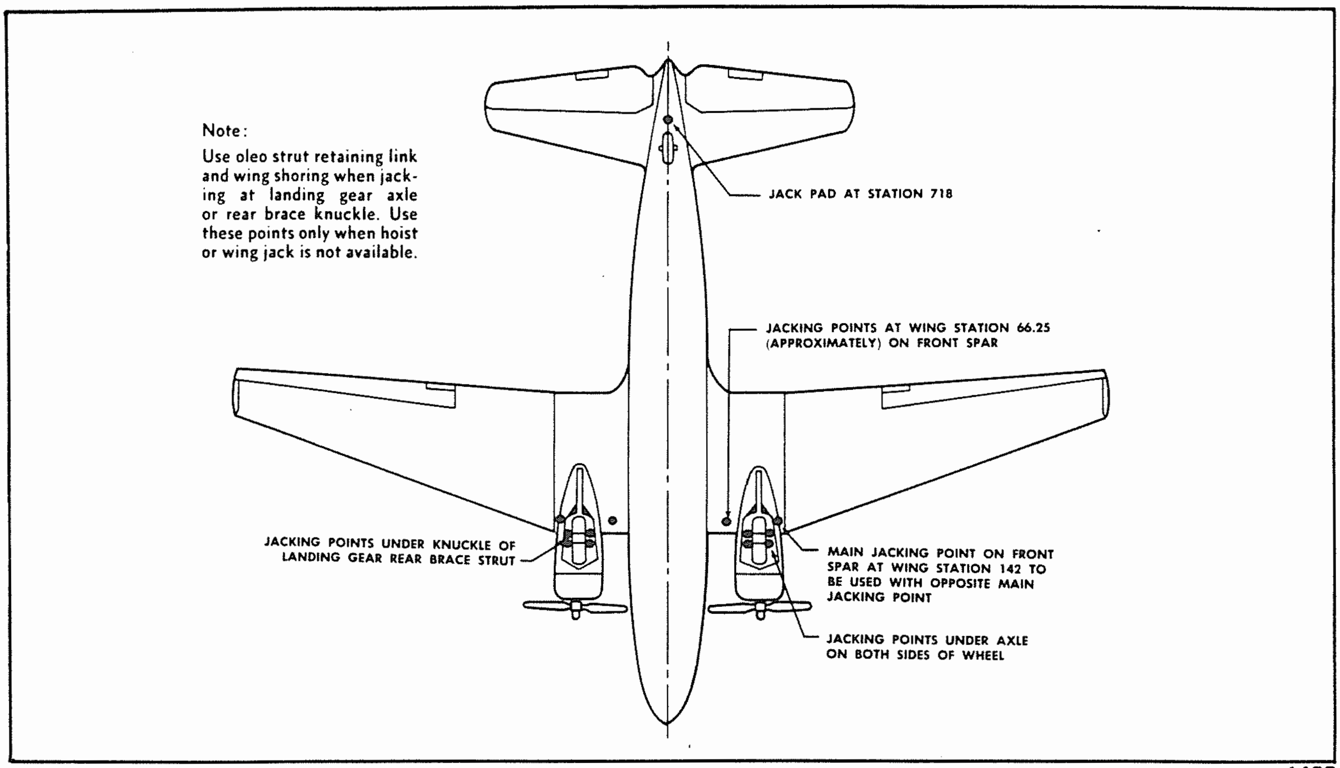

Figure 1-14. Jacking Points

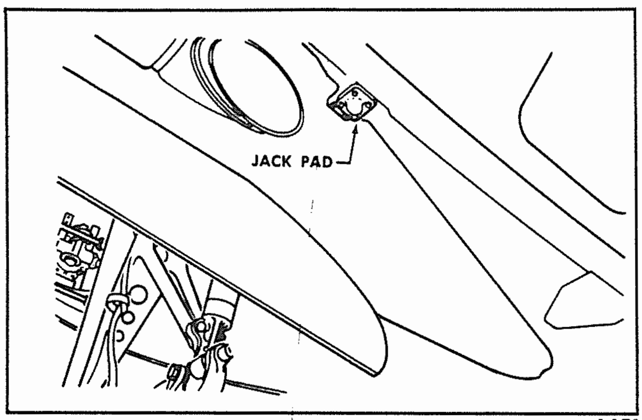

Figure 1-15. Wing Jack Pad

craft safely. Do not jack the aircraft on uneven ground. First move the aircraft to a level area and make certain that the jacks are seated on a level surface. Head the aircraft into the wind before operating the jacks. Do not attempt to jack the aircraft if the wind exceeds 15 miles per hour. Parking brakes should be off and the tail wheel locked when either the forward section or the tail section of the aircraft is jacked. When raising the entire aircraft, operate all jacks simultaneously to keep the aircraft in the normally inclined position. The safety pins, provided with the jacks, should be used as a precaution against hydraulic failure, after the aircraft has been raised to the required height. Prevent any movement of the aircraft while it is resting on the jacks.

CAUTION

A certain amount of settling occurs after the full weight of the aircraft is taken by the alighting gear; therefore, make certain that the jacks are retracted sufficiently to allow for settling while they are being removed. If this procedure is not followed, and the jacks are not directly below the jack pads, the aircraft may be damaged by settling on the jacks.

1-67. JACKING ENTIRE AIRCRAFT. *(See figure 1-14.)* Jack points are provided for raising the entire aircraft. The forward section is raised by using either of 2 sets of jack points, located on the wing front spar, one at station 75 and the other at station 142. (See figure 1-19 for allowable gross weights.) The jack pads at station 75 should be used for all normal jacking operations. When the fuel tank cover doors are removed, it will be necessary to use the jack pads at station 142 in addition to the jack pads at station 75, due to structural limitations when the fuel tank covers are removed. The outboard nacelle doors must be removed to gain access to the jack pads at station 142. The empennage is raised by using a single jack point at station 718 (see paragraph 1-68, following).

1-68. JACKING EMPENNAGE. *(See figures 1-20, 1-21, and 1-22.)* A jack pad is first attached to the

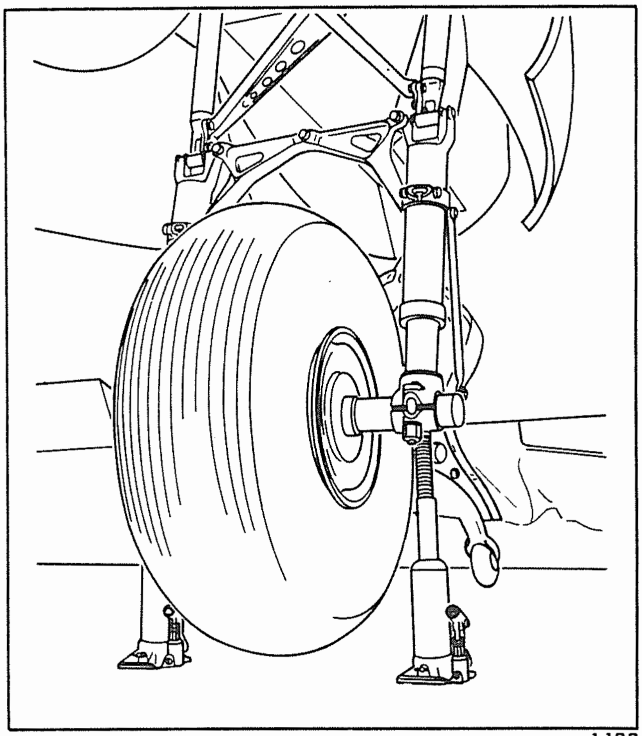

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Figure 1-16. Jacks Installed at Main Wheel Axle

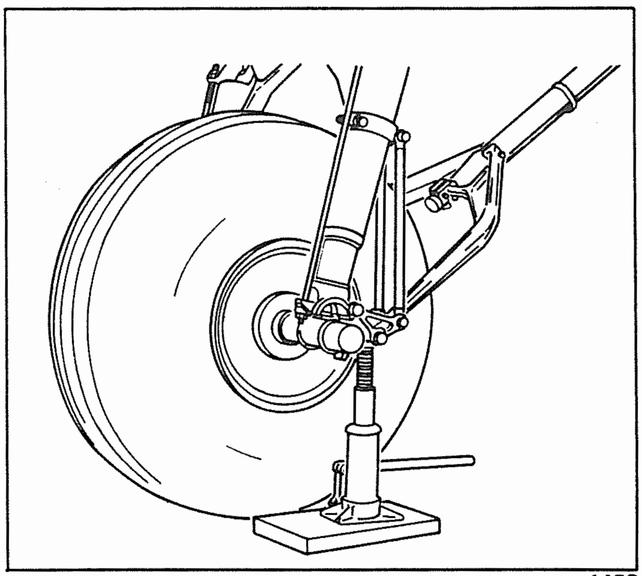

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Figure 1-17. Wheel Ready for Removal

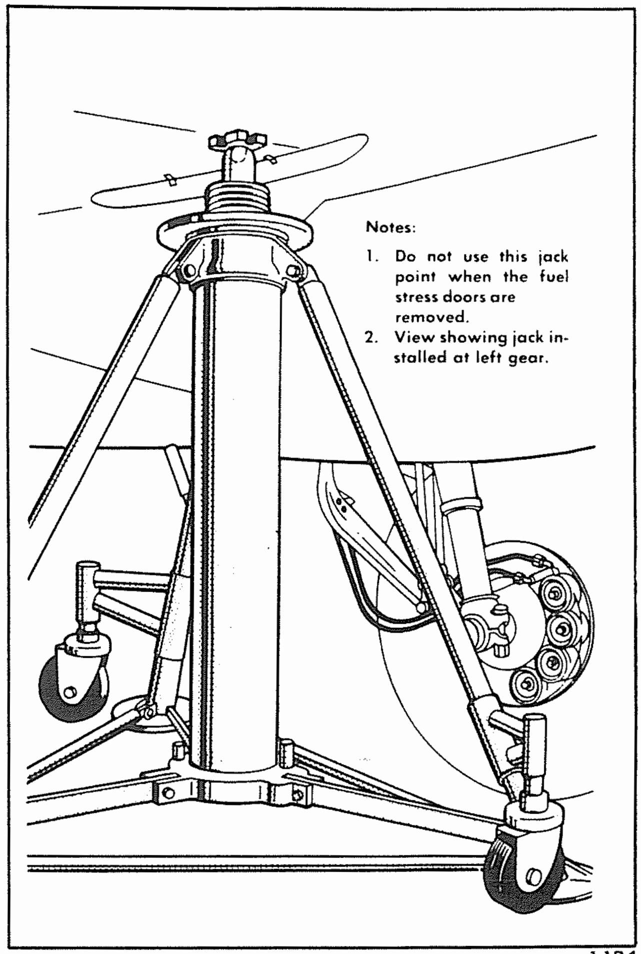

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Figure 1-18. Wing Jack Installed

tail skid, at station 718. The empennage may then be raised by engaging a jack with the jack pad. It is difficult to jack the empennage to any great height when the forward section is not to be raised at the same time. The top of the jack moves aft as it is raised; therefore, to counteract this movement, the jack should be inclined slightly forward before it is put into operation. Ordinarily the empennage should not be jacked higher than five to seven inches, which is sufficient to permit servicing of the tail gear. When raising the empennage so that the aircraft is in level flight position, a tail hoist must be used.

1-69. EMERGENCY JACKING AT MAIN WHEEL AXLES. *(See figures 1-23 and 1-24.)* An emergency method of raising a main wheel, when neither wing jacks nor hoists are available, may be used. Two short jacks are placed under the main wheel axle. The jacking points for this procedure are located on the bottom of the axle housing. Service jobs which can be accomplished when a main wheel is raised in this manner are limited. Since the axle housings are trapped by the jacks, it is impossible to remove a main wheel; therefore, it is not possible to service the main wheel bearings, brakes, or tires. If it is necessary to remove a main wheel where hoists or wing jacks are not available, proceed as follows:

a. Using two short jacks, one on either side of the wheel, jack the wheel by the axle ends.

Notes:

1 Below solid lines, aircraft may be raised using jacks and tail hoist
2 Below dashed line, tail hoist may be used (main gear on ground)
3 Below dotted line, one side may be jacked (other main wheel and tail wheel on ground)

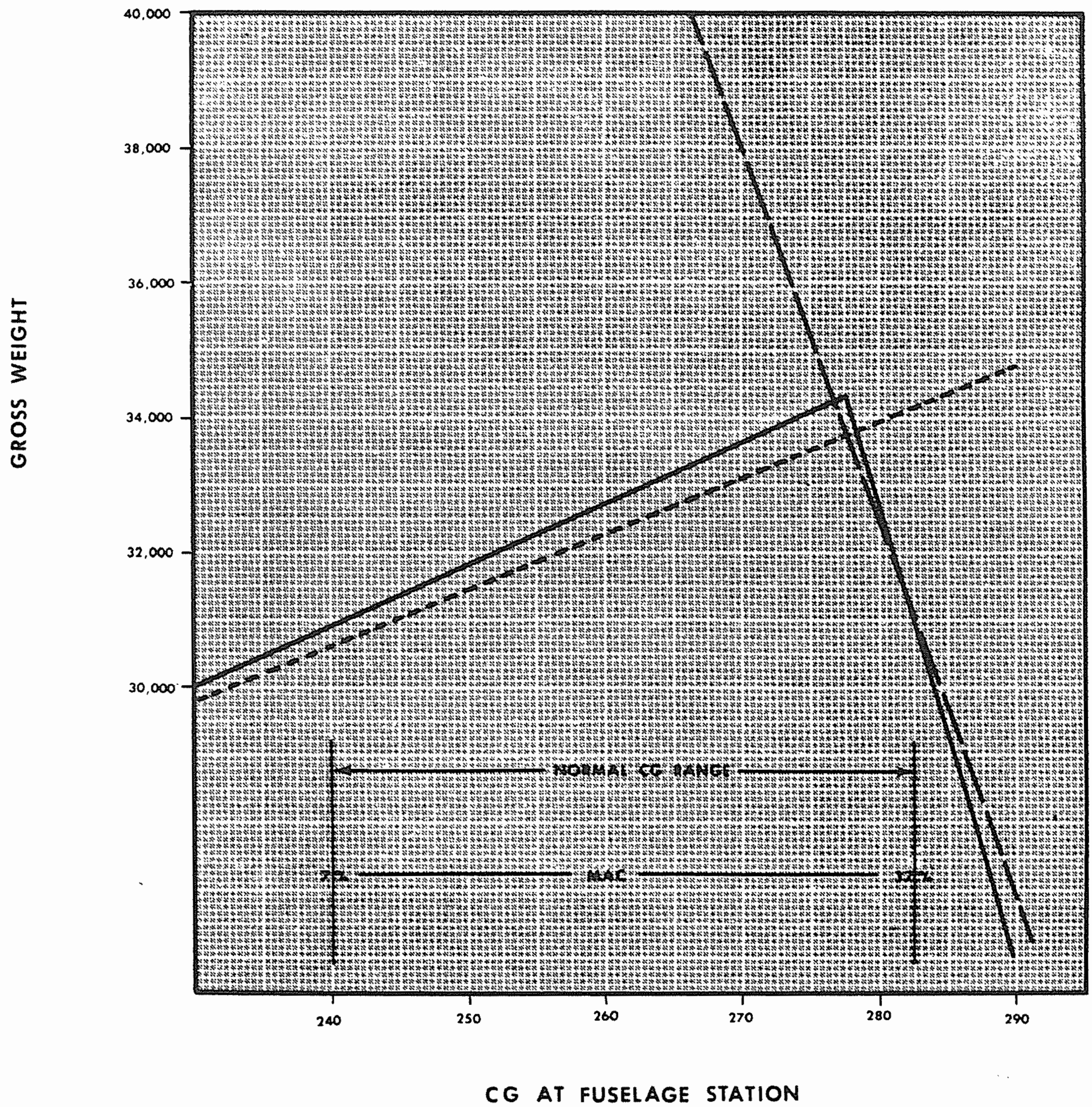

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Figure 1-19. Wing Jack Point Load vs. Gross Weight

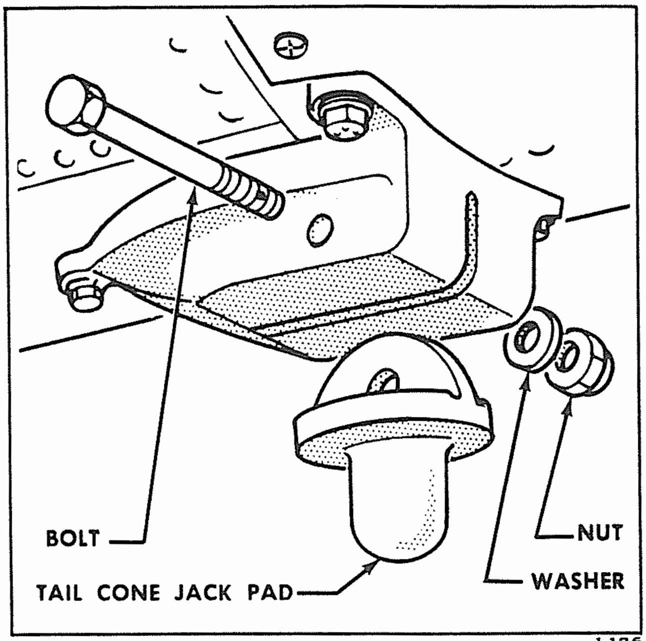


Figure 1-20. Tail Cone Jack Pad

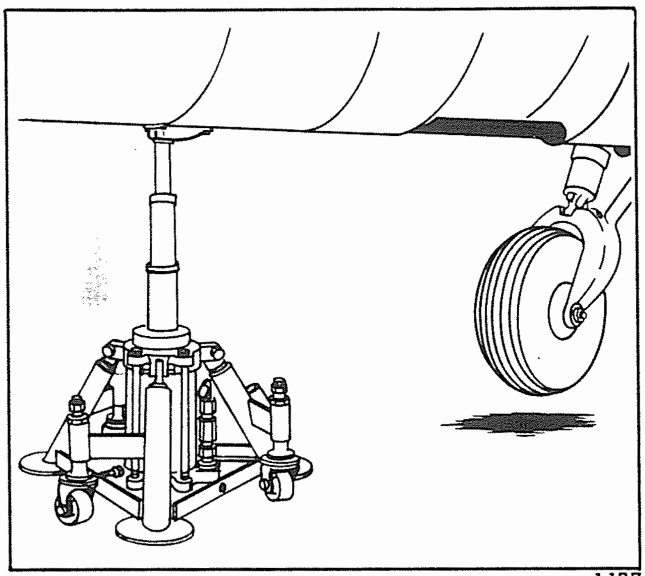


Figure 1-21. Jack Installed at Tail

b. Support the raised wing with shoring.

c. Remove the two jacks.

d. Remove the wheel and accomplish the necessary servicing.

e. Rejack the wheel and remove the shoring.

f. Lower and remove the jacks.

1-70. JACKING AT MAIN WHEEL BRACE STRUT ENDS.
(See figure 1-25.)

a. Install the shock strut restraining link assembly (Douglas Part No. 2140682).

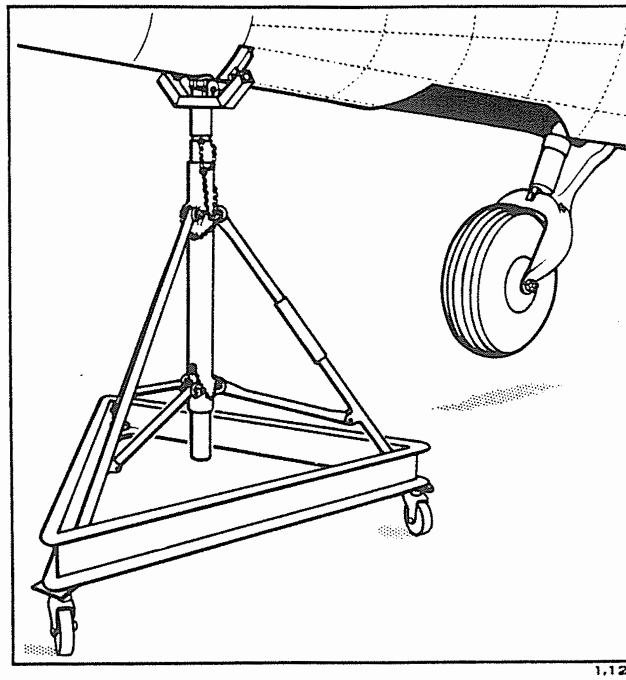
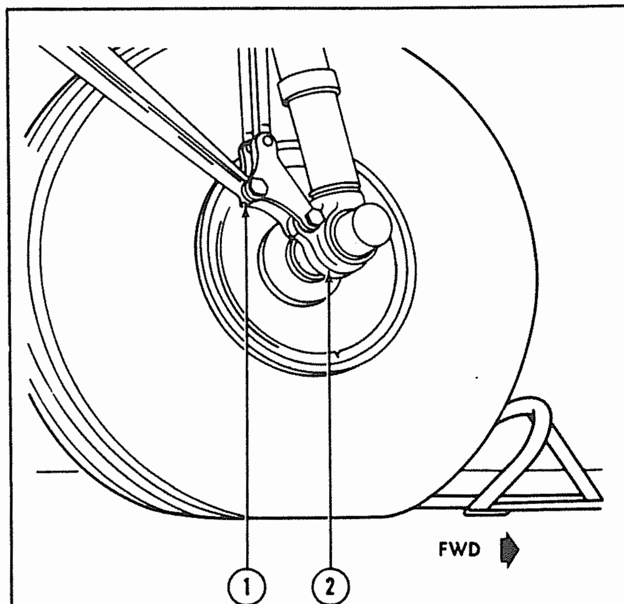


Figure 1-22. Tail Stand Installed



1. Rear Brace Strut Jacking Point
2. Main Gear Axle Jacking Point

1,129

Figure 1-23. Main Gear Jacking Points

b. Place a short jack under the forward Y-shaped end of the rear brace strut.

c. Raise the forward section of the aircraft.

1-71. LEVELING.

1-72. DESCRIPTION. *(See figure 1-26.)* Three sets of leveling pins are provided so that the aircraft can be leveled on both the longitudinal and the lateral axes. One set of leveling pins is located on the right exterior of the fuselage, just below the aft cabin window, at stations 389 and 410. The other two sets are located on the lower side of the left and right center wing, just aft of the leading edge.

1-73. LEVELING PROCEDURE. Leveling is required before weighing the aircraft to determine its basic weight and the location of its center of gravity. To level the aircraft longitudinally, place a sensitive-type spirit level against the fuselage reference pins and adjust the aircraft until it is level. When considerable leveling and weighing is to be done, an inclined ramp may be constructed for the tail wheel. If such a ramp is used, hoisting the tail is not necessary in leveling the aircraft longitudinally. If a tail ramp is not available, a tail hoist is required. Do not attempt to jack the tail to the level position. To level the aircraft laterally, from wing tip to wing tip, use the center wing reference pins and adjust the aircraft by manipulating either the jacks or the tire and shock strut pressures until a level attitude is attained.

Note

Use differential tire and shock strut pressures in leveling the aircraft only if jacking facilities are not available.

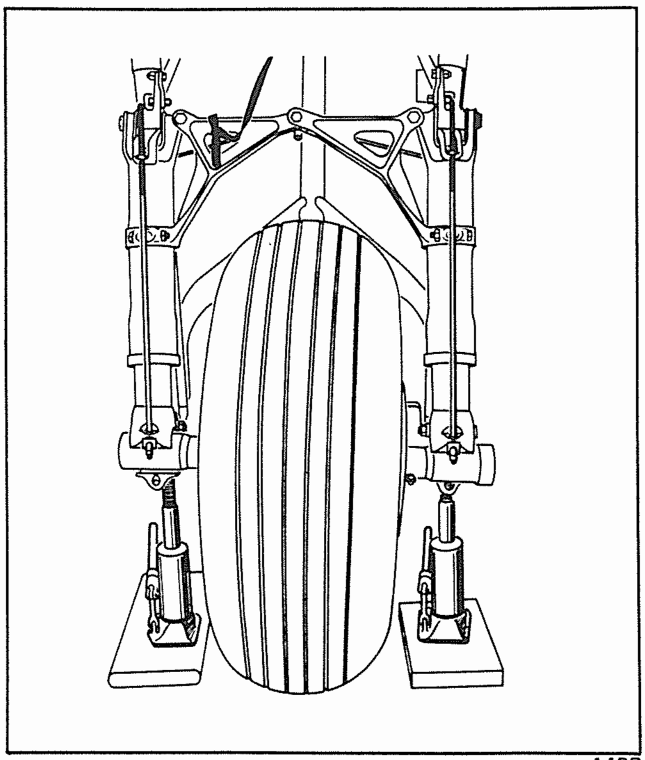


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Figure 1-24. Jacks Installed at Knuckles of Rear Brace Strut

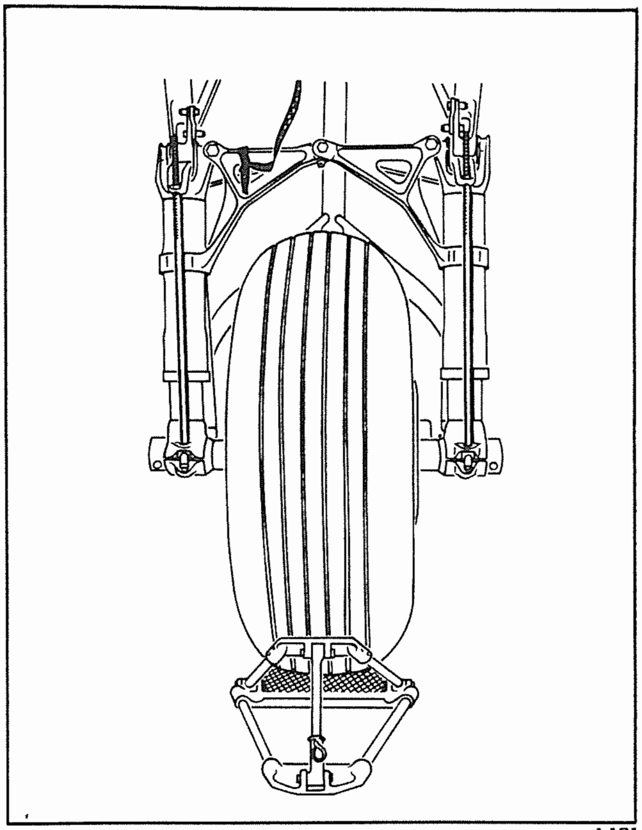

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Figure 1-25. Shock Strut Restraining Links Installed

1-74. MOORING.

1-75. DESCRIPTION. *(See figures 1-27 and 1-28.)* During normal weather conditions, it is not necessary to tie down the aircraft when other precautions, such as setting the parking brakes and chocking the wheels, have been taken. If high winds are expected, tie down the aircraft to a prepared mooring base as follows:

Note

When high winds are anticipated, spoilers must be installed on the wings in order to give added protection to the moored aircraft. A spoiler is a device which is fastened to the upper surface of a wing to break the air flow passing over it and thus minimize potential lift.

a. Place chocks snugly against both main and tail wheel gears.

b. Install the rudder, aileron, and elevator control-surface locks *(see figures 1-29 and 1-30).*

c. Attach two tie-down cables, one leading forward and the other leading aft, to each main gear. Tie-down lugs are not provided; therefore, the cable will be

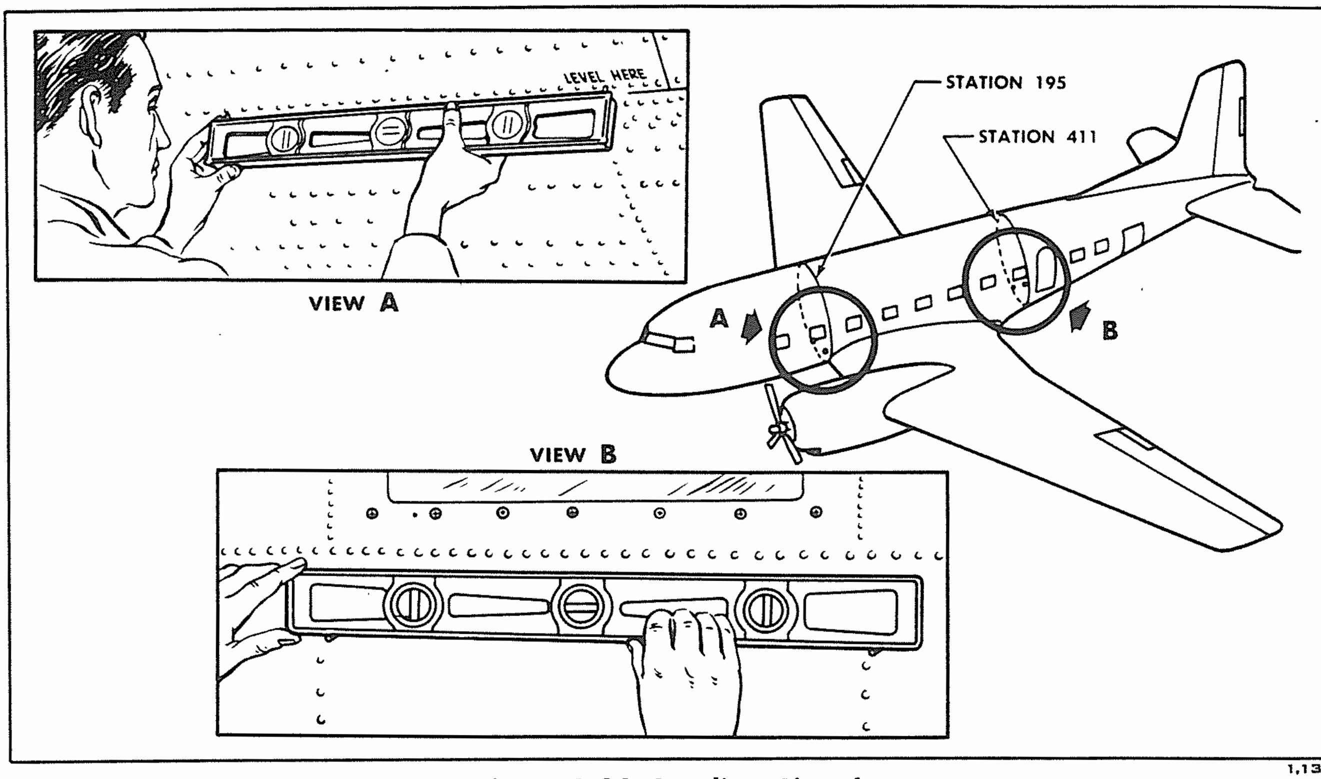

Figure 1-26. Leveling Aircraft

secured to the fixed portion of the main gear above the shock struts. These moorings, when anchored, must be taut.

d. Connect 4 mooring lines to the retractable ring, on the bottom surface of each wing, at station 350; anchor the lines in 4 directions, as shown on figure 1-28. Allowing 16 inches of slack in each line will prevent undue stress on the mooring line installation, in the event that a tire or strut, on the opposite side of the aircraft, should become deflated.

e. Lead two lines aft from the tail wheel fork to the ground anchor points, as shown in figure 1-28. These lines must be taut.

1-76. GROUND ANCHOR POINTS. See figure 1-28 for proper location of ground anchor points.

1-77. PARKING.

1-78. DESCRIPTION. *(See figures 1-29 and 1-30.)* When parking the aircraft, be certain that the aileron, rudder, and elevator locks are installed and the wheels are chocked. The parking brakes should never be left set, particularly when the brakes are hot. Make certain that all electrical equipment is OFF and doors and windows are closed.

1-79. TOWING.

1-80. DESCRIPTION. *(See figures 1-31 through 1-34.)* The aircraft may be towed from the front, by attaching tow lines to the main wheel axles, or from the tail by attaching a tow bar to the tail wheel axle. The tow bar hooks over towing lugs which extend from each end of the tail wheel axle. During the towing operation, there should always be one man in the cockpit to operate the brakes, and a man at each wing tip when the aircraft is towed near hangars or other aircraft. When the aircraft is towed either from the front or the rear, the tail wheel must be in the UNLOCKED position. The aircraft must never be moved by pushing or pulling on the control surfaces or stabilizers. The following precautions should be observed in towing:

a. Do not attempt to tow the aircraft when the parking brakes are set.

b. Do not attempt to tow the aircraft over rough ground.

c. A suitable double-acting spring and friction shock absorber, if available, should be used in all towing operations.

d. In starting, when towing from the front, the acute angle between the center line of the aircraft and the line of the towing force must not exceed 60 degrees.

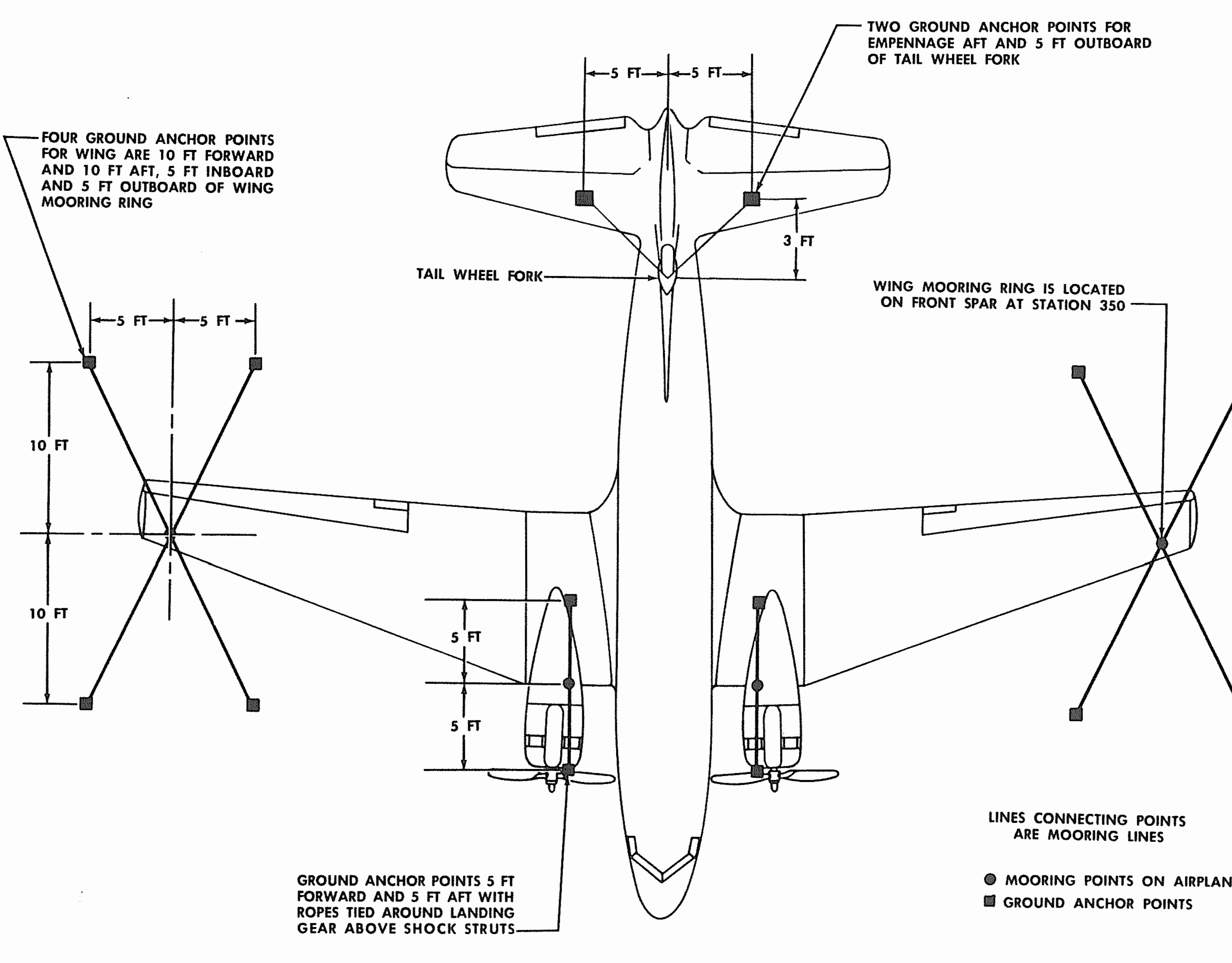

Figure 1-27. Mooring Points

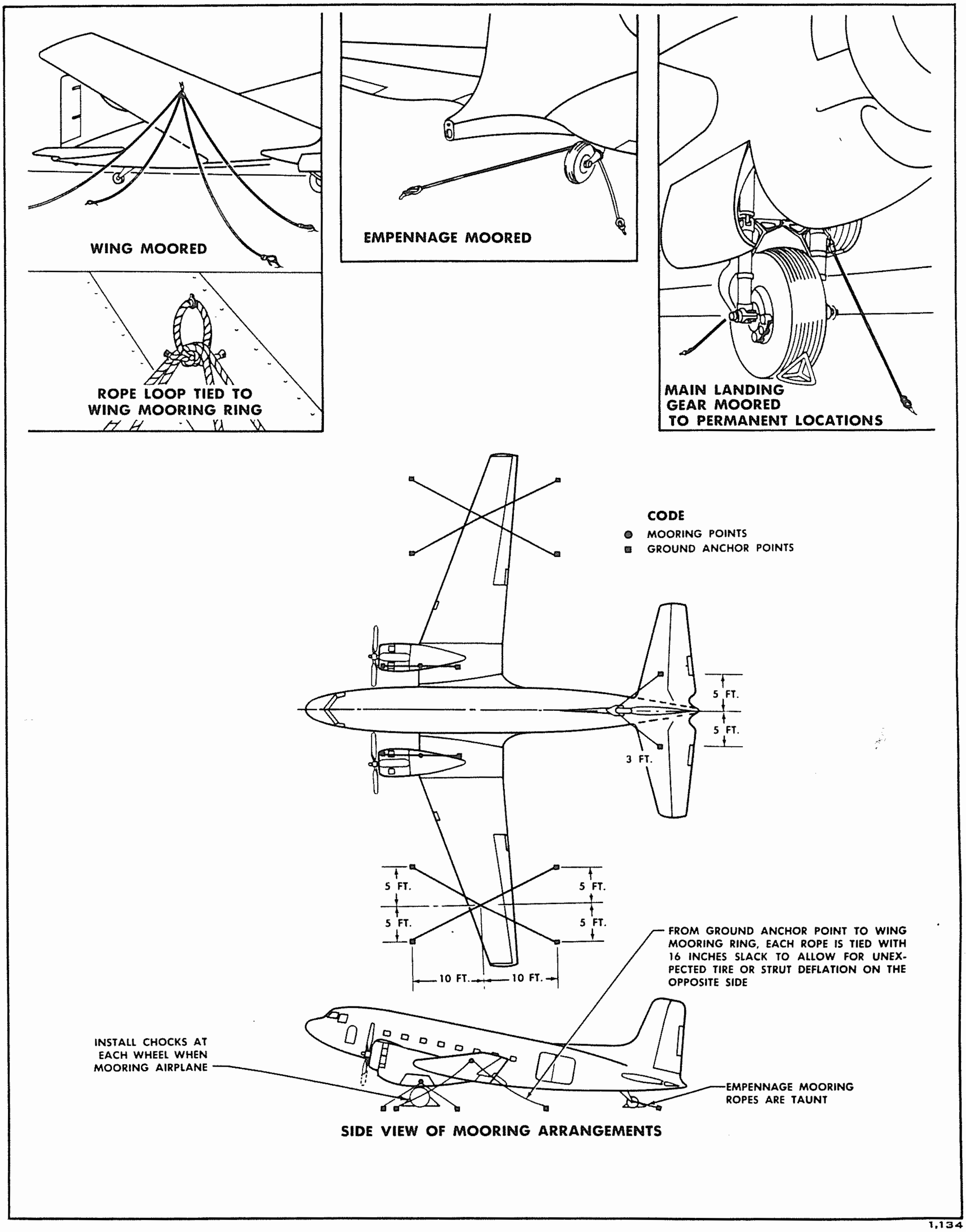

Figure 1-28. Mooring Diagram

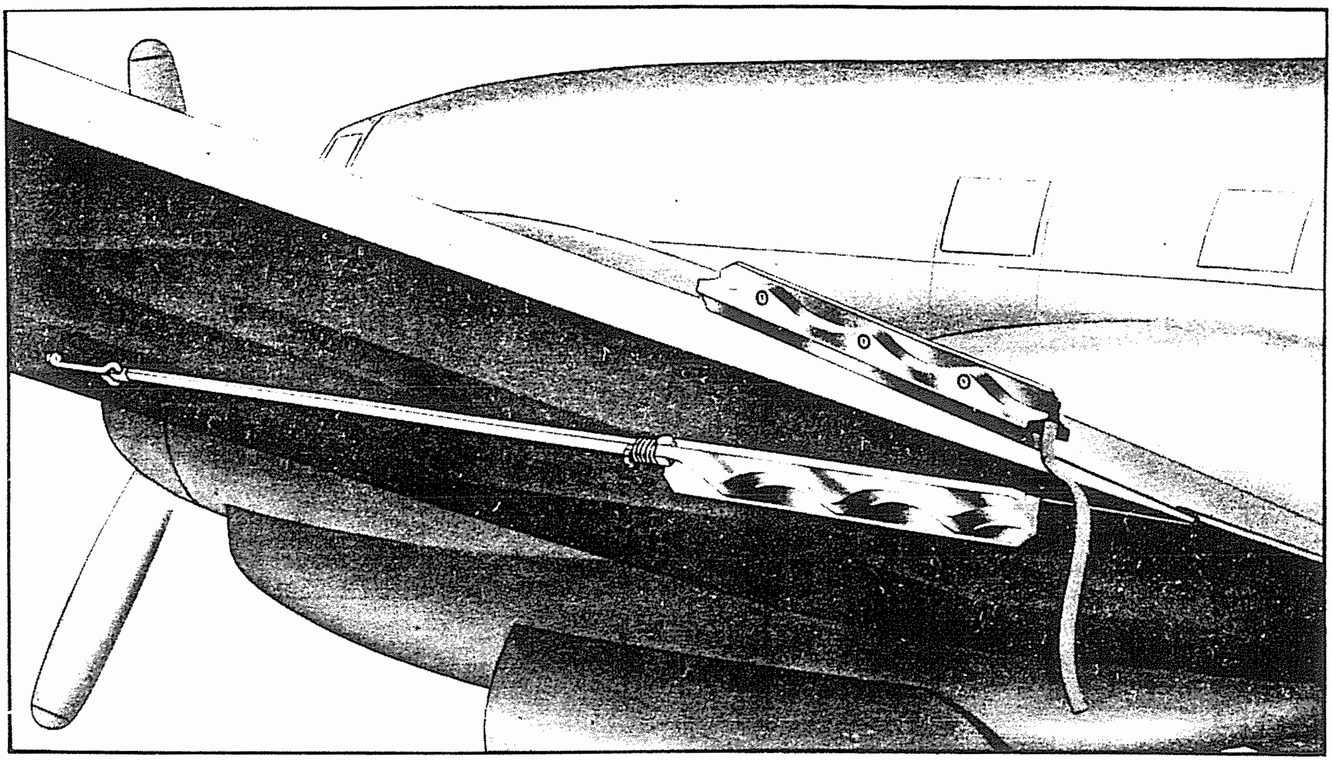

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Figure 1-29. Aileron Lock Installed

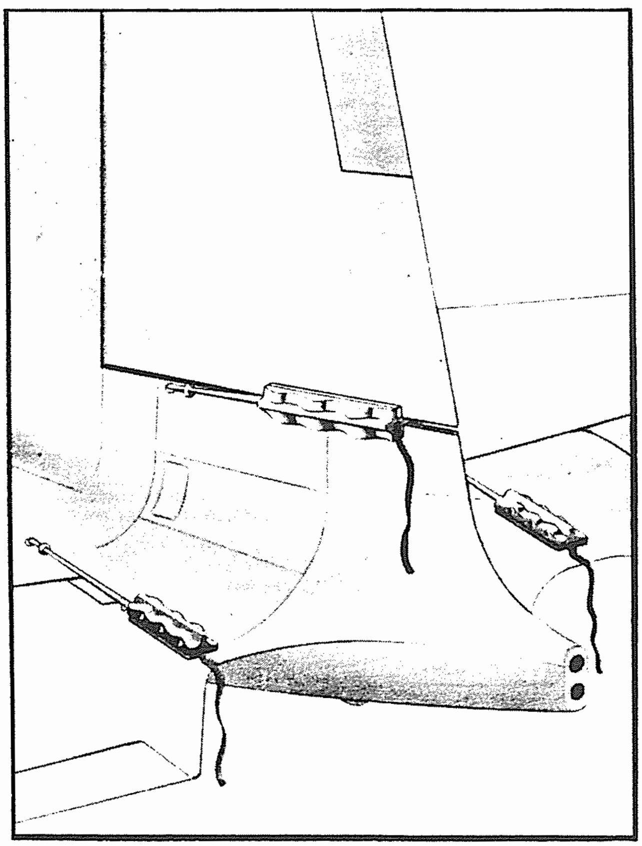

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Figure 1-30. Rudder and Elevator Locks Installed

e. When moving the aircraft by use of the tail tow bar, the acute angle between the tail wheel and the center line of the aircraft must not exceed 60 degrees at any time; in starting, this angle must not exceed 45 degrees *(see figure 1-35)*. The maximum travel of the tail wheel is 120 degrees on each side of the center line of the aircraft.

f. Towing or pushing from the tail must never be attempted up a grade steeper than 11 per cent (on hard ground); it should be avoided on soft, boggy, or rough ground.

g. If the aircraft is loaded, or is being moved over soft ground, it must be towed from the front.

h. Use low speed in towing from either the front or the tail; avoid sudden stops, particularly when towing from the tail.

i. Never attempt to tow by a rope, or ropes, from the tail wheel; use a tail wheel tow bar.

1-81. TAXIING.

1-82. DESCRIPTION. Observe the following precautions while taxiing the aircraft *(see figure 1-35)*:

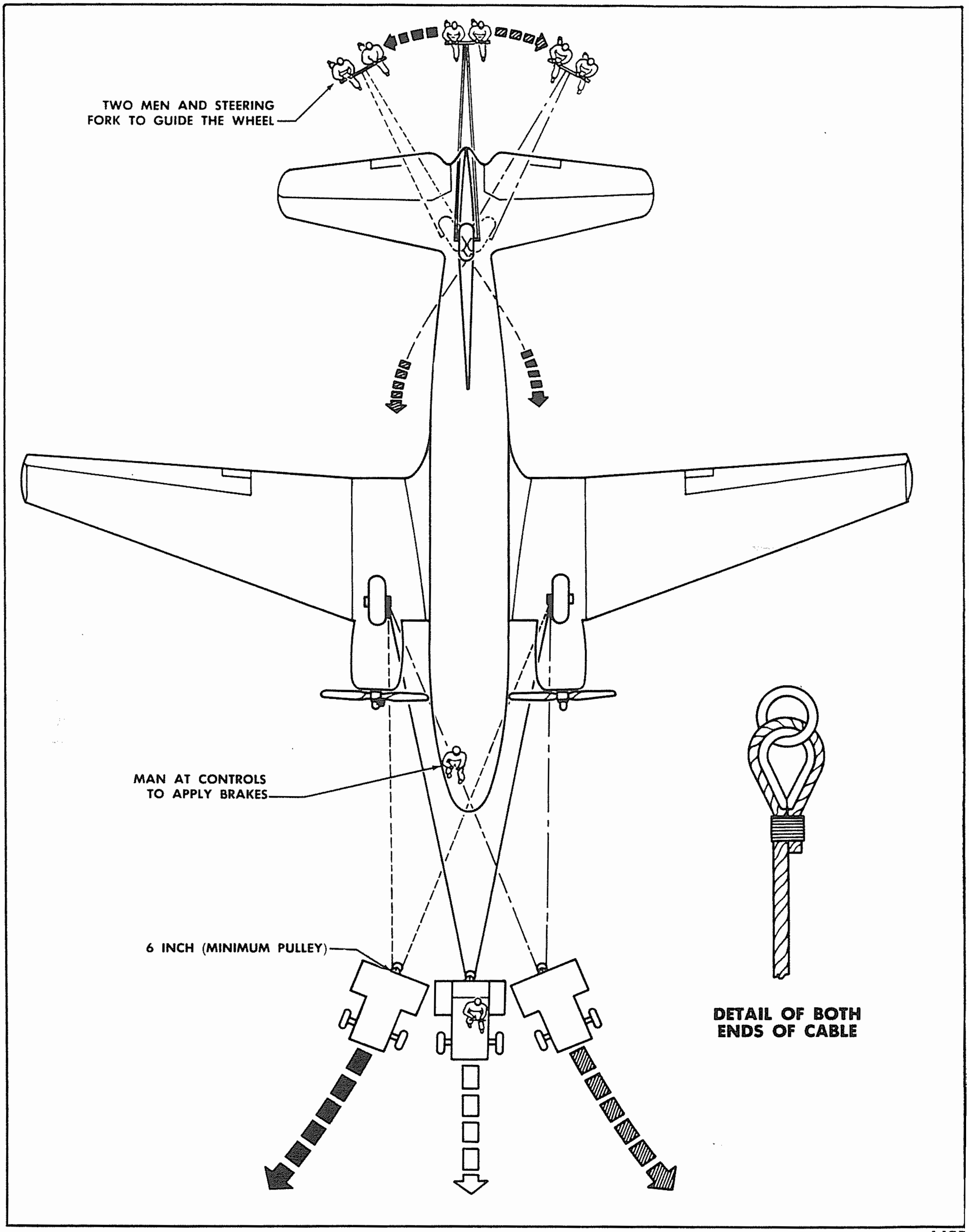

Figure 1-31. Arrangement for Towing Aircraft from the Front

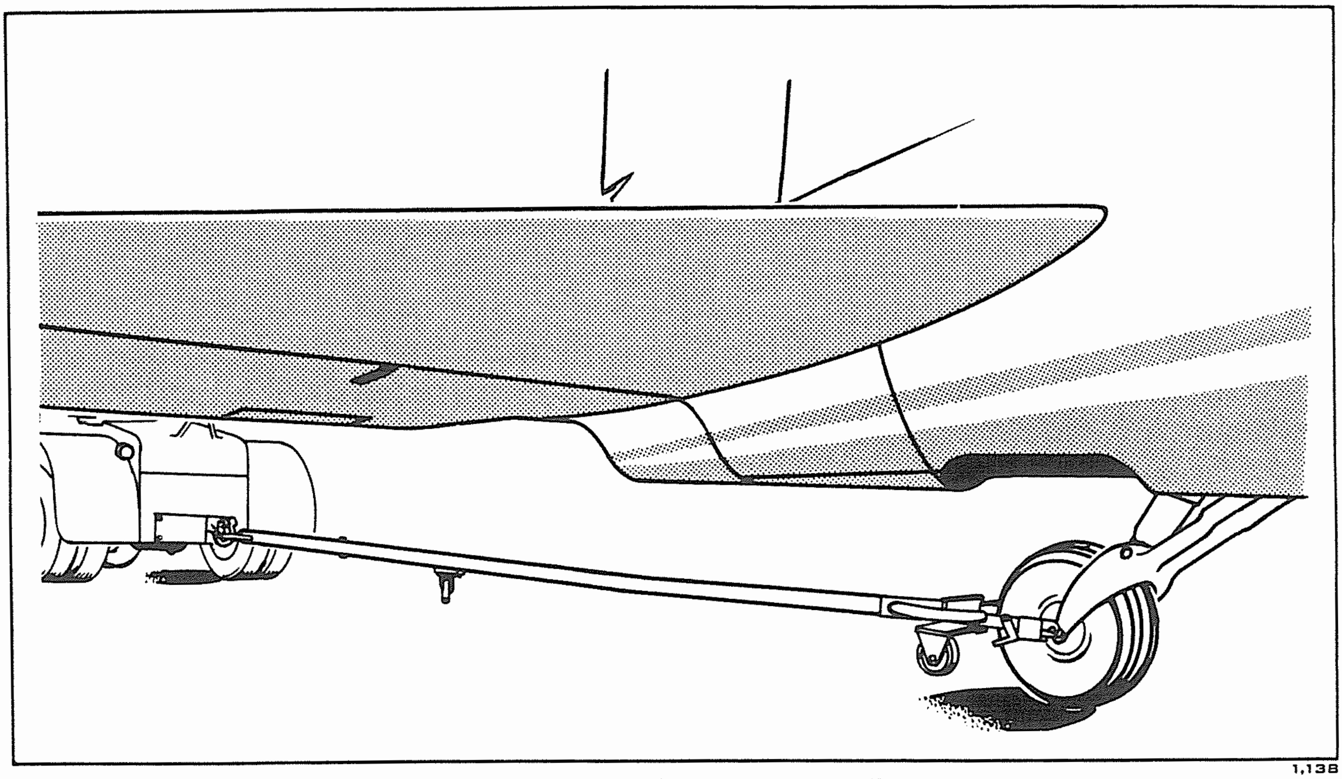

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Figure 1-32. Tail Tow Bar Installed

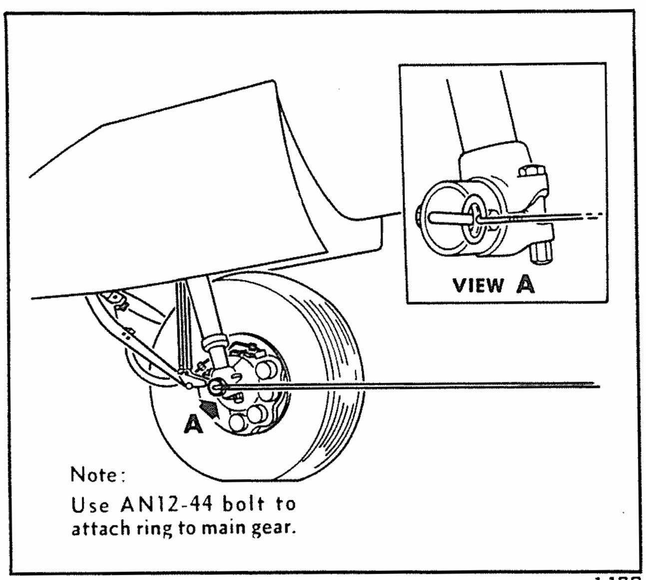

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Figure 1-33. Attachment for Towing from Front

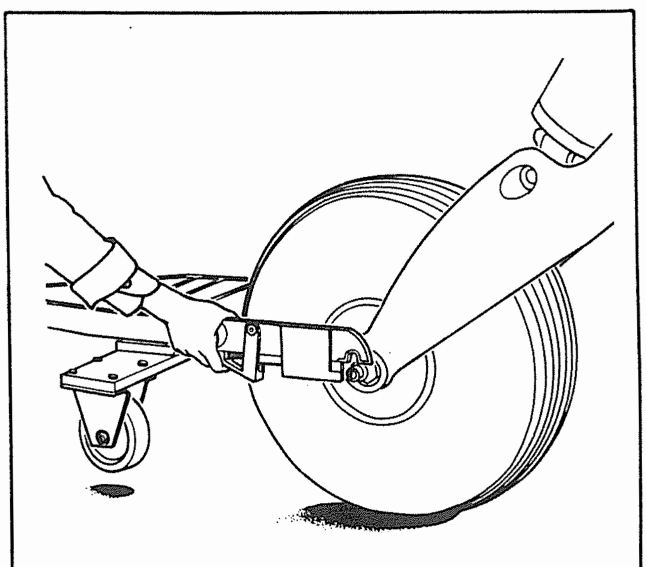

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Figure 1-34. Attaching Tow Bar to Wheel Axle

a. Remove the main gear safety ground chocks.

b. Close all doors; the door warning light on the main instrument panel should not be illuminated.

c. Fully open the cowl flaps before taxiing.

d. Check to make certain that the wing flaps are up.

e. Release the parking brake by a quick depression of the brake pedals.

f. Pull the tail wheel lock lever aft to unlock the tail wheel.

Note

Before unlocking the tail wheel, the aircraft should be rolled forward. This will relieve the locking pin from possible side load.

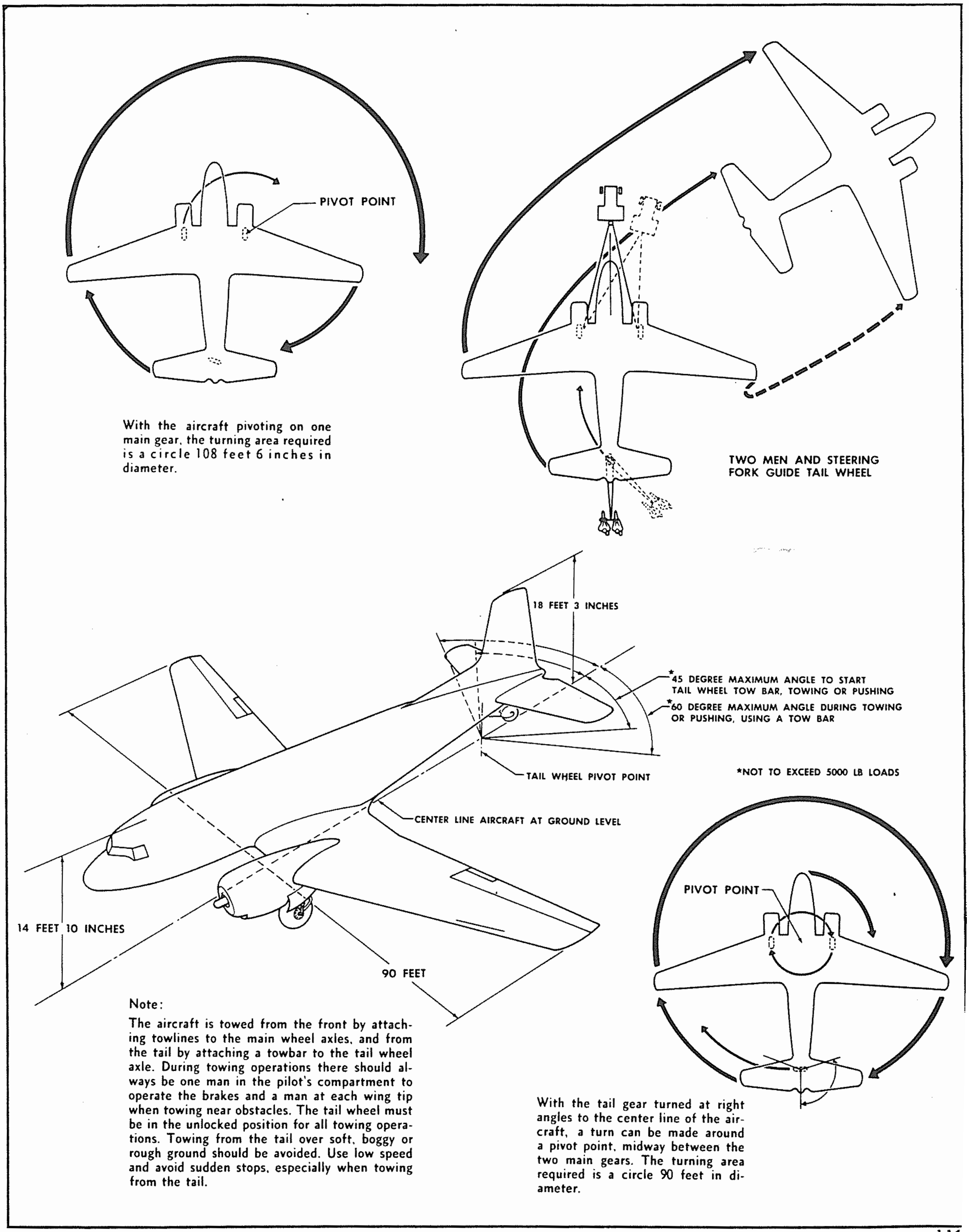

Figure 1-35. Turning Radius of Aircraft

g. Maneuver the aircraft by differential engine power or differential braking. Do not apply full brake on an alighting gear to effect a turn.

h. Avoid high taxiing speeds; use only sufficient power to achieve forward movement.

i. Avoid ruts, soft ground, etc., which will cause sudden deceleration or acceleration.

j. Watch engine temperatures carefully.

1-83. SERVICING AIRCRAFT.

1-84. SERVICING PROVISIONS. For access to servicing provisions, see figure 1-36.

1-85. FUEL SERVICING. Due to the danger involved in handling fuel, all fueling operations, including filling and draining, must be accomplished outside the hangar.

1-86. FUEL TANKS. Four main fuel tanks are provided in the wing center section; two of the tanks are installed on the left side and two on the right side of the aircraft. Five interconnected fuel cells are installed in each of the wing outer panels. The tanks have a total capacity of 1626 US. (1353.81 Imp.) gallons, consisting of 202 gallons for the front wing center section tanks, 200 gallons for each of the rear wing center section tanks, and 411 gallons for each group of 5 fuel cells in the wing outer panels.

1-87. FILLING FUEL TANKS. As a safety precaution, the aircraft must be electrically grounded during all refueling operations *(see figure 1-37)*. Use grade 100/130 fuel. Fill the fuel tanks in the following sequence: front center wing fuel tanks, rear center wing fuel tanks, and outer wing fuel cells. Filler necks are located on the upper surfaces of the wing center section and the outer wing panels. When refueling in rainy weather, place a protective cover over the filler neck. If any sizeable amount of fuel is spilled during the refueling operation, the aircraft should be towed to a safe distance before the engines are started, or sufficient time should elapse to allow for evaporation of the spillage. Make certain that all filler neck caps are secured following the refueling operation.

Note

To avoid damage to the filler necks and covers, the delivery hose must be held by personnel and must not be allowed to remain in the tank without support. If all tanks are not to be filled to capacity, fill the front center wing tanks and rear center wing tanks before filling the outer wing fuel cells.

1-88. DRAINING FUEL TANKS AND SUMPS. Small quantities of water will accumulate in the fuel tanks, due mostly to condensation within the tank itself. Being heavier than the fuel, the water seeks a lower level in the tanks and may be tapped by opening the drain cocks at the bottoms of the sumps, located on the lower sides of the center section and outer wing panels. Much of the condensation occurs soon after landing. Therefore it is advisable to service the aircraft with fuel as soon as practical after landing, thus preventing the condensation of water vapor on the inside of the fuel tanks. Fuel is removed from the front or rear center wing tanks or the outer wing fuel cells by 1 of 2 methods: by draining from the 4 drain cocks, located on the lower surface of the center wing, and from the 10 drain cocks, located on the lower surface of the outer wing panels; or by pumping through the filler necks of the tanks. Pumping is usually more satisfactory because it is faster; however, pumping will not completely drain the tanks and, therefore, the drain valves should also be opened.

1-89. FILLING OIL TANKS. Two oil tanks are provided, 1 for each engine, with capacities of 27¾ US. (22.2 Imp.) gallons each of MIL-O-6082, grade 1100 (winter or summer) oil. Additional tank space allows for oil expansion without waste. The oil tank sump is fitted with a drain cock to drain trapped water. Filler necks are accessible through an access door *(see figure 1-36)*, in the tops of the nacelles. The quantity of oil in the tank can be determined by using an oil level dip stick, which is a component part of the tank filler cap.

1-90. DRAINING OIL TANKS. To drain an oil tank, proceed as follows:

a. Provide suitable containers to hold the drained oil.

b. Place the oil emergency shutoff valve in the OFF position.

c. Disconnect the oil supply pipe below the oil emergency shutoff valve.

d. Open the shutoff valve and drain the oil into a container.

e. Remove the plugs from the engine sump and the oil cooler drain.

f. Disconnect the propeller feathering oil supply pipe at the pump.

g. Remove and clean the Cuno screen.

1-91. FILLING ALCOHOL TANK. The filler neck of the alcohol supply tank, which supplies the anti-icing alcohol for the windshields, carburetors, and propellers, is located between fuselage stations 411 and 429 on

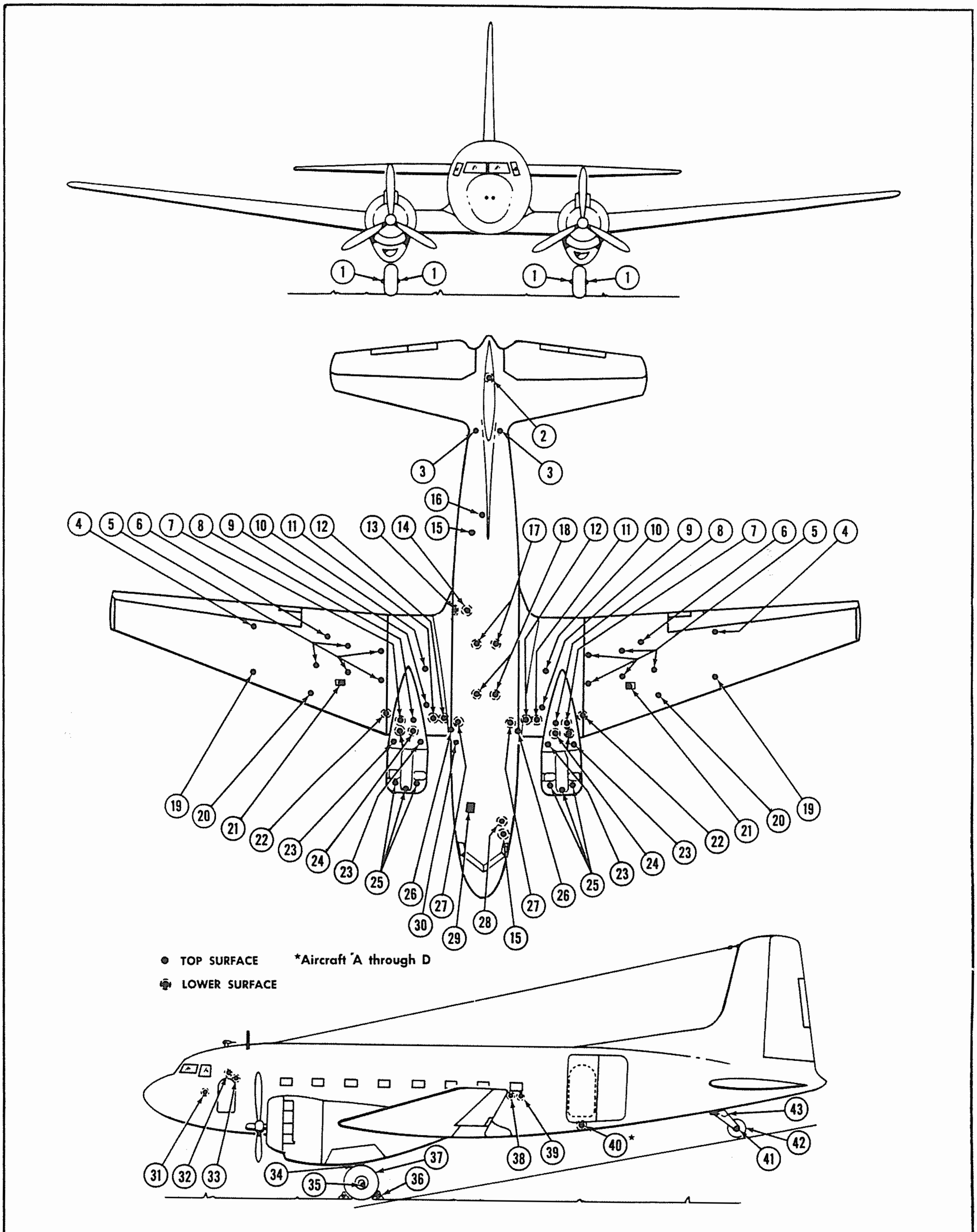

Figure 1-36. Ground Servicing and Ground Handling

Key to Figure 1-36

Index	*Unit*	*Contents*	*Specification*	*Capacity*				*Remarks*
1	Main Gear Towing Point	AN12-44 Bolt or Axle						See paragraph 1-79, and figures 1-31 through 1-35.
2	Tail Section Jacking Point	Jack Pad						Lower surface—Fuselage station 718. See figures 1-20 through 1-22.
3	Tail Section Hoisting Point	Hoist Point						Fuselage station 623. See figure 1-11 and paragraph 1-56.
4	Wing Outer Panel Hoisting Point	Hoist Point						Wing Station 173. See figure 1-12 and paragraph 1-58.
5	Fuel Drain Valves	Fuel						See paragraph 1-88.
6	Wing Outer Panel Hoisting Point	Hoist Point						Wing station 91. See figure 1-12 and paragraph 1-56.
7	Engine Oil Tank Drain	Engine Oil						Lower nacelle area. See paragraph 1-90.
				Total Usable Capacity	*Sump*	*Trap*	*Expansion*	
8	Engine Oil Tank Filler Neck (2)	Engine Oil	MIL-O-6082, Grade 1065 (Winter and Summer)	27¾ US. (22.2 Imp.) gallons (each tank)	*	*	*	Inside nacelle. See paragraph 1-89.
9	Front Fuel Tank Filler Neck (2)	Fuel	MIL-F-5572, Grade 100/130	202 US. (161.6 Imp.) gallons (each tank)	*	*	*	See paragraph 1-87. Two tanks.
10	Rear Fuel Tank Filler Neck (2)	Fuel	MIL-F-5572, Grade 100/130	200 US. (160 Imp.) gallons (each tank)	*	*	*	See paragraph 1-87. Two tanks.
11	Center Wing Jacking Point	Jack Pad						Lower surface—Center wing station 75. See figures 1-14 and 1-15.
12	Center Wing Leveling Point	One Set of Pins						Lower surface—Center wing station 70. See figure 1-26 and paragraph 1-71.
13	Anti-Icing Alcohol Tank Drain	Isopropyl Alcohol						Lower surface.
14	Anti-Icing Alcohol Tank Filler Neck	Isopropyl Alcohol	MIL-F-5566	20 US. (16.65 Imp.) gallons				Between fuselage station 411 and 429. See paragraph 1-91.
15	Oxygen System (2 Cylinders)	Oxygen						Interior. See paragraph 1-111.
16	Water Tank Filler Neck (2)	Water		7 US. (5.83 Imp.) gallons each				Upper surface—Fuselage station 411. Aircraft A through D.

*Capacities not available.

Key to Figure 1-36 (Continued)

Index	Unit	Contents	Specification	Capacity: Total Usable Capacity	Sump	Trap	Expansion	Remarks
17	Rear Fuel Tank Drain	Fuel						Lower surface. See paragraph 1-88.
18	Front Fuel Tank Drain	Fuel						Lower surface. See paragraph 1-88.
19	Wing Outer Panel Hoisting Point	Hoist Point						Wing station 169. See figure 1-12 and paragraph 1-56.
20	Wing Outer Panel Hoisting Point	Hoist Point						Wing station 119. See figure 1-12 and paragraph 1-56.
21	Outer Wing Fuel Tank Filler Cap (2)	Fuel	MIL-F-5572, Grade 100/130	411 US. (329 Imp.) gal (each side)	1.1 gal	3.0 gal	13.5 gal	See paragraph 1-87. Five tanks on each side.
22	Center Wing Jacking Point	Jack Pad						Lower surface—Center wing station 142.
23	Nacelle Hoisting Point	Hoist Point						See figure 1-11 and paragraph 1-56.
24	Emergency Axle Jacking Point	Two Jacking Points for Each Gear						See figures 1-23 and 1-24, and paragraph 1-69.
25	Engine Hoisting Point	Three Hoist Fittings for Each Engine						See figure 1-13 and paragraph 1-62.
26	Fuselage Hoisting Point	Two Hoist Points						Fuselage station 202. See paragraph 1-57.
27	Center Wing Leveling Point	One Set of Leveling Pins						Center wing station 47. See figure 1-26 and paragraph 1-71.
28	Ground Blower	Blower						See section IV, Heating and Ventilating.
29	Battery	Distilled or Clear Drinking Water		Fill to 3/8 inch above protector plate				See figure 1-8 and paragraph 1-108.
30	D-C External Power Receptacle	Receptacle						See figure 1-8, items 42 and 43.
31	Air Brake Cylinder	Air		1590 psi				Interior. See figures 1-41 and 2-57, and paragraph 1-112.
32	Hydraulic Fluid Reservoir Filler Neck	Hydraulic Fluid	MIL-O-5606	2.3 US. (2.01 Imp.) gallons				Interior. See figures 1-38, 2-53, and 3-2; see also paragraph 1-96.
33	Main Hydraulic Pressure Accumulator	Air		350 psi				Interior. See figures 1-38 and 2-53, and paragraph 1-98.

Key to Figure 1-36 (Continued)

Index	*Unit*	*Contents*	*Specification*	*Capacity*	*Remarks*
34	Shock Strut Main Gear	Air and Hydraulic Fluid	MIL-O-5606		See figure 1-39; see also paragraphs 1-99 through 1-101, and paragraph 1-104.
35	Hydraulic Brake Bleed Hole				One for each brake. See figure 2-54 and paragraph 2-436.
36	Wheel Chocks				
37	Main Gear Tire	Air		60 psi	All operations irrespective of gross weight.
38	Fuselage Leveling Point	One Set of Leveling Pins			Right side—Station 389. See figure 1-26 and paragraph 1-71.
39	Fuselage Leveling Point	One Set of Leveling Pins			Right side—Fuselage station 410. See figure 1-26 and paragraph 1-71.
40	Toilet Service Panel	Servicing Panel			Aircraft A through D. See paragraph 2-298.
41	Tail Gear Towing Point	Towing Point			See figures 1-31 through 1-34, and paragraph 1-79.
42	Tail Wheel Tire	Air		70 psi	All operations irrespective of gross weight.
43	Shock Strut Tail Wheel	Air and Hydraulic Fluid	MIL-O-5606		See figure 1-40, and paragraphs 1-103 and 1-105.

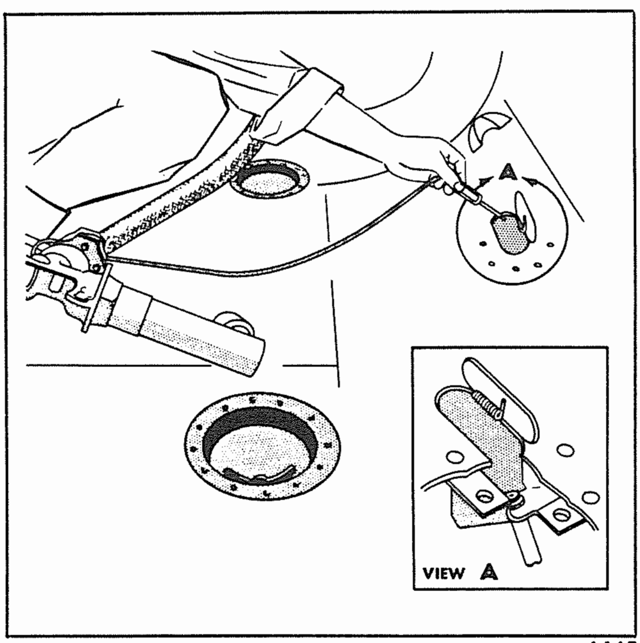

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Figure 1-37. Bonding Fuel Discharge Nozzle to Aircraft

the right side. Open the filler neck access door *(see figure 1-36)*, and fill the tank with 20 US. (16.65 Imp.) gallons of MIL-F-5566 isopropyl alcohol (see paragraph 1-18).

1-92. FILLING WATER SUPPLY TANK (AIRCRAFT 1 THROUGH 96). The wash basin supply tank is installed on the forward face of the bulkhead at station 583. The tank has a capacity of 3 US. (2.50 Imp.) gallons. To fill with water, remove the cap on the top of the tank *(see figure 1-36).*

1-93. FILLING WATER SUPPLY TANK (AIRCRAFT A THROUGH D). The lavatory and buffet water supply tanks are located in the lavatory area at approximately station 411. The tanks are filled through an access hole on the top of the fuselage, to the right of the dorsal fin. The hot and cold water tanks each have a capacity of 7 US. (5.83 Imp.) gallons *(see figure 1-36).*

1-94. DRAINING WATER SUPPLY TANK (AIRCRAFT 1 THROUGH 96). The tank can be drained by operating the valve at the wash basin.

1-95. DRAINING WATER SUPPLY TANK (AIRCRAFT A THROUGH D). Both the hot and cold water units can be drained by operating the valve on the lower side of the fuselage *(see figure 1-9).*

1-96. FILLING HYDRAULIC RESERVOIR. The hydraulic fluid reservoir has a capacity of 2.3 US. (1.91 Imp.) gallons and is installed in the fuselage accessories compartment. The reservoir is equipped with a sight gage and a placard which contains instructions for refilling. The filler neck is located adjacent to the sight gage at the top of the hydraulic panel. Fill the reservoir with mineral oil hydraulic fluid, Specification MIL-O-5606. The consumption of hydraulic fluid is ordinarily negligible.

Note

To avoid contamination of the hydraulic system, use only fluid which has been filtered through a paper-type micronic filter.

1-97. DRAINING HYDRAULIC RESERVOIR.

a. Provide a container of sufficient size to hold the fluid drained from the reservoir and a suitable length of hose to be connected between the container and the pipe from the reservoir.

b. Disconnect the fluid supply pipe at the hand pump.

c. Quickly connect the length of hose to the supply pipe, and drain the contents of the reservoir into the container.

d. Reconnect the supply pipe to the hand pump.

1-98. INFLATING MAIN PRESSURE ACCUMULATOR. *(See figure 1-38.)* Make certain that the main hydraulic system pressure is zero by operating the wing flaps. Inflate the accumulator to 350 psi following the procedure outlined in paragraph 1-100.

WARNING

Do not inflate higher than 350 psi at any time.

1-99. INFLATING MAIN GEAR SHOCK STRUT. *(See figure 1-39.)*

1-100. To charge the strut with air, or to discharge air using gage assembly AN6286-1 or AN6286-2, proceed as follows:

a. Remove the valve cap.

b. Attach the air filling chuck to the valve by means of the gage assembly.

c. Loosen the ⅝-inch hex swivel nut to a maximum of ¾ of a complete turn. Turn counterclockwise to loosen. To prevent injury to personnel, do not loosen the larger nut with pressure in the strut.

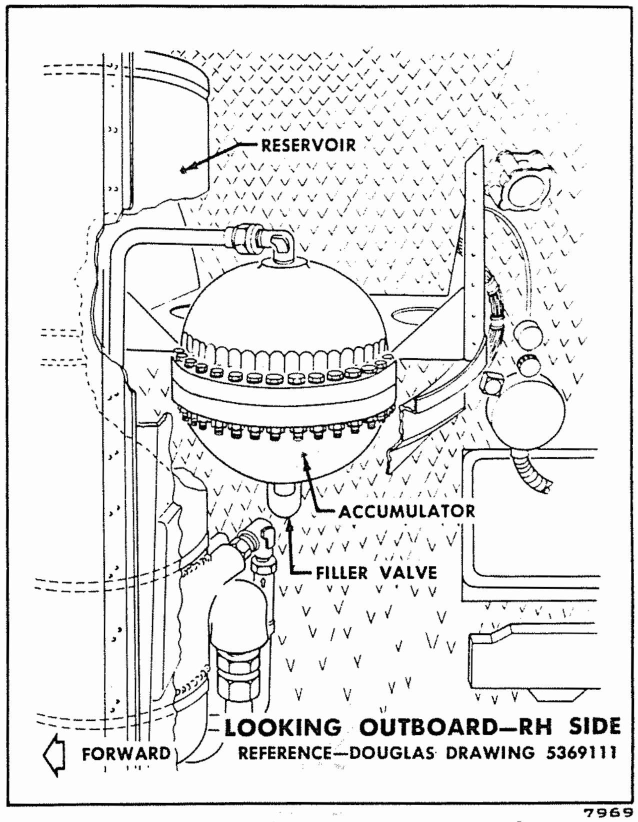


Figure 1-38. Main Pressure Accumulator

CAUTION

Excessive loosening will result in the stem assembly dropping into the unit to which the valve is attached. Furthermore, the ⅝-inch hex nut will interfere with the chuck on the gage assembly and cause damage to the valve core housing.

d. Charge or discharge to the proper strut extension *(see figure 1-39).*

WARNING

Inflate the shock struts with dry compressed air or dry nitrogen only. The use of other gases for this purpose may result in serious damage.

e. Tighten the ⅝-inch hex swivel nut to 50 to 70 inch-pounds torque. Turn clockwise to tighten.

f. Remove the gage assembly from the valve; replace and tighten the valve cap to extreme finger-tightness.

1-101. To charge the strut with air without using a gage assembly, proceed as follows:

a. Remove the valve cap.

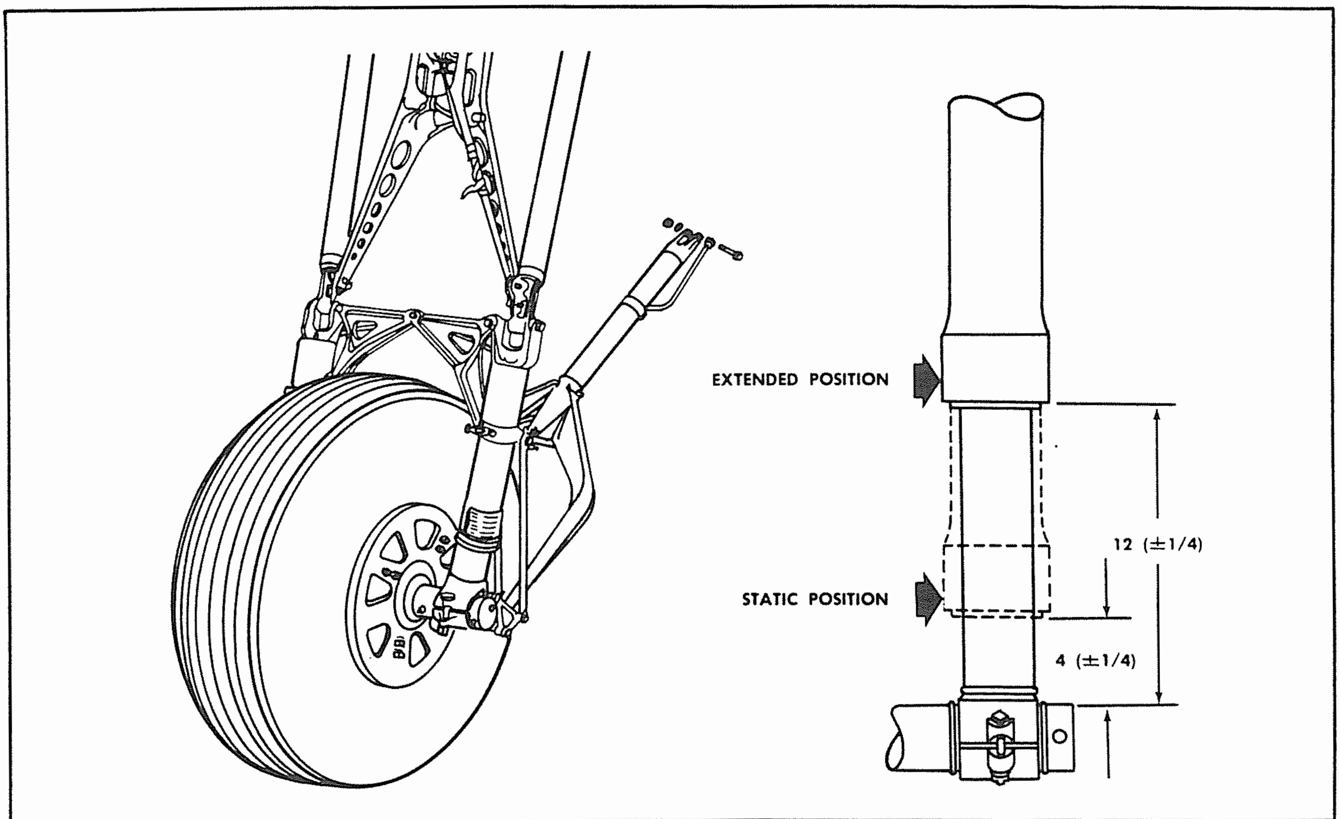


Figure 1-39. Inflating Main Gear Shock Strut

b. Attach the air filling chuck from the booster pump.

c. Loosen the ⅝-inch hex swivel nut to a maximum of three-quarters of a complete turn. Turn counterclockwise to loosen. To prevent injury to personnel, do not loosen the larger nut with pressure in the strut.

CAUTION

Excessive loosening will result in the stem assembly dropping into the unit to which this valve is attached. Furthermore, the ⅝-inch hex will interfere with the chuck and cause damage to the valve core housing.

d. Charge the strut to the proper strut extension *(see figure 1-39).*

WARNING

Inflate the shock struts with dry compressed air or with dry nitrogen only. The use of other gases for this purpose may result in serious damage.

e. Tighten the ⅝-inch hex swived nut to 50 to 70 inch-pounds torque. Turn clockwise to tighten.

f. Remove the air filling chuck from the valve; replace and tighten the valve cap to extreme finger-tightness.

1-102. To discharge air from the unit without using a gage assembly, proceed as follows:

a. Remove the valve cap.

b. Loosen the ⅝-inch swivel nut. The amount the nut is loosened, up to one complete turn only, will govern the rate of discharge of the air.

c. Depress the valve core with a suitable tool, comparable in size to a match stick. If only a small amount of air is to be discharged, loosen the ⅝-inch hex swivel nut about one-tenth of a complete turn and depress the valve core *(see figure 1-39).*

d. Tighten the ⅝-inch hex swivel nut to 50 to 70 inch-pounds torque. Turn clockwise to tighten.

e. Replace and tighten the valve cap to extreme finger-tightness.

1-103. INFLATING TAIL GEAR SHOCK STRUT. To inflate the tail gear shock strut, follow the instructions outlined in paragraph 1-100. See figure 1-40 for correct strut extension.

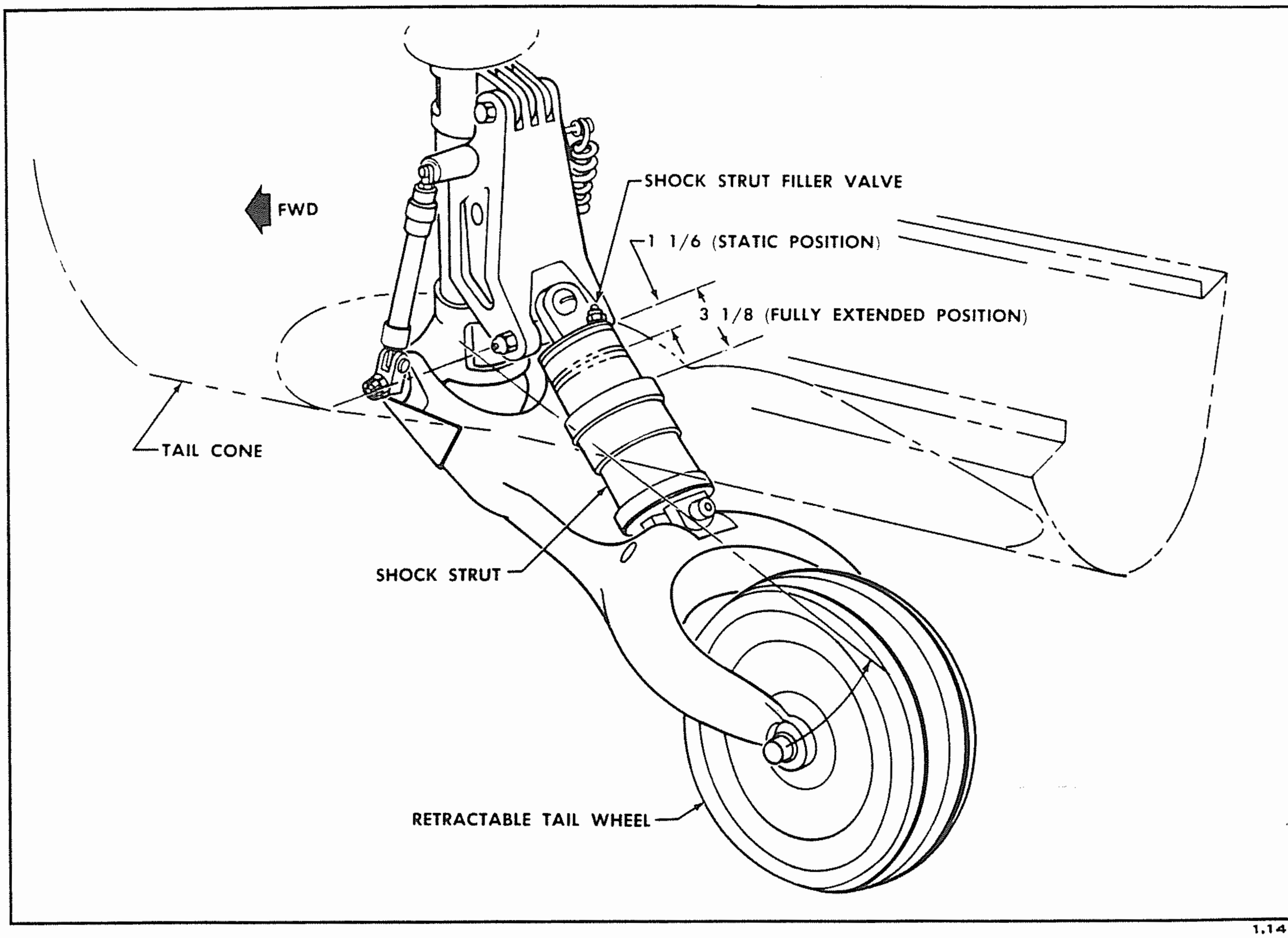


Figure 1-40. Inflating Tail Gear Shock Strut

1-104. FILLING MAIN GEAR SHOCK STRUT. To fill the main gear shock struts, proceed as follows:

a. Deflate the struts in accordance with the instructions contained in paragraphs 1-100 through 1-102.

b. Make certain that both struts of the pair are fully compressed.

c. Remove the filler plug from each strut.

d. Add fluid to each strut until the level reaches the filler hole. Use fluid, Specification MIL-O-5606.

e. Install the filler plugs.

1-105. FILLING TAIL GEAR SHOCK STRUT. To fill the tail gear shock strut, proceed as follows:

a. Deflate the strut in accordance with the instructions contained in paragraphs 1-100 through 1-102.

b. Make certain that the strut is fully compressed.

c. Remove the filler plug at the top of the strut.

d. Add fluid until the level reaches the filler hole. Use fluid, Specification MIL-O-5606.

e. Install the filler plug.

1-106. INFLATING MAIN GEAR TIRES. Inflate the main gear tires to 60 psi for maximum take-off gross weight.

1-107. INFLATING TAIL GEAR TIRE. Inflate the tail gear tire to 70 psi.

1-108. FILLING BATTERIES. The battery access doors are located on the lower surface of the fuselage nose section *(see figure 1-9)*. The level of the electrolyte should be 3/8 inch above the protector plate. Add distilled water with a self-leveling syringe.

1-109. SERVICING HAND FIRE EXTINGUISHERS. The two hand fire extinguishers are installed in the interior of the aircraft. One is located forward of the main cabin door, just above the floor level; the other is installed in the flight compartment, just aft of the co-pilot. To fill an extinguisher, remove the plug on the top of the cylinder and fill with an approved fire extinguisher agent. Hand fire extinguishers do not require periodic refilling, but periodic checks must be made to determine whether loss of fluid has taken place. Refill a cylinder immediately after use. Keep the cylinder filled to capacity.

1-110. CHARGING CO_2 CYLINDERS.

WARNING

A charged CO_2 cylinder can be very dangerous if gas is accidentally released while the cylinder is not clamped in place. When a cylinder is removed, always remove the control head from the flood valve first; when a cylinder is installed, replace the control head last. When disconnecting the flexible hose, disconnect it at the flood valve.

The correct adapters and blow-off valves (for 3/4-inch cylinder flood valves) must be a part of the charging assembly. Remove the cylinders from the fuselage compartment aft of station 47 and charge as follows:

Note

Use only dry CO_2. Any moisture will freeze in the tubes and valves under the extreme cold found at high altitudes, possibly rendering the extinguisher inoperative.

a. Connect the CO_2 supply to the outlet of the cylinder valve.

b. Place the cylinder on a scale and note its exact weight.

c. Open the CO_2 supply valve; charge the cylinder with 12.6 pounds of CO_2. Perform this operation carefully, to prevent overcharging of the cylinder. If the cylinder should be overcharged, reduce the pressure by using a blow-off valve, which can be screwed to the top (control head outlet) of the cylinder valve. When reducing to the desired weight, make certain that the weight of the blow-off valve (approximately seven ounces) is considered. When the cylinder charge is reduced to the proper weight, remove the blow-off head.

d. With the CO_2 charge at the proper weight, shut off the recharging pump, close the CO_2 supply valve, loosen the supply pipe at the cylinder valve, bleed the pipe, and disconnect it. The cylinder valve pilot check is closed during the charging.

e. To test the cylinder valve for leakage, place the cylinder in a horizontal position so that the discharge outlet is turned up; test the main valve for leakage by pouring water into the outlet, filling it almost to the top. Use care to prevent water from spilling over the top of the outlet, as it may enter the valve passages and prevent operation. The appearance of bubbles indicates leakage at the main check.

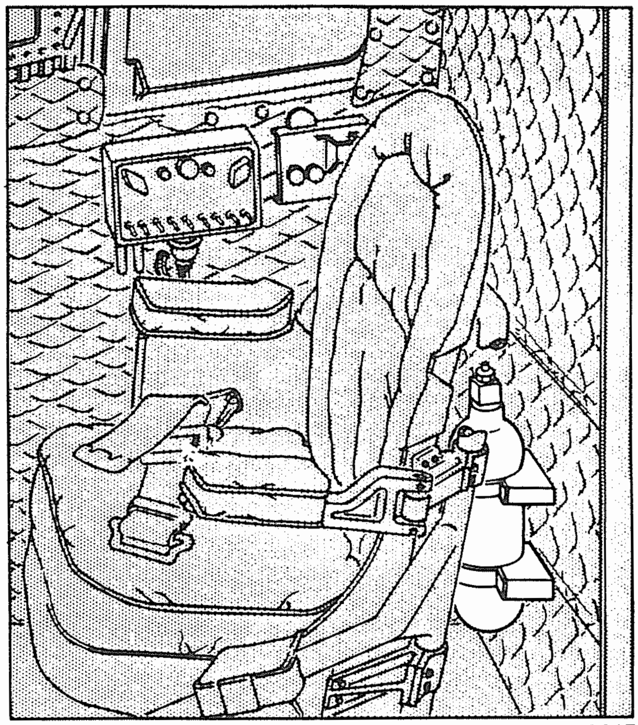


Figure 1-41. Brake Emergency Air Cylinder

f. Place the valve cover *only* (including the safety disc retainer) into a small cup of water. Bubbles will indicate leakage at the safety seal.

g. Tilt the cylinder so that the hole at the side of the control head connection well (which leads to the valve piston head) is higher than the hole in the center of the well (pilot check stem hole). With the valve held in this position, pour enough water into the control head connection well to cover the center hole, but do not permit it to run into the side hole. Water trapped in the small passage leading from the side hole will freeze at high altitudes and possibly prevent operation of the valve. Bubbles in the water covering the pilot check stem hole indicate leakage at the pilot check.

h. After completing water tests of the valves, wipe all parts perfectly dry. Any spots which cannot be reached with the drying cloth should be dried with compressed air.

i. Replace the CO_2 cylinders in the fuselage compartment.

1-111. CHARGING HIGH-PRESSURE OXYGEN CYLINDER.

The high-pressure oxygen cylinder is *not* to be charged by service personnel. When the charge in the cylinder is sufficiently depleted, remove the cylinder from its position behind the pilot's seat and replace it with a fully charged cylinder. For the cylinder removal and installation procedure, see paragraph 4-189.

1-112. CHARGING BRAKE EMERGENCY AIR CYLINDER. *(See figure 1-41.)* The brake emergency air cylinder is located directly behind the co-pilot's seat. The air cylinder may be filled while it is in position in the aircraft. Attach the air supply hose to the cylinder and fill until the pressure gage to the right of the main instrument panel indicates 1590 psi.

Note

Make certain that the air-brake cylinder is refilled after each application of the emergency system.

1-113. SERVICING TOILET. For a detailed description of the procedure for servicing the toilet, see paragraph 2-298.

1-114. CLEANING AIRCRAFT EXTERIOR. For a detailed description of the procedure to be followed in cleaning the aircraft exterior, see paragraph 2-4.

1-115. CLEANING AIRCRAFT INTERIOR. For a detailed description of the procedure to be followed in cleaning the aircraft interior, see paragraph 2-243.

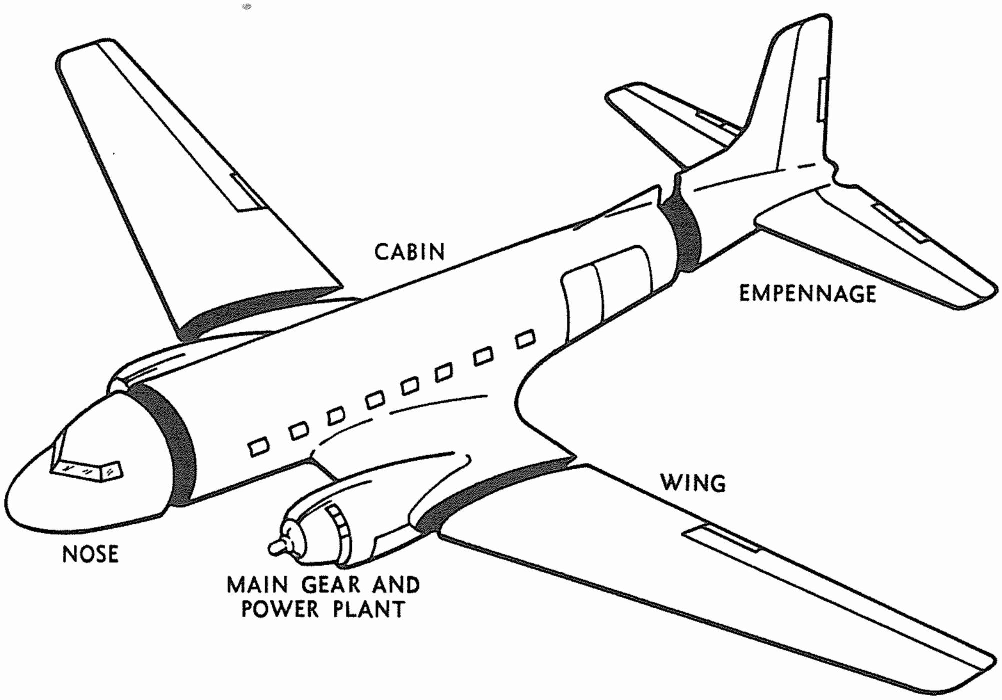
1-116. LUBRICATION REQUIREMENTS.

1-117. For lubrication requirements, see figure 1-42.

1-118. SPECIAL TOOLS AND HANDLING EQUIPMENT.

1-119. The special tools and handling equipment necessary for aircraft line maintenance operations are illustrated in figures 1-43 and 1-44. Illustrations showing the application of the tools appear in this handbook under the sections or paragraphs pertinent to a particular part of the aircraft or the equipment concerned.

R4D-8 LUBRICATION CHART



INDEX

Specific sections of the aircraft can be easily located by the diagram and index on this page. The illustrations on the following pages show the mechanisms to be lubricated, the lubricant to be used, the method of application, and the frequency intervals.

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Figure 1-42 (Sheet 1 of 12 Sheets). Lubrication Chart

GENERAL NOTES

1. TEMPERATURE CONSIDERATION:

a. Specified lubrication is designed for operating range of —46°C (—50°F) to 71°C (160°F).

2. ABNORMAL OPERATING CONDITIONS:

a. Lubrication frequency intervals expressed are based on an average of 100 flight hours per month. These intervals, however, can be revised to accommodate any abnormal condition that may be encountered, such as more or less hours of service per month, extremely wet or salt air, and/or dusty climates, and inactive or storage time. Where inactive or storage time is the consideration and long-time preservation of the ship is not contemplated, lubricate according to the following table:

At the following periods during idleness of airplane	Service points indicated on lubrication chart by following frequency symbol
Once every month	Daily to intermediate
Every two months	Intermediate
Every three months	Major

3. LUBRICATION APPLICATION:

a. Cleanliness is essential to good lubrication procedure. Clean and dry grease guns and oil cans before filling. Wash brushes, if dirty, in Stoddard-type solvent and dry thoroughly before use. Use only lubricants which are perfectly clean.

b. Wipe grease fittings, oil holes, etc., with clean, dry cloths before lubricating. Remove any dirt or rust from surfaces to be lubricated.

c. Work moving parts, if practical, to assure thorough lubrication.

d. Force grease into fittings until all old grease is extruded, unless otherwise noted.

e. Fill oil holes to near full, adding oil until the level does not drop. Controllable-type squirt oil cans are recommended for oil application.

f. After any lubrication, clean surplus lubrication from all but actual working surfaces.

4. ANTI-FRICTION BEARINGS:

a. Unless otherwise noted herein, grease-lubricated anti-friction bearings need not be relubricated except at major overhaul. Do not attempt to oil such bearings.

b. Keep exposed anti-friction bearings as clean as possible. Wipe with a clean cloth dampened with Stoddard-type solvent.

c. Do not expose anti-friction bearings to steam cleaning.

5. CONTROL CABLES:

Every 200-260 hours perform the following:

a. Inspect cable coatings on lengths running on pulleys, drums, and in fairleads. Touch up where necessary with corrosion preventative CHP.

b. Apply oil to all sections of cable as mentioned above with lubricant OGP.

c. Apply oil lubricant OGP to cable clevis attach points.

6. MISCELLANEOUS:

a. This lubrication chart does not cover engines, instruments, and certain accessories. Follow manufacturer's recommendations or consult the Douglas representative.

b. Apply lubricant GLT by hand to control surface adjusting screws as necessary.

c. Felt wipers installed at piston rod end of hydraulic actuating cylinders must be saturated with hydraulic oil every intermediate frequency. If oil hole is not provided in cylinder end or packing nut, back off nut to permit oiling.

d. Lubricate all points shown on lubrication chart at major overhaul. This is in addition to specified periodic service.

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Figure 1-42 (Sheet 2 of 12 Sheets). Lubrication Chart

LUBRICATION FREQUENCY SYMBOLS

DAILY | INTERMEDIATE | MAJOR | SPECIAL

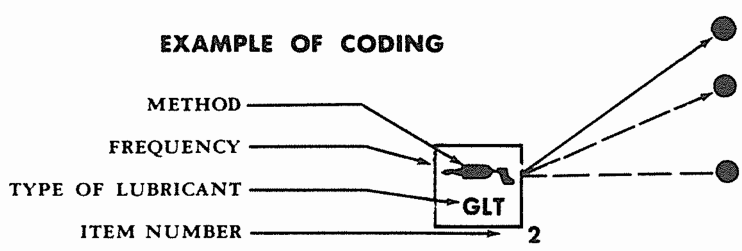
APPLICATION SYMBOLS

OIL CAN | PRESSURE GUN | HAND | BRUSH

TABLE OF LUBRICANTS

SYMBOL	TYPE OF LUBRICANT	PRINCIPAL USES (herein)	DOUGLAS RECOMMENDED PRODUCTS	PROCUREMENT SPECIFICATION
GH	Grease — High-Temperature	Wheel bearings	Commercially available	MIL-L-3545 (Was AN-G-5)
GG	Grease — Graphite	Infrequent service of exposed mechanisms	Commercially available (to specification)	MIL-G-7187 (Was AN-G-6)
GLT	Grease — Low-Temperature	Prelubricated bearings (sealed, etc.)	Commercially available (to specification)	MIL-G-3278 (AN-G-25)
OGP	Oil — General Purpose	General oil can lubrication	Commercially available (to specification)	MIL-L-7870 (Was AN-O-6)
FGS	Graphite — Stick	For sliding surfaces where "dry" lubrication is desired	Dixon G-7-11 Graphite Stick (J. Dixon Crucible Co.)	MIL-L-7870
FGL	Graphite — Lacquer Base	For cowl flap rub strips	Lubetex No. 10 (Lubrium Oil Co., Los Angeles, Calif.)	None
CSP	Compound — Anti-Seize, Graphite Base	For high-temperature, limited motion items such as exhaust manifold brackets, etc.	Commercially available (to specification)	MIL-C-5544 (AN-C-147)
CHP	Compound — Corrosion-Preventive, Exterior	For protection of cables	Commercially available (to specification)	MIL-C-6708 (AN-C-52) Type I

EXAMPLE OF CODING



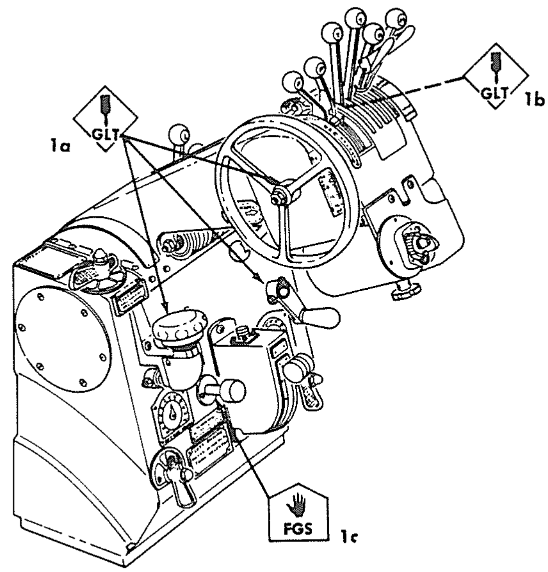
The example above indicates item (2) to be lubricated daily by pressure gun, using lubricant GLT.

- Solid line with arrowhead indicates a specific lubrication point.
- Broken line with arrowhead indicates a specific lubrication point and identical point or points of the item, not shown.
- Omission of arrowhead on broken lines indicates localized area requiring lubrication, the point or points of which are not specifically shown.

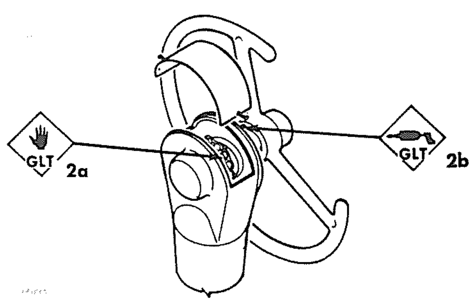
The number of items requiring lubrication and the number of lubricating points of the item are noted under "Items to be Lubricated and Remarks."

1,150

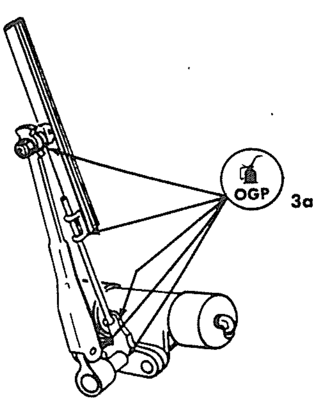
Figure 1-42 (Sheet 3 of 12 Sheets). Lubrication Chart



1 — CONTROL PEDESTAL

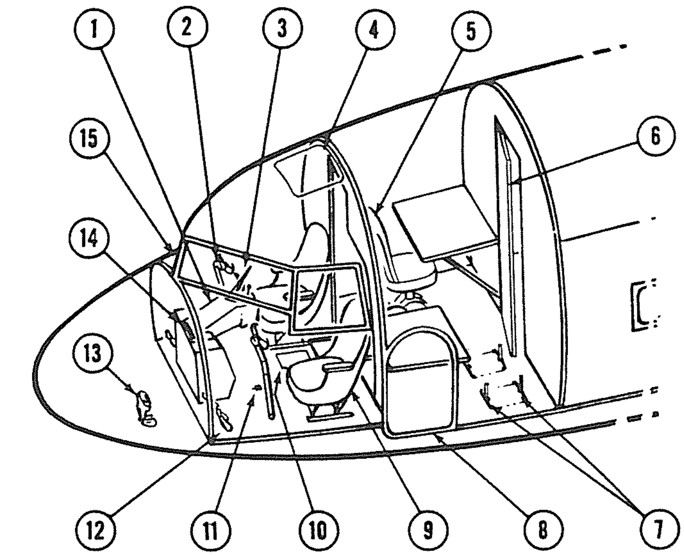


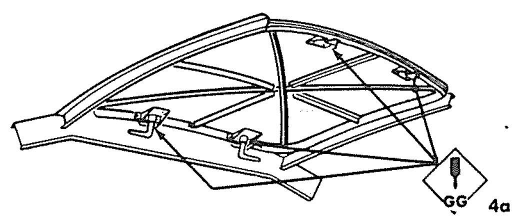
2 — CONTROL COLUMN



3 — WINDSHIELD WIPER

NOSE





4 — EMERGENCY EXIT

FREQUENCY SYMBOLS

INTERMEDIATE MAJOR

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
GLT	MIL-G-3278	Grease — Low Temperature
OGP	MIL-L-7870	Oil — General Purpose
FGS	None	Graphite — Stick
GG	MIL-G-7187	Grease — Graphite

ITEMS TO BE LUBRICATED AND REMARKS

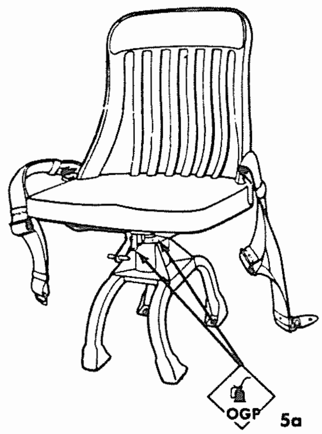
1. CONTROL PEDESTAL:
   a. Aileron, elevator and rudder tab gear train — apply light coat of grease.
   b. Lever and lever latch sliding surfaces — apply a coat of grease.
   c. Parking brake control rod sliding surface — apply light film of stick graphite.
2. CONTROL COLUMN (TWO REQUIRED):
   a. Chain and sprocket — remove cover to apply grease.
   b. Control wheel shaft bearings — apply to one grease fitting.
3. WINDSHIELD WIPER (TWO REQUIRED):
   a. Wiper turning points — oil lightly, wipe off excess.
4. EMERGENCY ESCAPE HATCH:
   a. Latch and catches mating surfaces — apply light coat of grease.

(Continued on next page)

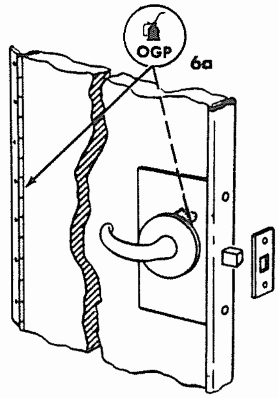
1,151

Figure 1-42 (Sheet 4 of 12 Sheets). Lubrication Chart

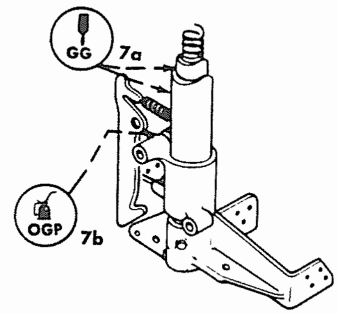
NOSE



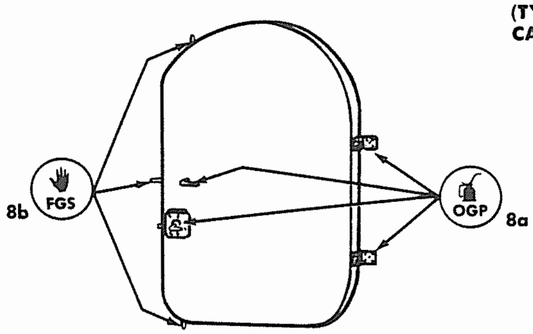
5 — RADIO OPERATOR'S SEAT



6 — INTERIOR DOORS (TYPICAL IN NOSE AND CABIN AREAS)



7 — BATTERY SUPPORT ASSEMBLY

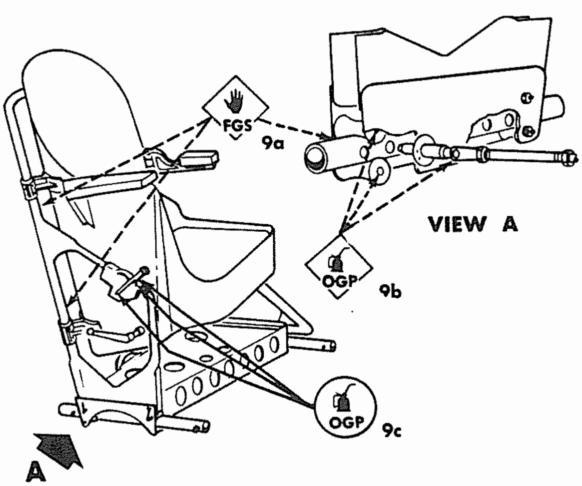


8 — SERVICE DOOR

FREQUENCY SYMBOLS

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
OGP	MIL-L-7870	Oil — General Purpose
GG	MIL-G-7187	Grease — Graphite
FGS	None	Graphite — Stick



9 — PILOT'S SEAT

ITEMS TO BE LUBRICATED AND REMARKS

(Continued from preceding page)

5. **RADIO OPERATOR'S SEAT:**
   a. Swivel mating surfaces and locking mechanism — oil lightly.
6. INTERIOR DOORS:
   a. Latch and hinge turning points — oil lightly, wipe off excess. Do not allow oil to come in contact with finished surfaces.
7. **BATTERY SUPPORT ASSEMBLIES (TWO):**
   a. Guide tubes, sliding surfaces — apply a light coat of grease.
   b. Locking mechanism turning points — oil lightly.
8. **SERVICE DOOR:**
   a. Hinges latch turning mechanism and catch hold-open mechanism turning points — oil lightly.
   b. Latch bolt sliding surface — apply light film of graphite stick.
9. **PILOT'S SEAT (TWO):**
   a. Forward and aft slide tubes, up and down slide tubes — apply light film of stick graphite.
   b. Slide roller pins, slide mechanism turning points, lock pin mechanism turning points — oil lightly.
   c. Shoulder harness inertia reel control lock assembly sliding surfaces and turning point — oil lightly.

(Continued on next page)

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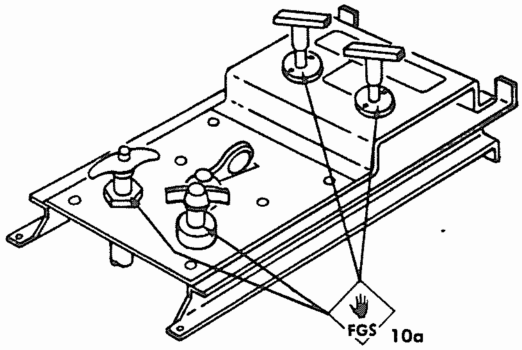
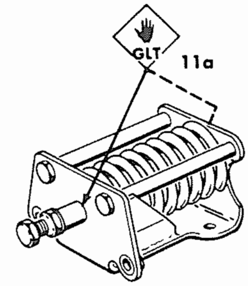
Figure 1-42 (Sheet 5 of 12 Sheets). Lubrication Chart

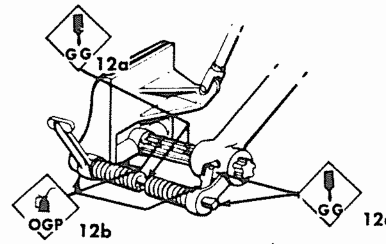
NOSE



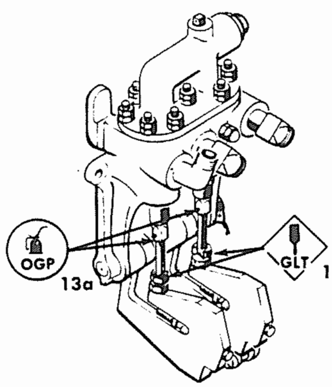
10 — FIRE EMERGENCY CONTROL PANEL



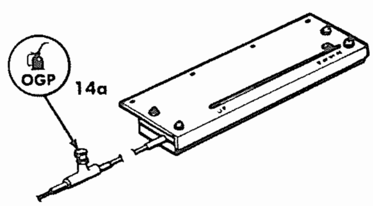
11 — ELEVATOR CONTROL COLUMN STOP



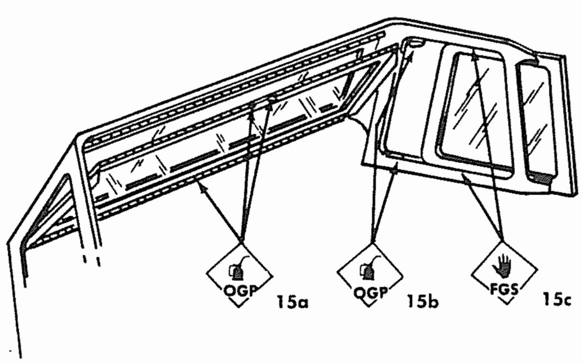
12 — RUDDER PEDALS



13 — POWER BRAKE CONTROL VALVE



14 — WING FLAP INDICATOR TUBE



15 — FLIGHT COMPARTMENT WINDOWS

FREQUENCY SYMBOLS

INTERMEDIATE MAJOR

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
OGP	MIL-L-7870	Oil — General Purpose
GG	MIL-G-7187	Grease — Graphite
GLT	MIL-G-3278	Grease — Low Temperature
FGS	None	Graphite — Stick

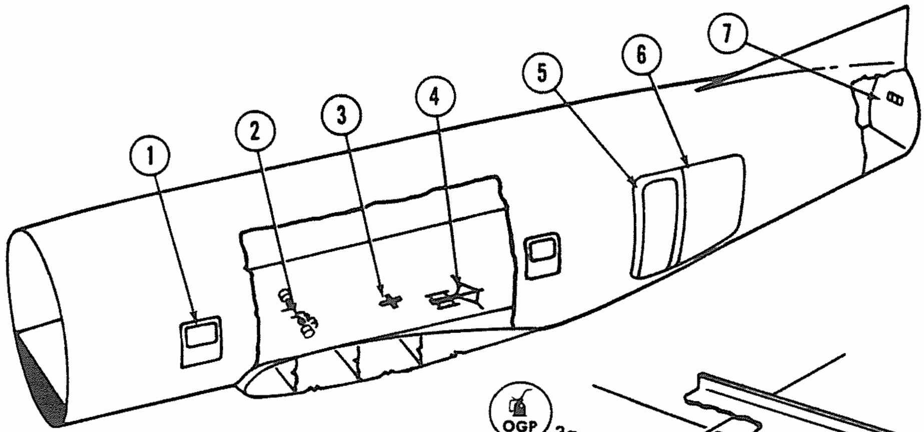
ITEMS TO BE LUBRICATED AND REMARKS

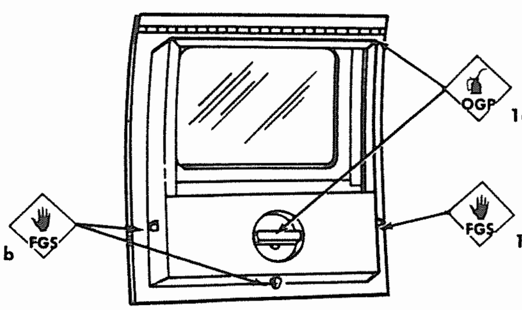
10. **FIRE EMERGENCY CONTROL PANEL:**
    a. Control rods (four) — apply light film of stick graphite to sliding surfaces.
11. **ELEVATOR CONTROL COLUMN STOP:**
    a. Spring sliding shaft — both ends — apply light coat of grease.
12. **RUDDER PEDALS (FOUR):**
    a. Pedal shaft splines — apply light coat of grease.
    b. Shaft turning points — oil lightly.
    c. Locking pin and torque shaft sliding surface — apply light film of grease.
13. **POWER BRAKE CONTROL VALVE:**
    a. Piston rod-link attach point (two) — oil lightly.
    b. Links (two) ball end — apply light coat of grease.
14. **WING FLAP INDICATOR TUBE:**
    a. Add oil slowly at fitting in line directly below indicator; fill to level of fitting.
15. **FLIGHT COMPARTMENT WINDOWS:**
    a. Push-pull rods at top and hinge pins at bottom — oil lightly, wipe off excess.
    b. Guide roller pins (two) — oil lightly, wipe off excess.
    c. Sliding area, top and bottom — apply light film of graphite stick.

1,153

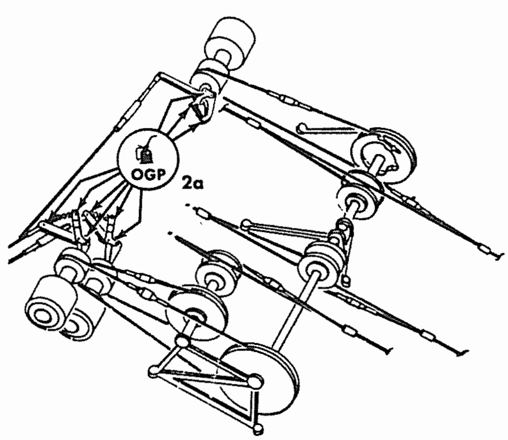
Figure 1-42 (Sheet 6 of 12 Sheets). Lubrication Chart

CABIN

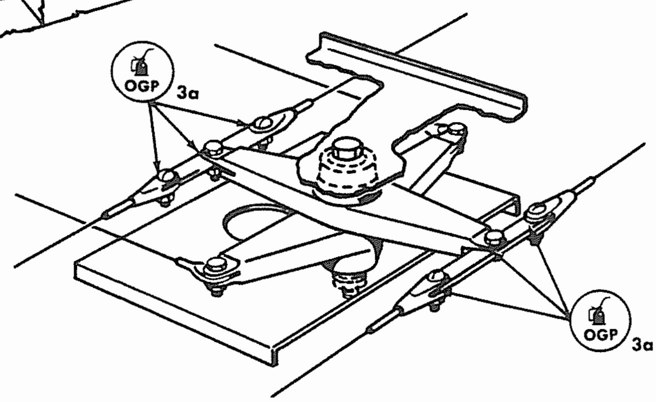




1. EMERGENCY EXITS



2. AUTOMATIC PILOT SERVO UNITS



3. AILERON HORN ASSEMBLY

FREQUENCY SYMBOLS

INTERMEDIATE MAJOR

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
OGP	MIL-L-7870	Oil — General Purpose
FGS	None	Graphite — Stick

ITEMS TO BE LUBRICATED AND REMARKS

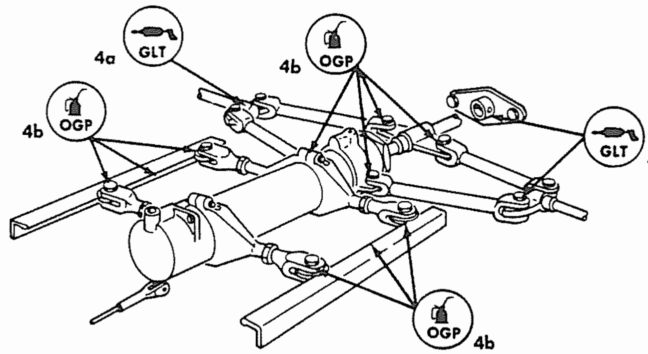
1. **EMERGENCY EXITS:**
   a. Piano hinge, latch operating mechanism turning points — oil lightly, wipe off excess. Do not allow oil to come into contact with finished surfaces.
   b. Latch bolts sliding surface — apply light film graphite stick.
2. **AUTOMATIC PILOT SERVO UNITS (THREE):**
   a. Link actuating levers turning points — oil lightly.
3. **AILERON HORN ASSEMBLY:**
   a. Cable connector links (two) — oil turning points lightly.

(Continued on next page)

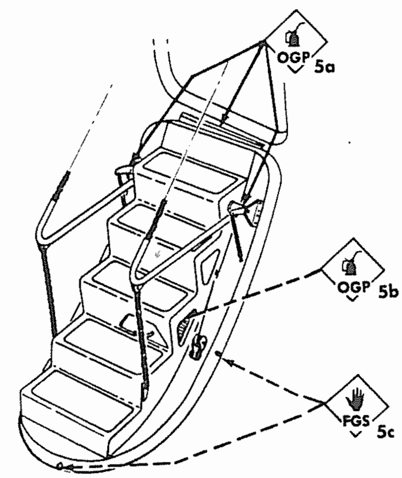
1,154

Figure 1-42 (Sheet 7 of 12 Sheets). Lubrication Chart

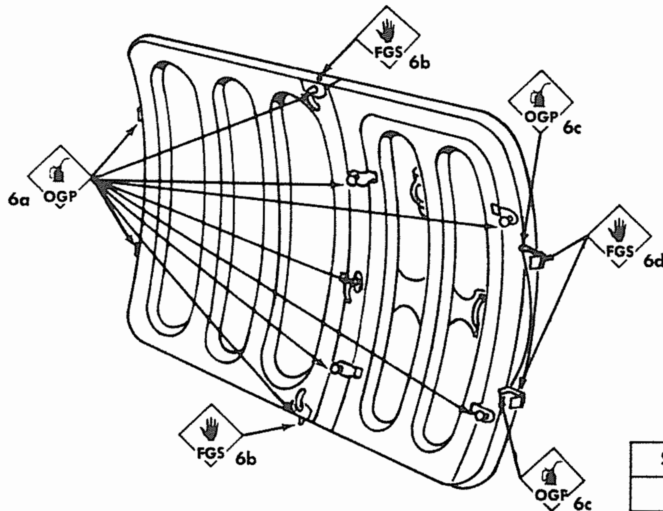
CABIN



4 — WING FLAP ACTUATING CYLINDER ASSEMBLY



* 5 — DOOR ASSEMBLY MAIN CABIN



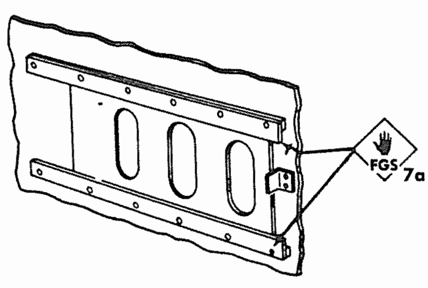
† 6 — DOOR ASSEMBLY MAIN CARGO FORWARD AND AFT SECTIONS

FREQUENCY SYMBOLS

INTERMEDIATE MAJOR

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
GLT	MIL-G-3278	Grease — Low Temperature
OGP	MIL-L-7870	Oil — General Purpose
FGS	None	Graphite — Stick



* 7 — CABIN VENT OUTLET

ITEMS TO BE LUBRICATED AND REMARKS

(Continued from preceding page)

4. **WING FLAP ACTUATING CYLINDER ASSEMBLY:**
   a. Support assemblies (three) — apply to grease fittings.
   b. Guides — operating rods and roller turning points — oil lightly.

* 5. **DOOR ASSEMBLY, MAIN CABIN:**
   a. Hinge latch and door jamb rollers and hand rail turning points — oil lightly, remove excess oil.
   b. Latch operating mechanism turning points—remove side cover plate and oil lightly.
   c. Latch bolts sliding surface — apply light film of graphite stick.

† 6. **DOOR ASSEMBLY, CARGO, FORWARD, AND AFT SECTIONS:**
   a. Hinges, locks, and latch turning points — oil lightly.
   b. Latch bolts — apply light film of graphite stick.
   c. Emergency release mechanism — oil turning points.
   d. Emergency release pins — apply light film of graphite stick.

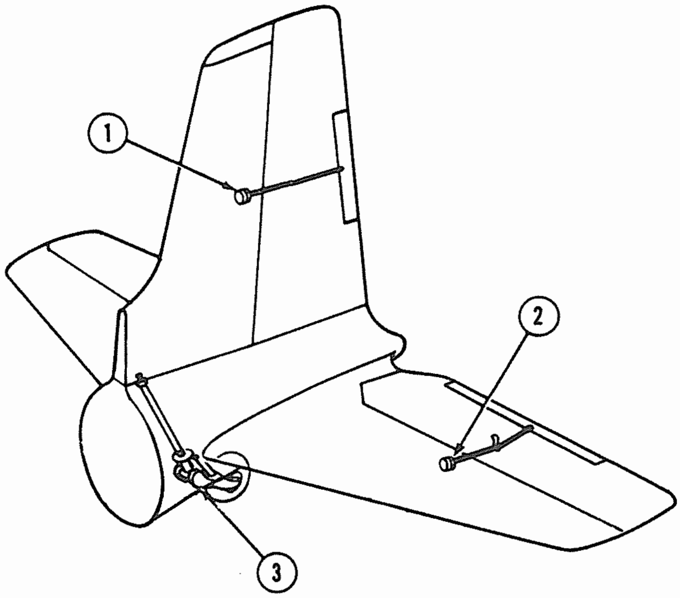
* 7. **CABIN VENT OUTLET:**
   a. Draw door open and apply a light film of graphite stick to grooves.

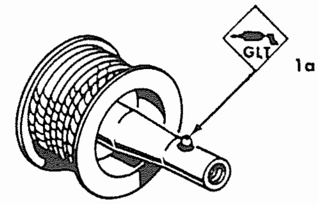
*Aircraft A through D
†Aircraft 1 through 96

1,155

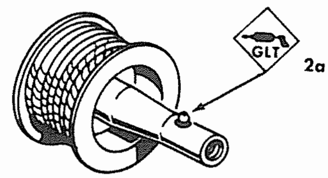
Figure 1-42 (Sheet 8 of 12 Sheets). Lubrication Chart

EMPENNAGE

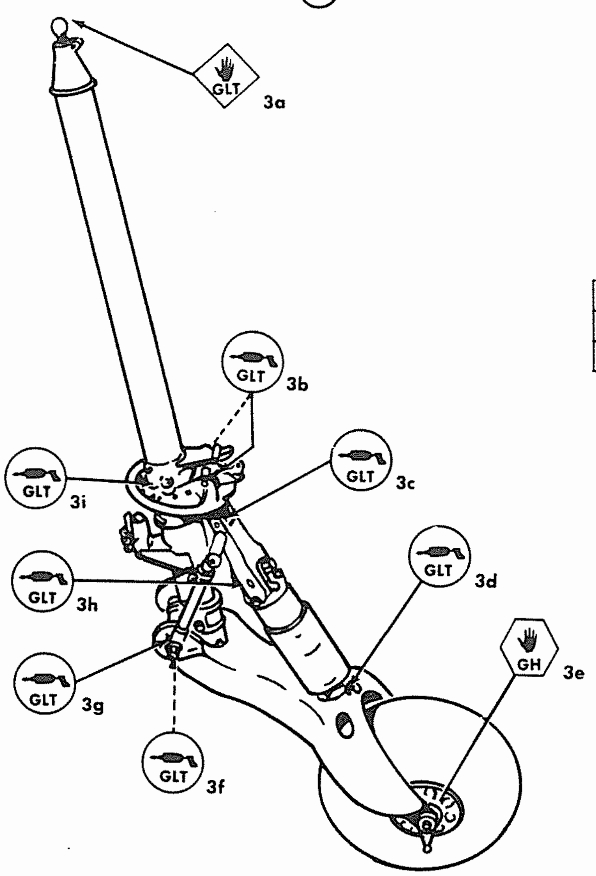




1 — RUDDER TRIM TAB ACTUATOR



2 — ELEVATOR TRIM TAB ACTUATOR



3 — TAIL WHEEL ASSEMBLY

FREQUENCY SYMBOLS

INTERMEDIATE — MAJOR — SPECIAL

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
GLT	MIL-G-3278	Grease — Low Temperature
GH	MIL-L-3545	— High Temperature

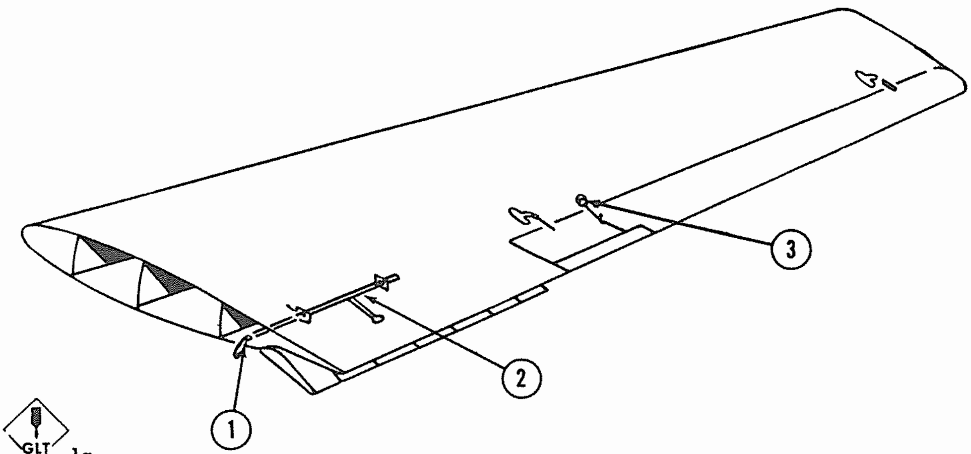
ITEMS TO BE LUBRICATED AND REMARKS

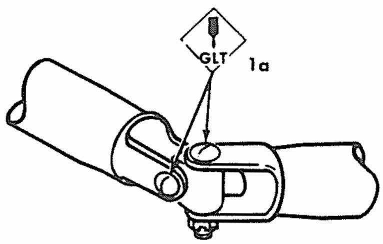
1. **RUDDER TRIM TAB ACTUATOR:**
   a. Tab control rod acme thread — apply to (one) grease fitting.
2. **ELEVATOR TRIM TAB ACTUATOR (ONE LEFT AND ONE RIGHT):**
   a. Tab control rod acme thread — apply to (one) grease fitting.
3. **TAIL WHEEL ASSEMBLY:**
   a. Spindle upper ball joint — remove lock nut, raise aircraft to expose ball joint, clean and repack.
   b. Lock assembly — apply to (two) grease fittings (one on each side).
   c. Spindle — link attach point — apply to (one) grease fitting (left side).
   d. Shock strut — fork attach point — apply to (one) grease fitting.
   e. Tail wheel bearing — clean old grease from axle and bearings, apply light coat of lubricant GH to axle and pack bearings.
   f. Spindle — fork attach points — apply to (two) grease fittings.
   g. Retract cylinder lower attach point — apply to (one) grease fitting.
   h. Link — shock strut attach point — apply to (one) grease fitting.
   i. Spindle center joint — apply to (one) grease fitting (top front).

1,156

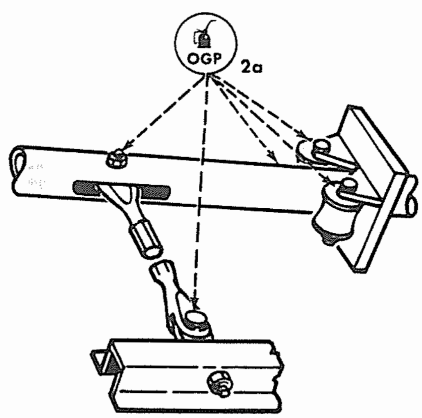
Figure 1-42 (Sheet 9 of 12 Sheets). Lubrication Chart

WING

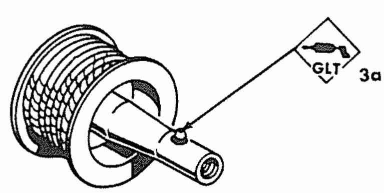




1 — WING FLAP UNIVERSAL JOINTS

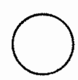


2 — WING FLAP ACTUATING MECHANISM

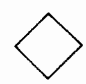


3 — AILERON TRIM TAB ACTUATOR

FREQUENCY SYMBOLS



INTERMEDIATE



MAJOR

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
GLT	MIL-G-3278	Grease — Low Temperature
OGP	MIL-L-7870	Oil — General Purpose

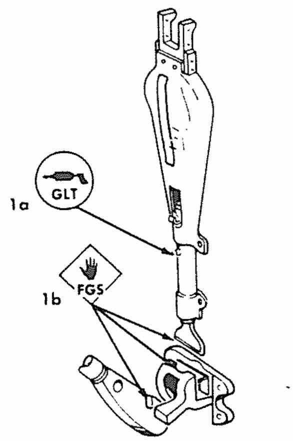
ITEMS TO BE LUBRICATED AND REMARKS

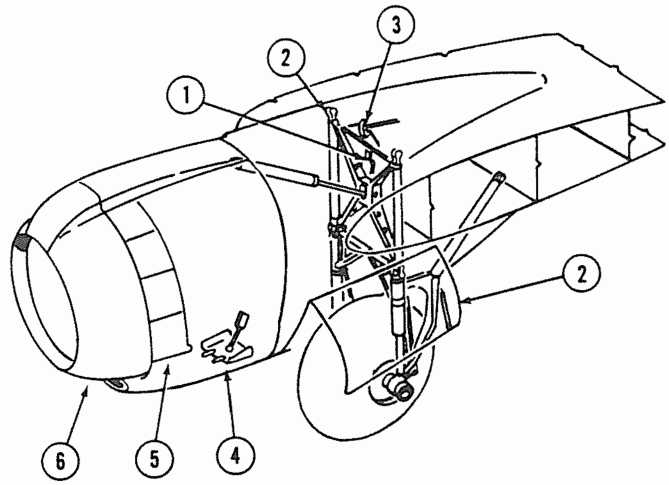
1. **WING FLAP UNIVERSAL JOINTS (ONE REQUIRED EACH WING):**
   a. Apply to turning points.
2. **WING FLAP ACTUATING MECHANISM:**
   a. Oil roller tube guide, turnbuckles and roller turning points.
3. **AILERON TRIM TAB CONTROL MECHANISM (RIGHT WING ONLY):**
   a. Tab control rod acme thread — apply to (one) grease fitting.

1,157

Figure 1-42 (Sheet 10 of 12 Sheets). Lubrication Chart

MAIN GEAR AND POWER PLANT SECTION





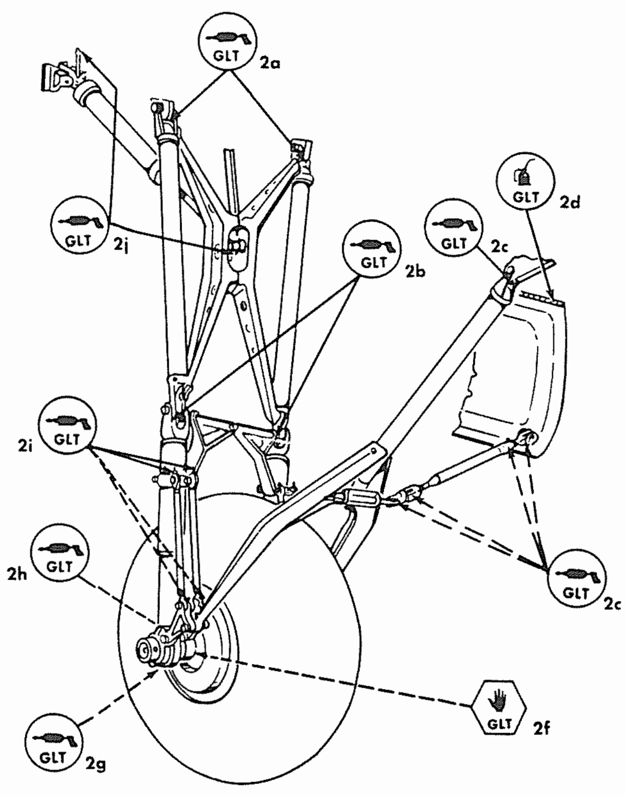
1 — MECHANICAL LATCH DOWN LOCK

FREQUENCY SYMBOLS

INTERMEDIATE MAJOR SPECIAL

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
GLT	MIL-G-3278	Grease — Low Temperature
OGP	MIL-L-7870	Oil — General Purpose
FGS	None	Graphite — Stick



2 — MAIN GEAR — DOORS AND DOOR OPERATING LINKAGE

ITEMS TO BE LUBRICATED AND REMARKS

1. **MECHANICAL LATCH DOWN LOCK (ONE PER GEAR, TWO GEARS)**
   a. Cylinder assembly — apply to (one) grease fitting.
   b. Latch face and actuating piston hook — apply light film of graphite stick.
2. **MAIN GEAR (TWO) MAIN GEAR DOORS (FOUR) AND DOOR OPERATING LINKAGE:**
   a. Truss upper attach points — apply to two grease fittings (front side).
   b. Truss lower attach points — apply to (two) grease fittings (one front, one rear inboard side).
   c. Rear brace strut upper attach point apply to (one) grease fitting.
   d. Main gear door (two per nacelle) — oil piano hinge.
   e. Main gear door operating linkage, operating rod and adjusting screw — apply to grease fittings (four fittings per link).
   f. Wheel bearings — clean old grease from axle and bearings, apply light coat of lubricant GLT to axle and pack bearings.
   g. Rear brace strut lower attach points — apply to (two) grease fittings (one each side).
   h. Link assembly (two) link — shock strut attach point — apply to (one) grease fitting each link.
   i. Link — arm assembly (four arms) — apply to (one) grease fitting at upper and lower attach point of each assembly.
   j. Retract cylinder — apply to upper attach point (one) grease fitting (through cover plate on top of nacelle) and to lower attach point to truss (one) grease fitting.

(Continued on next page)

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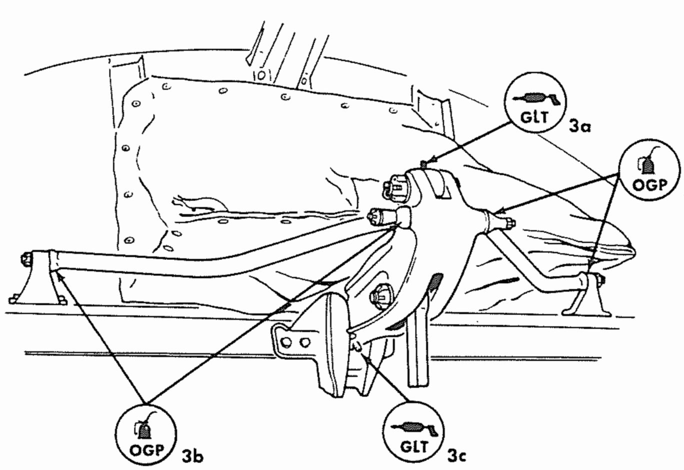
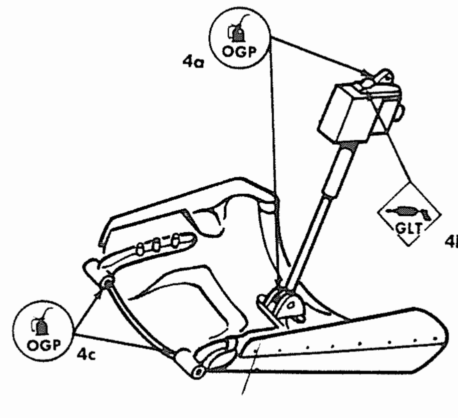
Figure 1-42 (Sheet 11 of 12 Sheets). Lubrication Chart

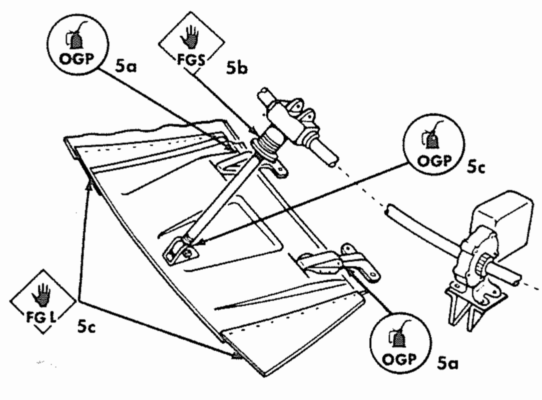
MAIN GEAR AND POWER PLANT SECTION



3 — BUNGEE ARM ASSEMBLY



4 — OIL COOLER DOOR



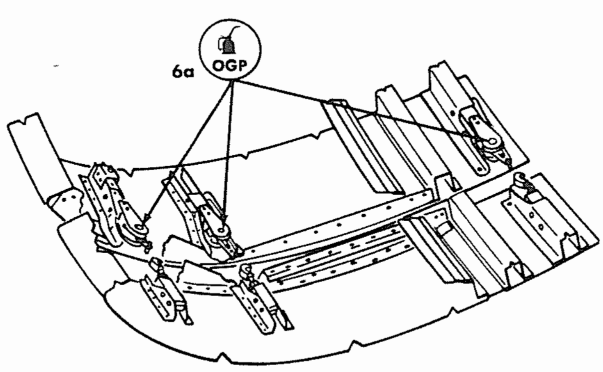
5 — COWL FLAP ASSEMBLY

FREQUENCY SYMBOLS

INTERMEDIATE   MAJOR

TABLE OF LUBRICANTS

Symbol	Specification	Type of Lubricant
GLT	MIL-G-3278	Grease — Low Temperature
OGP	MIL-L-7870	Oil — General Purpose
FGS	None	Graphite — Stick
FGL	None	Graphite — Lacquer Base



6 — ANTI-DRAG RING

ITEMS TO BE LUBRICATED AND REMARKS

(Continued from preceding page)

3. **BUNGEE ARM ASSEMBLY (ONE EACH NACELLE):**
   a. Toggle — bungee yoke attach point — apply to (one) grease fitting.
   b. Toggle — brace rods (two) — oil both ends of each rod.
   c. Toggle — spar bracket attach point — apply to (one) grease fitting.
4. **OIL COOLER DOOR:**
   a. Actuator upper and lower attach points — oil lightly.
   b. Actuator operating screw — apply to (one) grease fitting.
   c. Door hinge points — oil lightly.
5. **COWL FLAP ASSEMBLY:**
   a. Hinges (two) per flap (eight flaps per assembly) — oil lightly.
   b. Jack shaft dust tube (eight per assembly) — apply light film graphite stick to rubbing surface with flaps open.
   c. Jack shaft — flap attach point (eight per assembly) — oil lightly.
   d. Flap rub strips, previously treated surfaces — clean rub points with P-S-661 solvent (Stoddard-type), wipe dry and apply Lubetex No. 10.
6. **ANTI-DRAG RING:**
   a. Latches (three per cowling) — oil turning points lightly.

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Figure 1-42 (Sheet 12 of 12 Sheets). Lubrication Chart

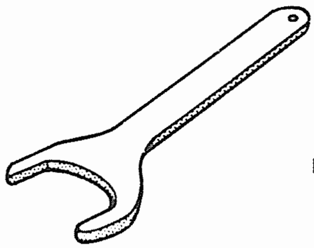
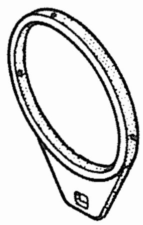
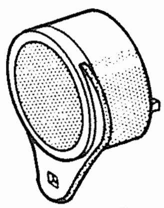
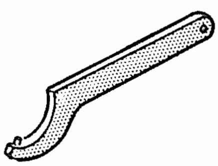
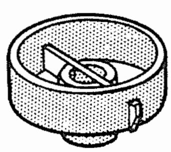
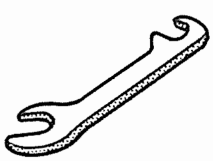
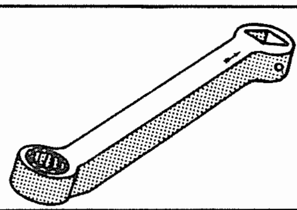
TOOLS	DESCRIPTION	APPLICATION
 K-20002	OPEN END WING FLAP STRUT WRENCH	Used to remove or tighten wing flap actuating cylinder gland nut.
 K-20102	MAIN ALIGHTING GEAR AXLE NUT SPANNER WRENCH	Use to tighten or torque main alighting gear axle nut.
K-20105	ALIGHTING GEAR RETRACT CYLINDER RETAINER SPANNER WRENCH	Used to tighten and torque the 2395117 retainer. May also be used to tighten and torque the 2395115 nut.
 K-20501	TAIL WHEEL SHOCK STRUT SPANNER WRENCH	Used to tighten and torque tail wheel shock strut packing gland nut.
 K-20801	TAIL WHEEL SPINDLE SPANNER WRENCH	Used to adjust 2371894 tail wheel spindle ball.
 K-20901	TAIL WHEEL SPINDLE SPANNER WRENCH	Used to remove or tighten tail wheel spindle ball socket retainer nut.
 K-23307	WRENCH-OPEN END PROPELLER GOVERNOR OPEN-END WRENCH	Used to remove or tighten propeller governor mounting nuts.
 K-39352	TORQUE WRENCH ADAPTER	Used to tighten and torque the EB-080 aileron hinge nuts.

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Figure 1-43 (Sheet 1 of 2 Sheets). Special Tools

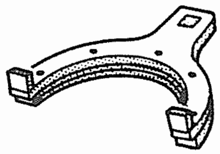
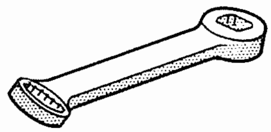
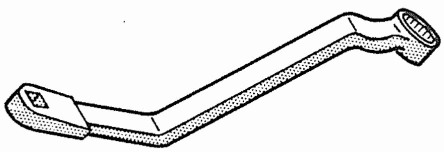
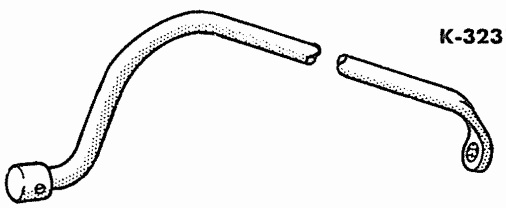
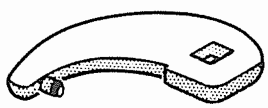
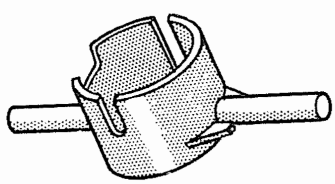
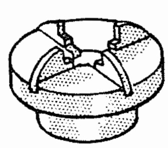
TOOLS		DESCRIPTION	APPLICATION
	K-26414	MAIN GEAR OLEO PACKING NUT SPANNER WRENCH	Used to remove or tighten main gear oleo packing nut.
	K-27703	TAIL WHEEL SHOCK STRUT PACKING NUT SPANNER WRENCH	Used to remove or tighten tail wheel shock strut packing nut.
	K-32313	TORQUE WRENCH ADAPTER	Used to tighten and torque the two center 9/16 inch mounting nuts on the forward side of the carburetor.
	K-32314	TORQUE WRENCH ADAPTER	Used to tighten and torque the two center 9/16 inch mounting nuts on the aft side of the carburetor.
	K-32316	GENERATOR INSTALLATION OFF-SET WRENCH	Used to remove or tighten generator mounting nuts.
	K-33707	TAIL WHEEL RETRACT CYLINDER END NUT SPANNER WRENCH	Used to tighten and torque the tail wheel retract cylinder and 2369430 nut.
	K-37001	WING FLAP STRUT BOX WRENCH	Used to remove or tighten wing flap cylinder cap nut.
	K-70401	ALIGHTING GEAR SPOT BRAKE SPANNER WRENCH	Used to remove or tighten the brake cap assembly.

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Figure 1-43 (Sheet 2 of 2 Sheets). Special Tools

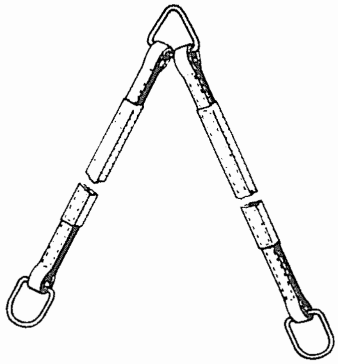
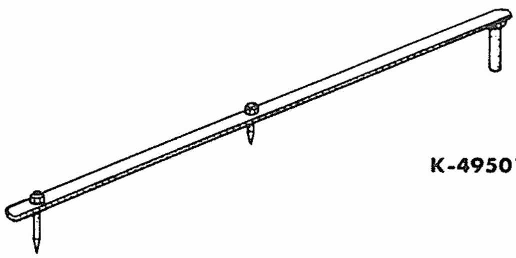
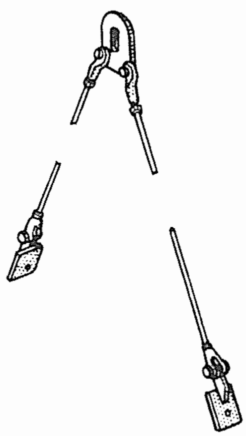
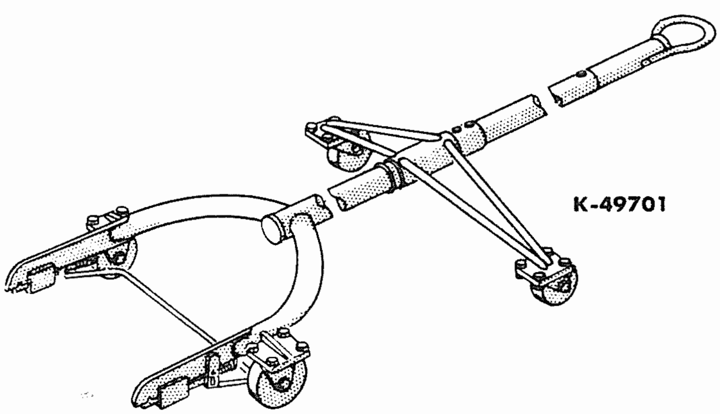
TOOLS	DESCRIPTION	APPLICATION
 K-14801	PROPELLER SLING	Used to hoist propeller.
 K-49501	ALIGNMENT TAIL WHEEL RETRACT FIXTURE	Used to check over-center condition of tail gear linkage in the extended position.
 K-49601	TAIL SLING	Used to hoist the aft end of the aircraft.
 K-49701	TAIL WHEEL TOW BAR	Used to tow the aircraft during ground handling operations.

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Figure 1-44 (Sheet 1 of 3 Sheets). Ground Handling Equipment

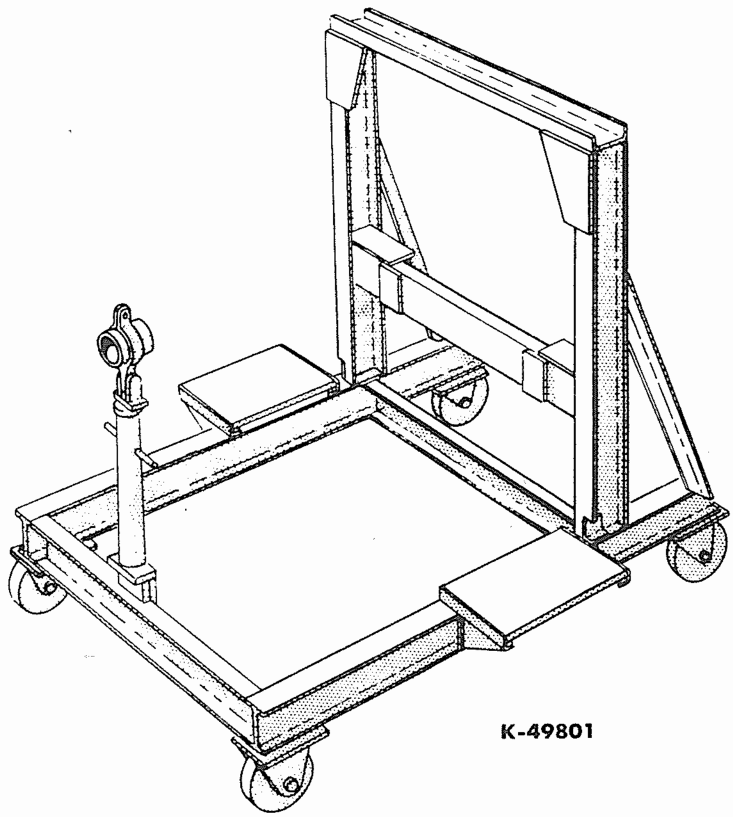
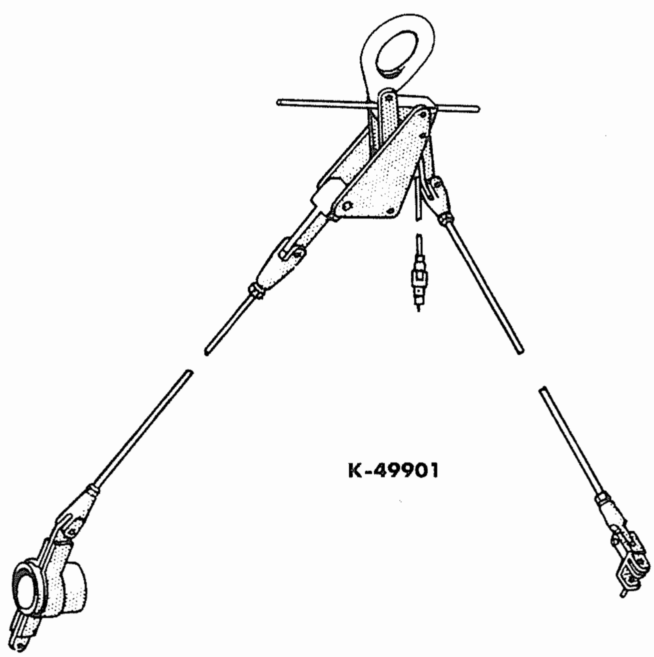
TOOLS	DESCRIPTION	APPLICATION
 K-49801	ENGINE DOLLY	Used as a means of conveyance for the power plant assembly during engine change.
 K-49901	POWER PLANT SLING	Used to hoist the power plant assembly during removal or installation of the engine.

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Figure 1-44 (Sheet 2 of 3 Sheets). Ground Handling Equipment

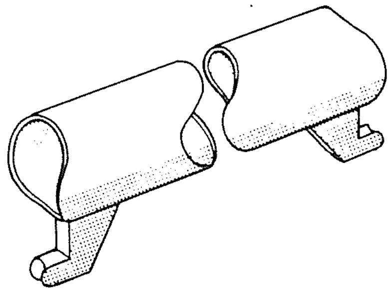
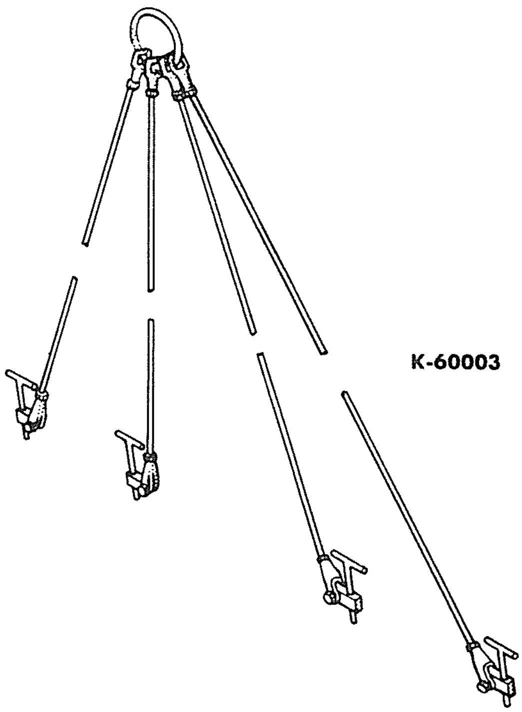
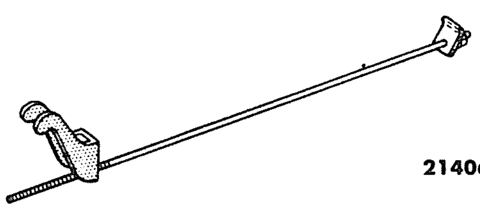
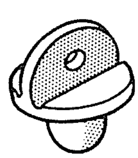
TOOLS	DESCRIPTION	APPLICATION
 K-51902	BUNGEE SPREADER BAR	Used to spread bungee cords during installation or removal of bungee assembly
 K-60003	OUTER WING SLING	Used to hoist wing during installation or removal of wing
 2140682	MAIN GEAR OLEO LINK ASSEMBLY	Used to restrict the oleo extension when aircraft is supported on jacks for wheel or tire change CAUTION Release air pressure before jacking aircraft.
 K-60012	TAIL JACK PAD	Used as a means of jacking the tail section of the aircraft during service or maintenance of the tail wheel gear

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Figure 1-44 (Sheet 3 of 3 Sheets). Ground Handling Equipment

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z AIRCRAFT

SECTION II AIRFRAME GROUP

THIS SECTION SUPERSEDES SECTION II OF AN 01-40NK-2
DATED 15 MAY 1952 REVISED 1 NOVEMBER 1952

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AND THE CHIEF OF THE BUREAU OF AERONAUTICS

15 April 1953

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SECTION II

AIRFRAME GROUP

TABLE OF CONTENTS

SECTION II

AIRFRAME GROUP

2-1. AIRFRAME GROUP.

2-2. DESCRIPTION. The airframe group consists of the wings, empennage, surface controls, fuselage, and alighting gear.

2-3. CLEANING AIRFRAME GROUP. Both the exterior and interior of the airframe should be cleaned periodically. Recommended methods and materials for maintaining a clean, sanitary, and odor-free interior are discussed in detail in paragraphs 2-243 through 2-248.

2-4. CLEANING AIRCRAFT EXTERIOR. Detailed information and instructions regarding the cleaning and polishing materials and methods for cleaning the aircraft exterior are described in the following paragraphs. Cleaning the exterior of the aircraft involves two general procedures: washing off dust, dirt, and other deposits that tend to promote corrosion; and the removal of corrosion products by the use of phosphoric acid-alcohol solution, followed by a chromic acid treatment, or by the use of abrasives or polishes.

2-5. The frequency of cleaning the aircraft depends partly on the appearance and partly on the degree of corrosion present. Exhaust gas deposits on wing and flap areas cause severe corrosion. In order to prevent this condition, frequent cleaning of the affected areas is necessary. If frequent cleaning is performed, only light deposits will form, which may be removed by a light-duty cleaning agent. If relatively heavy deposits are allowed to form, removal may require the use of paint strippers. If deposits have been permitted to accumulate to the point where corrosion is present, mechanical or chemical removal of the corrosion products may be required, to be followed by chromodizing, repriming, and repainting of the affected area.

2-6. CLEANING MATERIALS FOR AIRFRAME SURFACES.

Specification, Material and Applicable Stock Numbers	*Specific Operation*	*Applicable Procedures*	*Remarks*
a. C-120 CLEANING COMPOUND (for airframe surfaces). R51C1569-100 R51C1569-125	General washing of airframe surfaces.	Use the compound in a water solution containing not more than four ounces (avoir.) per gallon, except for heavy degreasing. The compound may be used in sea water if no fresh water is available and may be used as a hand wash and cleaner. Do not use as a tank solution. Wash the airframe in the shade when possible, cooling the surface with cold water. Nonspecular finishes should not be cleaned more than necessary and should never be scrubbed with coarse brushes or coarse rags. A soft sponge or cheese cloth used with a minimum of rubbing is advisable. Remove oil or exhaust stains on the surface with a solvent such as kerosene or Petroleum Solvent (P-S-661A). The surface should be rinsed immediately after cleaning to prevent the compound from drying hard on the surface.	This cleaner is a general-purpose compound and may be used on fabrics, leather, glass, ceramics, and transparent plastics. This material is also suitable for use as the wetting and suspending agent for wet blast process grit.

CLEANING MATERIAL FOR AIRFRAME SURFACES *(Continued)*

Specification, Material and Applicable Stock Numbers	*Specific Operation*	*Applicable Procedures*	*Remarks*
b. P-S-600A LOW-TITER SOAP (for low-temperature washing). G51S-1715-90	General washing of airframe surfaces.	Apply in water solution in concentration as necessary, using the same procedure as outlined above. For use in fresh water only.	This soap is intended for use in cleaning finished airframe surfaces, articles of equipment, leather, fabric, and other materials which require a neutral soap.
c. MIL-C-5410A NONFLAMMABLE CLEANER AND BRIGHTENER COMPOUND. Type I Type II (concentrate for use on aluminum surfaces). R51C1564-200	Cleaning and brightening of aluminum surfaces.	Type I is used as packaged. Type II must be diluted with an equal volume of water before using. Apply with soft brush, rags, or sponges. Begin application on the top surfaces. Apply with circular motion, brushing enough to loosen the surface film. Allow the compound to remain on the surface a maximum of 15 minutes, then rinse with a stream of water.	Do not use on or rinse over magnesium surfaces. When practicable, cleaner and brightener applications should be done outdoors in a shaded area or in a well ventilated room. Rubber or synthetic rubber gloves and goggle-type eye-glasses should be worn when applying the compound. In case the compound is splashed onto the skin, it should be immediately washed off with water or with a dilute solution of sodium bicarbonate, followed by application of glycerin or petrolatum jelly. If compound or mist gets into the eye, wash with copious quantities of cold water for 10 minutes.
d. TT-N-95 ALIPHATIC PETROLEUM NAPHTHA (supersedes AN-N-3). R52N450	Hand cleaning of plastics.	To remove grease and oil, wipe with a clean soft cloth soaked in aliphatic naphtha.	**CAUTION** Aliphatic Naphtha (TT-N-95) is not be confused with Aromatic Naphtha (TT-N-97), which is harmful to plastics. Do not use gasoline, alcohol, benzene, xylene, acetone, carbon tetrachloride, fire extinguisher or de-icing fluids, lacquer thinner or window cleaning sprays or plastics as they may soften the plastic or cause crazing. Precautions regarding the use of flammable liquids must be observed. Do not rub plastic with a dry cloth as it may cause scratches or build up a static charge which attracts dust. Pat or blot with a damp chamois.
e. P-S-661A DRY-CLEANING SOLVENT R51C1326-67 R51C1326-75	Hand or machine cleaning of fabrics, etc.	Use by dipping, scouring by hand, or in a dry-cleaning machine for clothing, fabrics, etc. For heavy duty cleaning of airframe surfaces apply with rags, brush, or sponge and follow with a cleaner listed above.	An all-purpose cleaning solvent to be used by itself or in conjunction with other cleaners. It is preferable to kerosene. Use with adequate ventilation to avoid prolonged breathing of vapors. Keep away from open flame, flash point is above 37.8°C (100°F).
f. MIL-P-6888 ALUMINUM POLISH (supersedes AN-P-88) R51P1023 R51P1025	Hand polishing of aluminum surfaces.	Use in accordance with manufacturer's instructions. Surfaces should be cleaned to remove soil or oil prior to application of aluminum polish.	The abrasive quality enables the polish to remove tarnish and to produce a high, lasting polish on unpainted, aluminum-clad surfaces. It is not to be used on anodized surfaces. Do not use mechanical buffers. Use with adequate ventilation to avoid excessive breathing of the vapor. Avoid excessive contact with the skin.

CLEANING MATERIAL FOR AIRFRAME SURFACES *(Continued)*

Specification, Material and Applicable Stock Numbers	*Specific Operation*	*Applicable Procedures*	*Remarks*
g. MIL-C-5547A POLISHING COMPOUND (for acrylic plastic) (supersedes AN-C-154) R51C1606-25	Polishing of acrylic plastic.	Use in accordance with manufacturer's instructions. A small amount of the compound is generally applied to a soft flannel pad and rubbed over the plastic surface to be polished. Allow to dry. The remaining powder is removed by wiping or brushing lightly.	Remove dirt with a flow of water before using the compound. Use clean cloths.
h. C-71A WATERPROOF SELF-POLISHING LIQUID WAX COMPOUND (airframe) R52C3126	Waxing of miscellaneous airframe materials and finishes.	Wring out cheesecloth pad saturated with wax (to doped fabric only) within a temperature range of 10° to 37.8°C (50° to 100°F). Standard spray equipment may also be used. Avoid applying under a hot sun. Rubbing or polishing is not required. To remove, for purposes of patching doped fabric or additional coats of varnish or enamel, use gasoline on cheesecloth and 3F or 4F pumice stone.	Do not use on non-specular finishes. Do not apply to new finish until thoroughly dry, which may require 10 days. A coat of this wax may be applied as a final operation in cleaning or polishing acrylic plastics.
i. 52-R-17 (Aer) RUBBING COMPOUND R51C1606-50	Hand and machine smoothing of lacquered surfaces.	Use in accordance with manufacturer's instructions. Allow sufficient time for the finish to dry before rubbing. Hand rubbing should always be performed in locations difficult to rub with a disc. Rubbing should always be in the direction of airstream. Smoothing of glossy lacquer may be done by hand rubbing or disc buff, but smoothing of glossy enamel shall be done by hand rubbing only. Polishing wax may be applied to the smooth surface and polished with a mechanical polisher.	CAUTION Heavy rubbing which might rub through the finish, or materially reduce the coat thickness in one spot, should not be performed on rivet heads, edges, etc.

2-7. FINISHES. Most of the alloys used in the construction of the aircraft require protection from corrosion. Corrosion takes place quickly, particularly in damp coastal climates; under such climatic conditions, all scratches, abrasions, or marred finish surfaces must be promptly cleaned and must then receive an application of a protective coating as soon as possible, to avoid corrosion and extensive repairs. Do not apply paint to working surfaces, screw threads, oil holes, rubber, synthetic rubber, or hot and cold aluminum air ducts. Do not apply paint to fittings in such a way that it would cause the bearings to seize.

2-8. FINISH MATERIALS FOR AIRFRAME GROUP COMPONENTS.

Materials	*Specification*	*Application*
a. Primer:		
Zinc chromate	MIL-P-6889A (supersedes P-27, AN-TT-P-656, AN-P-656, AN-P-656a, and MIL-P-6889)	Airframe metal surfaces.
b. Dopes:		
Cellulose acetate butyrate:		
Fungicidal first coat	D-34	Fabric-covered surfaces.
Clear	MIL-D-5549 (supersedes AN-D-1)	
Glossy pigmented	MIL-D-5551 (supersedes AN-D-2)	
Camouflage pigmented	MIL-D-5550 (supersedes AN-D-3)	
Cellulose nitrate:		
Clear	MIL-D-5553 (supersedes AN-TT-D-514)	
Clear aluminizing	MIL-D-5552 (supersedes AN-TT-D-551)	
Glossy pigmented	MIL-D-5554 (supersedes AN-TT-D-554)	

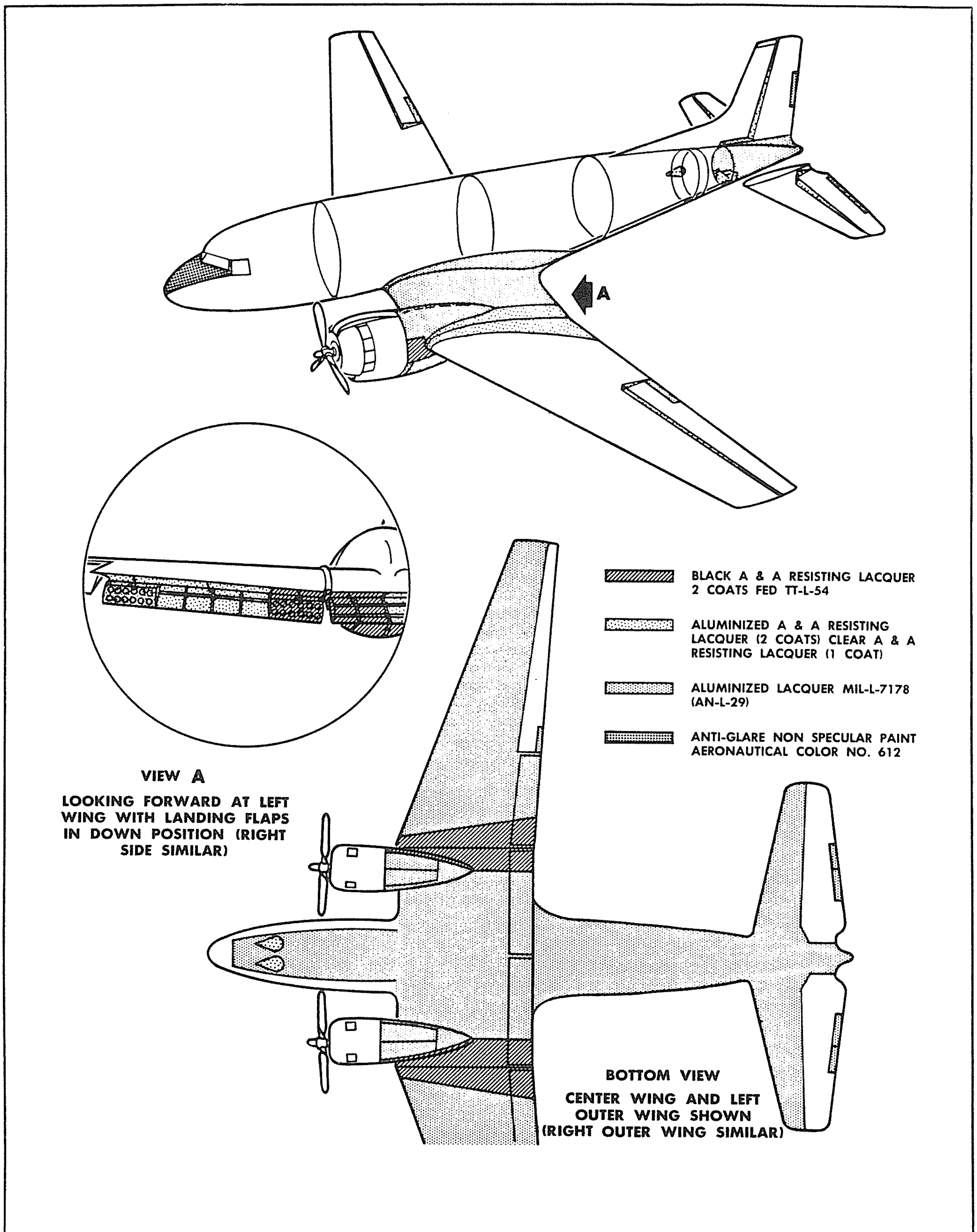


Figure 2-1. Protective Paint Requirements (Aircraft 1 through 96) — Fuselage Exterior

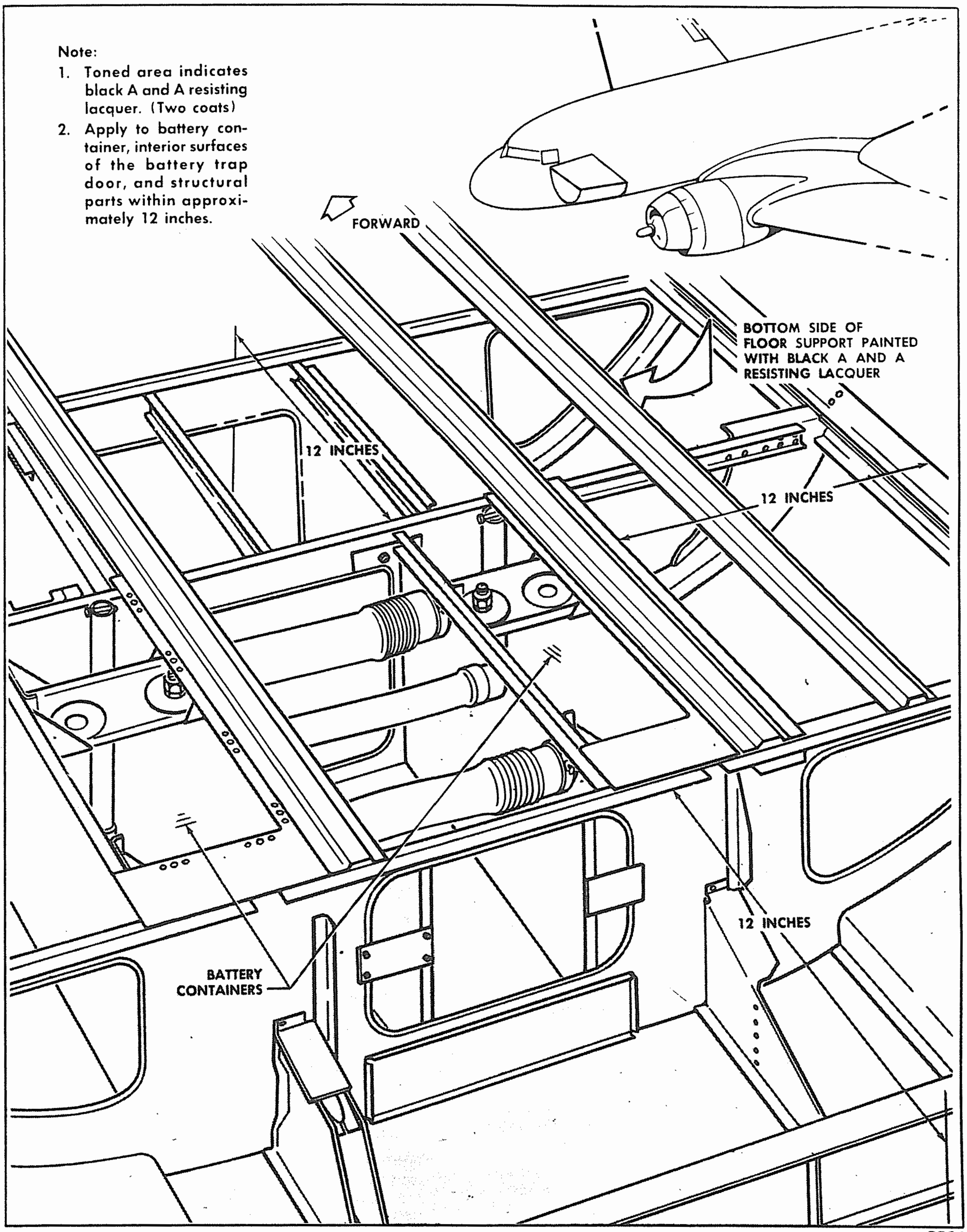


Figure 2-2. Protective Paint Requirements (Aircraft 1 through 96) — Fuselage Interior

Materials	*Specification*	*Application*
b. *(Continued)*		
Camouflage pigmented	MIL-D-5555 (supersedes AN-D-8)	
c. Lacquers:		
Glossy	MIL-L (general specification number)	Primed metal, wood.
Camouflage	MIL-L-6805 (supersedes AN-L-21)	Insignia and markings on doped fabric in emergencies only.
Clear	MIL-L-6806 (supersedes AN-L-37)	Aluminum-clad aluminum-alloy surfaces.
Hydraulic fluid resistant	MIL-L-7146 (Aer)	Primed metal.
Aromatic fuel resistant	MIL-L-6047 (supersedes M-769)	Steel.
Acid resistant	TT-L-54	Primed metal.
d. Enamels:		
Glossy	MIL-E (general specification number) (supersedes AN-E-3)	Primed metal.
Camouflage	MIL-E-5556 (supersedes AN-E-7)	Wood, metal.
Black heat-resisting	MIL-E-5557 (supersedes AN-TT-E-501)	Prohibited on fabric.
Type I, air-drying gloss black		
Type II, baking gloss black		
Type III, air-drying instrument black		
Type IV, baking instrument black		
Black wrinkle finish	MIL-E-5558 (supersedes AN-E-51)	Metal, primed metal.
e. Varnishes:		
Spar	MIL-V-6894 (supersedes AN-TT-V-116)	Wood, metal, metal to wood contacts.
Phenol formaldehyde spar	MIL-V-6893 (supersedes AN-V-26)	Wood, metal, metal to wood contacts.
Decalcomania adhesive	MIL-V-6093	Painted or unpainted surfaces.
f. Resin Coating	MIL-R-3042 (supersedes AN-C-148)	Metal airframe and engine parts.
g. Temporary Identification Paint	MIL-P-6884 (supersedes AN-P-85)	Any aircraft surface.
h. Bitumastic Paint	MIL-P-6883 (supersedes AN-P-31)	Plywood, metal, metal to wood contacts.
i. Walkway Materials, Non-Slip	MIL-C-5044 (supersedes 52C48 (Aer) and W-52)	Primed metal.
Type I, smooth		
Class I, brush		
Class II, spray		
Type II, rough		
Class I, brush		
Class II, spray		
Matting	W-51	Primed metal.
j. Predoped Fabrics:		
Cellulose acetate butyrate	MIL-C-5642 (supersedes AN-C-132)	Air frames, plywood.
Cellulose nitrate	MIL-C-5643 (supersedes AN-C-113)	Plywood.
k. Aluminum Pigment Powder and Paste	TT-A-468, Types I and II, Class A	Pigment for dopes.
l. Sanding Surfacer	MIL-S-974 (Aer) (supersedes 40S6 (Aer))	Primed aircraft surfaces.
m. Rubbing Compound	52R17 (Aer)	Lacquered surfaces.
Type I, hand use		
Type II, machine use		
Type III, hand or machine use		

2-9. SPECIFIC ITEM MAINTENANCE CHART.

Area	*Finish*	*Figure No.*
a. Exterior:		
Fuselage skin		
Station 538 (wash basin outlet)	1 spray coat primer (MIL-P-6889A) 2 spray coats lacquer (Aluminized) (FED TT-L-54)	Figure 2-1.

Area	*Finish*	*Figure No.*
a. *(Continued)*		
Station 24 (pilot relief tube outlet)	1 spray coat primer (MIL-P-6889A) 2 spray coats lacquer (Aluminized) (FED TT-L-54)	Figure 2-1.
Center wing Top	1 spray coat primer (MIL-P-6889A) 2 spray coats lacquer (Aluminized) (FED TT-L-54)	Figure 2-1.
Bottom	1 spray coat primer (MIL-P-6889A) 2 spray coats black lacquer (FED TT-L-54)	Figure 2-1.
b. Interior:		
Instrument panel	2 coats Instrument Black Lacquer (AN-L-29)	None.
Battery compartment	1 spray coat primer (MIL-P-6889A) 2 spray coats black lacquer (FED TT-L-54)	Figure 2-2.
Felt pads, covered	2 heavy spray coats or 1 dip coat of Blue Tinted Acetate Butyrate Dope	None.
Water tanks, wash and drinking	1 coat Sherwin-Williams Water Tank Coating No. 240, interior only; outside to match adjacent areas.	None.
Alcohol tanks	Anodic (AN-A-21)	None.

2-10. TREATMENTS FOR MAGNESIUM PARTS. Magnesium parts should be cleaned either with dry cleaning solvent (Specification P-S-661), kerosene, or MIL-M-7752 (Aer). Do not use carbon remover solutions or any preparations containing carbon tetrachloride, which corrodes magnesium. Parts having magnesium and aluminum combined integrally may be cleaned with a carbon remover, but should be thoroughly rinsed in the dry cleaning solvent (Specification P-S-661) referred to in paragraph 2-6.

2-11. FINISHING OF INTERIOR WOOD SURFACES.

a. Wood parts and surfaces must be thoroughly cleaned and dried prior to finishing. If sanding is required on veneered plywood surfaces, do not sand down more than five per cent of the exterior ply.

b. Wood surfaces, in the flight compartment and crew compartment of all aircraft and in the main cargo compartment of aircraft C through 96, are finished with interior green lacquer (TL-9960, AN-L-29, Color 611).

c. Wood trim, in the main cabin of aircraft A and B, is primavera and is finished with a light stain and a flat lacquer.

2-12. WING GROUP.

2-13. DESCRIPTION. The wing is an aluminum alloy structure consisting of a center section, two outer panels, and two tips. The wing center section, including two integral nacelles, is joined to the fuselage by eight vertical fittings, four on each side at the fuselage-center wing juncture. The wing outer panels are attached to the wing center section by closely spaced steel bolts and elastic stop nuts. A floating or false rib is installed between the wing center section and the wing outer panels. The wing tips are attached to the outer panels by bolts and nuts. Inspection doors provide access to wing tip attachment points. The movable surfaces of the wing group consist of a partial-span wing flap, two ailerons, and two aileron geared trim control tabs. See paragraph 2-43 for further information concerning the movable wing surfaces.

2-14. WING OUTER PANELS. The two wing outer panels are aluminum alloy structures containing three main spars, which are continuations of the three wing center section spars. Fittings are provided within the wing structure for the support and actuation of the ailerons and flaps. A de-icer shoe is installed on the leading edge of each of the outer wing panels.

2-15. REMOVAL OF WING OUTER PANELS. After making certain that the fuel tanks have been drained, proceed as follows:

a. Working through a small access door located between the center and rear spars inboard on the lower surface of the outer wing, disconnect the fuel pipe connection.

b. Remove the fairing strip that covers the wing attachment angles and bolts by detaching the hold-down clips and taking out the screws at the trailing edge fitting.

c. Detach the small gap cover installed on the under side of the trailing edge of the wing by removing the machine screws.

d. Working through the wing inboard access door, disconnect the aileron control cables (see paragraph 2-88). The aileron tab cables should also be removed at this point.

CAUTION

Before disconnecting the aileron tab cables, the tab drums, which are located in the control pedestal and in the wing outer panel, must be taped to prevent the cables from unwinding.

e. Disconnect the outer wing electrical wiring at the junction box located in the nacelle.

f. Lower the wing flaps and disconnect the flap control tube at the union outboard of the wing attaching point. Measure the distance between the adjusting nuts on the flap control tube ends with calipers before removing the nuts. This method will facilitate the re-adjustment of the flaps when the tube is being installed.

g. Working through the access door on the outboard side of the nacelle, disconnect the de-icer pipes at the wing joint by removing the clamps from the hose.

h. Remove the machine screws from the hoisting sling attach-points in the top surface of the outer wing panel, and attach the hoisting sling.

i. Remove the steel bolts that attach the outer wing panel to the wing center section. Remove the bolts from the under surface first.

j. Lift the outer wing panel clear of the wing center section, after making certain that all the pipes, cables, and wires have been disconnected.

k. Remove the floating or false rib.

2-16. INSTALLATION OF WING OUTER PANELS. *(See figure 2-3.)* To install the wing outer panels, proceed as follows:

a. Attach the outer wing panel sling to the hoisting points on the upper surface of the wing.

b. Make certain that the floating or false rib and the rubber bumpers on the spars are installed.

c. Install the attach-angle bolts at the top and bottom of the spars. These must be installed first.

d. Install the rest of the bolts in the wing attach angles and tighten gradually in sequence around the wing.

e. A torque wrench should be used to tighten the nuts to a value of 135 to 150 foot-pounds.

Note

Bolts must be absolutely free of grease and oil when installed, and cadmium-plated steel washers should be used.

f. Connect the de-icer pipes at the wing joints by installing hose clamps; access to the pipes is through the door on the outboard side of the nacelle.

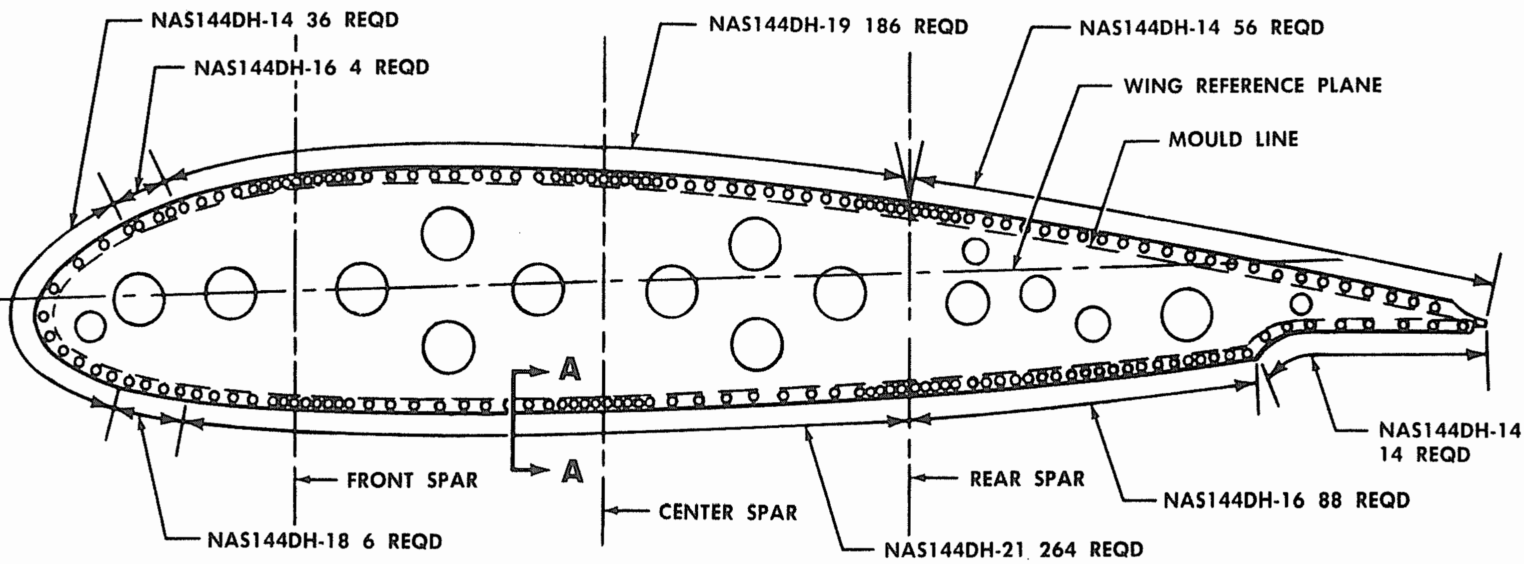
g. Connect the flap control tube at the union outboard of the wing attaching point.

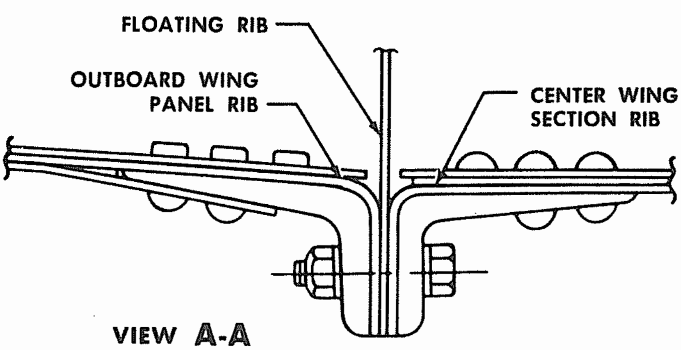
h. Connect the outer wing electrical wiring at the nacelle junction box.

i. Working through the wing access door, connect the aileron control cables. The aileron tab cables should also be connected at this point.

j. Attach the small gap cover on the under side of the trailing edge of the wing by installing the machine screws.

k. Install the fairing strip which covers the wing attachment angles and bolts by installing the screws at the trailing edge fitting and attaching the hold-down clips.





Notes:

1. Use one washer under bolt head and one washer under nut on all bolts except — use two washers under heads of bolts at leading edge and top side rear.
2. Install attach angle bolts in spars first, top and bottom. Then install bolts in wing attach angles, tightening them gradually around wing in sequence to torque value of 135 to 150 inch-pounds.

OUTER WING PANEL TO STUB WING ATTACHME

Figure 2-3. Installation of Wing Outer Panel

2-17. WING CENTER SECTION. *(See figure 2-4.)* The wing center section is an aluminum alloy structure containing three main spars and an auxiliary spar. The wing center section trailing edge is removable, being attached to the rear spar with bolts and elastic stop nuts. The nacelles, which are constructed as separate units, are attached to the wing center section by bolts and rivets. Incorporated in each nacelle structure, on the lower hat sections, are two rubber fittings against which the alighting gear axle rests when the alighting gear is retracted. These fittings provide a support to protect the nacelle structure if landing with the gear retracted becomes necessary. Alighting gear attachments are provided on the front spar of the wing center section, within the nacelle area, for the installation of the alighting gear. The wing center section also incorporates provisions for the installation of the fuel tanks.

2-18. REMOVAL OF WING CENTER SECTION. After making certain that the fuel tanks are drained, proceed as follows:

a. Remove both wing outer panels (see paragraph 2-15).

b. Remove both engines.

c. Hoist the entire aircraft by means of the nacelle hoists and the rear fuselage hoist (see paragraph 1-56).

d. Remove the alighting gear (see paragraphs 2-349 and 2-376).

e. Remove the fillets and neoprene strips installed between the wing and fuselage (see paragraph 2-24).

f. Attach the hoist sling to the hoist fitting at the auxiliary spar.

g. Remove the fuselage floor panels, which are located over the wing.

h. Remove the bolts that attach the two fuselage floor beams to the wing (stations 216 and 234).

i. Disconnect all pipes at the unions that are located at the intersection of the fuselage and wing tunnels.

j. Disconnect all surface control cables at the turnbuckles located under the companionway floor. Pull the empennage control cables out of the fuselage, and pull the aileron control cables out of the wing.

k. Disconnect the empennage de-icer pipes.

l. Disconnect the electrical plugs at the fuel booster pumps.

m. Remove the wing flap indicator control wires from the wing by removing the clevis located at the wing flap actuating cylinder. Pull the wire out of the center wing.

n. To disconnect the electrical wires at the junction of the center wing and the fuselage, it will be necessary to remove the plugs located in the leading edge of the inner wing. Then disconnect the two starter wires and the flexible conduit in the junction box located on the forward spar.

o. Disconnect the electrical wires leading from the fuselage to the inner wing.

p. Remove the screws that attach the wing trailing edge to the fuselage skin.

q. Remove the bolts that attach the fuselage longitudinal bulb angles in the trailing edge of the wing.

r. Remove all screws that attach the fuselage frames to the upper surface of the inner wing.

s. Remove the screws from the five intercostal brackets that tie the lower inner wing nose to the lower fuselage circumferential.

t. Remove the bolts that attach the lower fuselage surface to the center wing attach angles.

u. Remove the fuel vent pipe supporting clips on the right and left sides of the fuselage above the wing.

v. Remove the bolts from the fittings that attach the wing spars to the fuselage.

w. Check to make certain that no connections have been overlooked and that the wing is free.

x. Lift the fuselage away from the center wing section.

2-19. INSTALLATION OF WING CENTER SECTION. Reverse the removal procedure.

2-20. WING TIPS. *(See figure 2-5.)* The wing tips are aluminum alloy structures and are attached to the wing outer panels with machine screws. The wing tip structures include fittings for the attachment of the wing navigation lights.

2-21. REMOVAL OF WING TIPS. To remove the wing tips, proceed as follows:

a. Remove the length of the fairing that secures the de-icer shoe to the wing tip.

b. Disconnect the electrical wiring (see paragraph 2-15, step e).

c. Working through the inspection doors, remove the bolts that attach the wing tip section to the outer wing panel.

d. Pull the wing tip outward until it is free of the outer wing panel.

2-22. INSTALLATION OF WING TIPS. Reverse the removal procedure.

2-23. WING-TO-FUSELAGE FILLETS. The chief function of the wing-to-fuselage fillets is to reduce drag by providing a smooth airflow at the wing root.

2-24. REMOVAL OF WING-TO-FUSELAGE FILLETS. Remove the machine screws, and remove the fillets and neoprene sealing strips.

2-25. INSTALLATION OF WING-TO-FUSELAGE FILLETS. Reverse the removal procedure.

2-26. WING GROUP BOLT TORQUE VALUES. For information on the wing group bolt torque values, see paragraph 2-453.

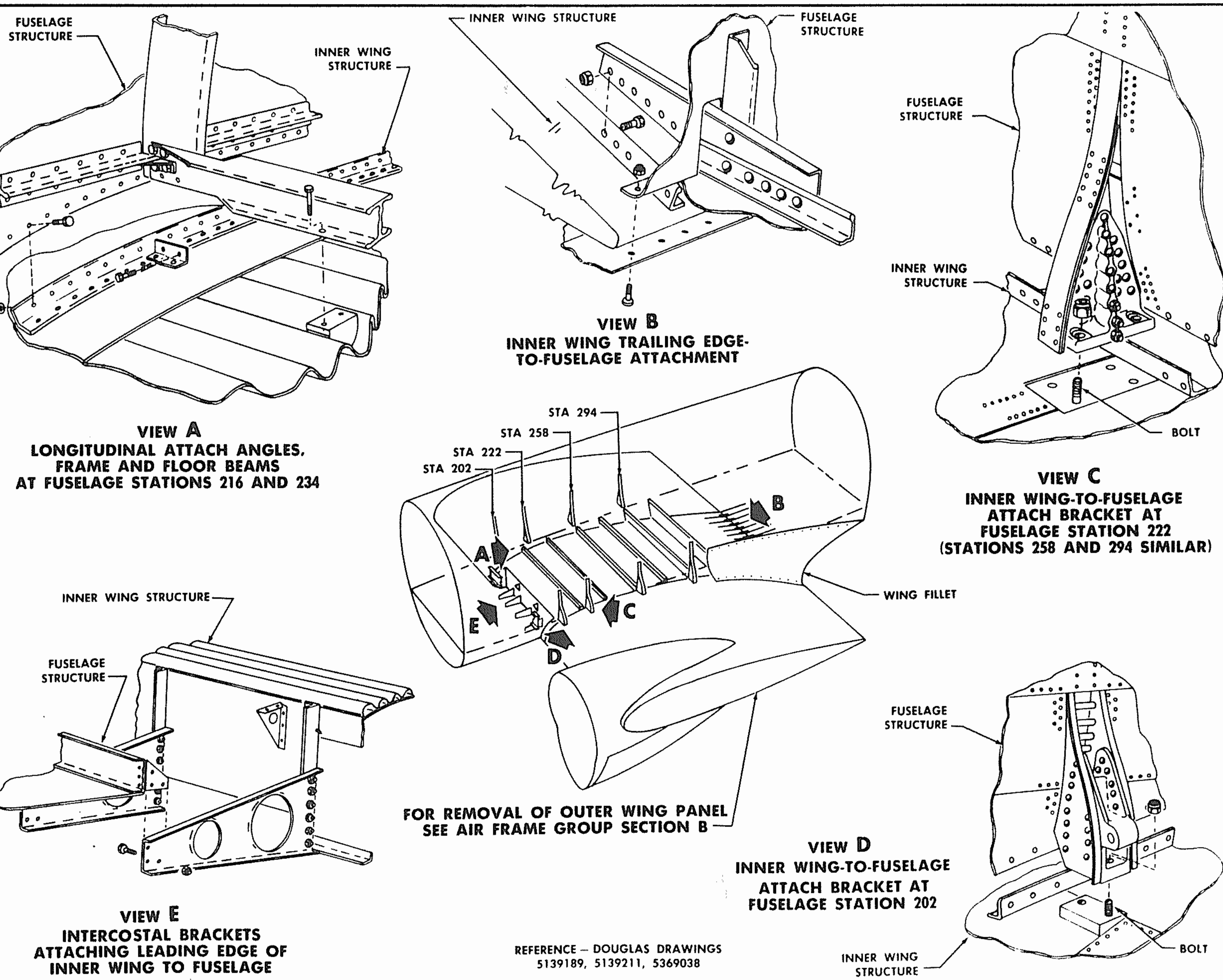


Figure 2-4 (Sheet 1 of 2 Sheets). Wing Center Section Installation

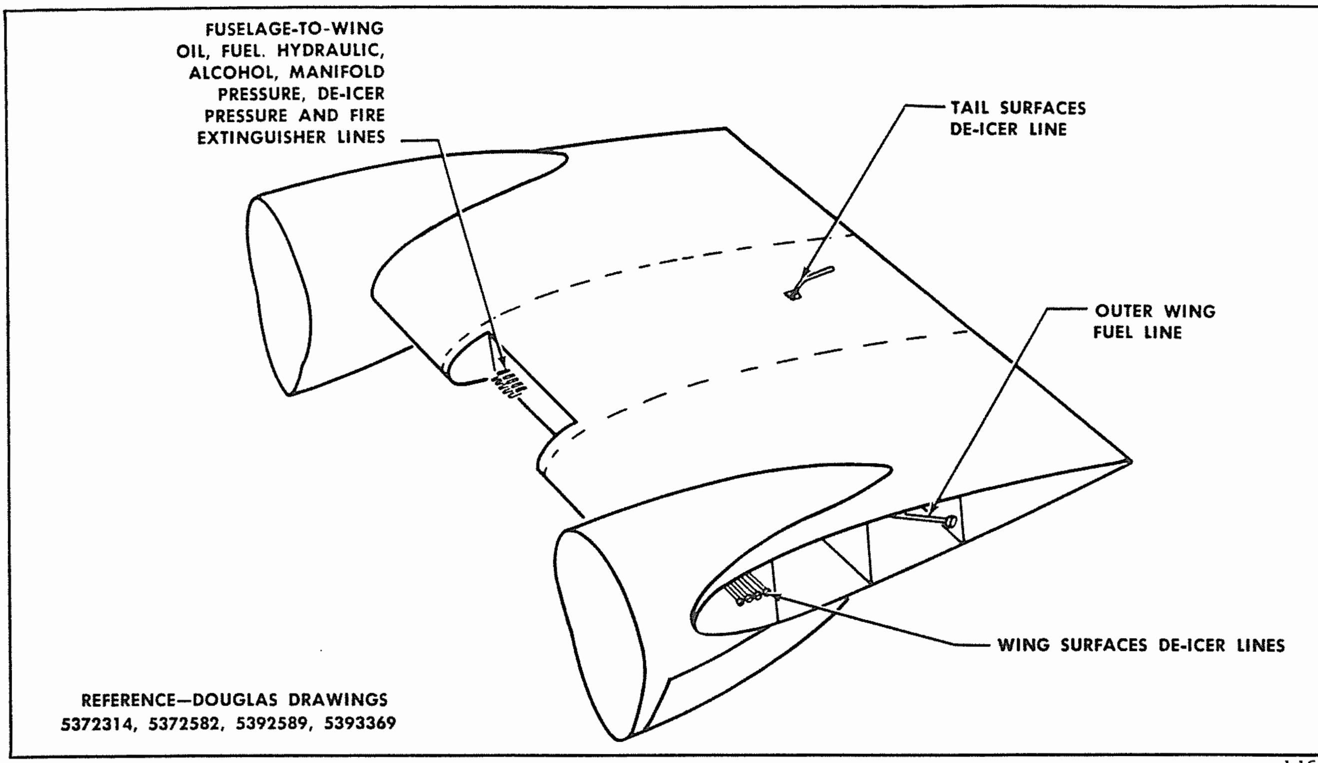


Figure 2-4 (Sheet 2 of 2 Sheets). Wing Center Section Installation

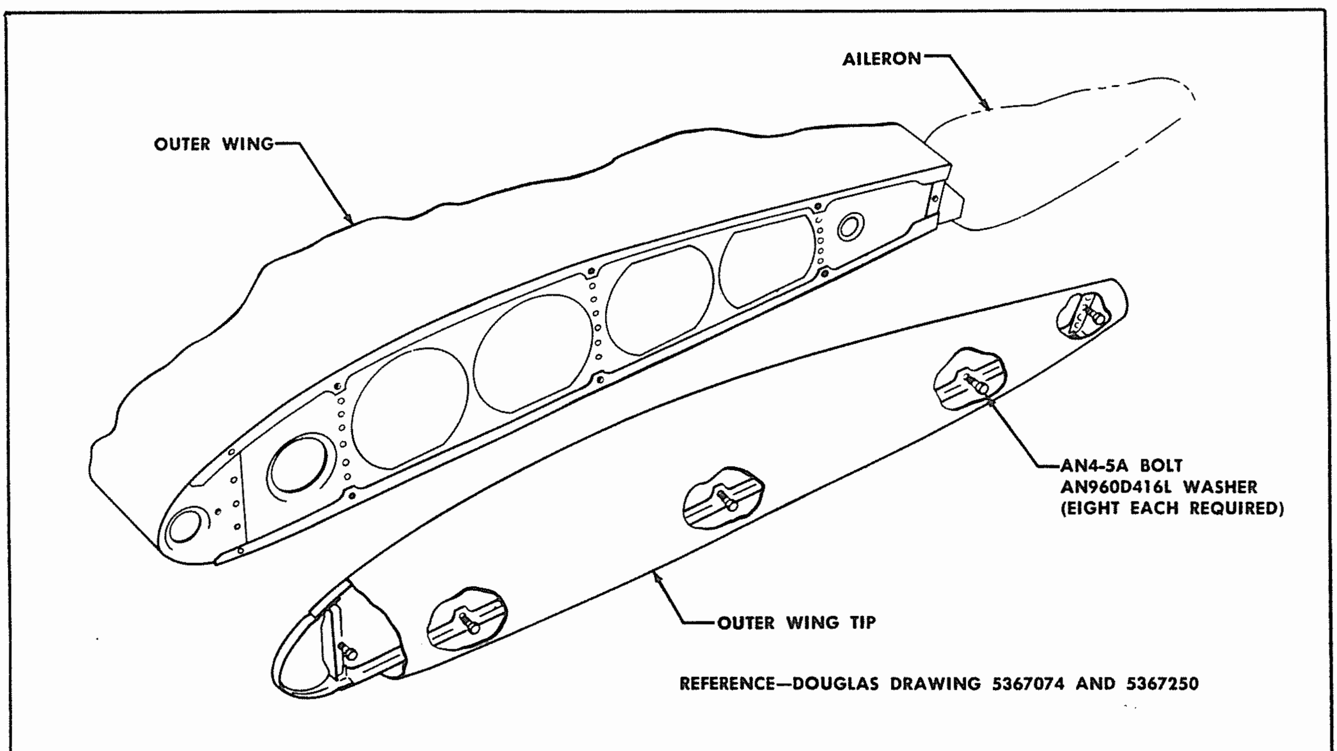


Figure 2-5. Wing Tip Installation

2-27. EMPENNAGE.

2-28. DESCRIPTION. *(See figures 2-6 and 2-7.)* The empennage, or tail group, consists of the horizontal and vertical stabilizers, the elevators and the rudder. The horizontal stabilizer is an aluminum alloy structure, and is interchangeable between left and right sides. The stabilizers are bolted together at the root ends of the surface structures and are removed or installed as an integral unit. A de-icer shoe is installed on the leading edge of each structure. The horizontal stabilizer is attached to the fuselage with bolts inserted through attach angles riveted to the stabilizer. Four hinge brackets are bolted to the horizontal stabilizer ribs for attachment to the elevators. The vertical stabilizer is of similar construction, with a de-icer shoe on its leading edge. The vertical stabilizer is attached to the fuselage by flush-head machine screws. The base of the aft end of the stabilizer is bolted to the fuselage frame. Two hinge fittings are bolted to the stabilizer structure for attachment to the rudder. For information on the elevators and the rudder, see paragraphs 2-117 and 2-171.

2-29. CLEANING OF EMPENNAGE. For the procedure to be followed in cleaning the empennage, see paragraph 2-3.

2-30. HORIZONTAL STABILIZER.
(See figure 2-6.)

2-31. REMOVAL OF HORIZONTAL STABILIZER. It is to be noted that no fuselage hoist, jack, or other special support is required for the removal of the horizontal stabilizer; however, four men are required for the operation.

a. Disconnect the tail light wires at the junction box in the tail cone, working through the access door on the right side of the tail cone *(see figure 1-9).*

b. Remove the tail cone by removing the machine screws that hold it to the fuselage.

c. Remove the elevators (see paragraph 2-118).

d. Disconnect the elevator control cables at the elevator horn.

e. Remove the elevator trim tab control cable (see paragraph 2-168).

f. Detach the fuselage-to-stabilizer fillets by removing the screws.

g. Detach the de-icer pipe leading to each stabilizer.

h. Remove the bolts that attach the stabilizer to the fuselage.

i. Remove the two channels bolted to the aft end of the fuselage.

j. Pull the stabilizer aft until it is clear of the fuselage.

Note

If the two halves of the stabilizer are to be separated, first remove the elevator torque tube.

2-32. INSTALLATION OF HORIZONTAL STABILIZER.
(See figure 2-6.)

a. Carefully slide the horizontal stabilizer into the opening at the aft end of the fuselage. If the two (interchangeable) stabilizer halves have not been mated, they should be bolted together before installation on the fuselage. Torque the bolts to 40 ($\pm$5) inch-pounds.

b. Attach the two channels with bolts to the aft end of the fuselage.

c. Install the bolts that attach the horizontal stabilizer to the fuselage.

d. Install the de-icer pipe leading to each side of the stabilizer.

e. Fasten the stabilizer fillets to the fuselage with their attaching screws.

f. Connect the elevator control cables at the elevator horns (see paragraph 2-122).

g. Install the elevators as described in paragraph 2-122.

h. Install the tail cone to the fuselage as described in paragraph 2-339.

i. Connect the tail light wires at the junction box in the tail cone, working through the access door on the right side of the tail cone *(see figure 1-9).*

2-33. HORIZONTAL STABILIZER TIP.

2-34. REMOVAL OF HORIZONTAL STABILIZER TIP. Open the access doors on the trailing edges of the right and left stabilizers at the tips *(see figure 1-9).* Working through the doors, disconnect the bolts that fasten the tips to the stabilizers.

2-35. INSTALLATION OF HORIZONTAL STABILIZER TIP. Reverse the removal procedure.

2-36. VERTICAL STABILIZER.
(See figure 2-7.)

2-37. REMOVAL OF VERTICAL STABILIZER. Proceed as follows:

a. Remove the rudder (see paragraph 2-172).

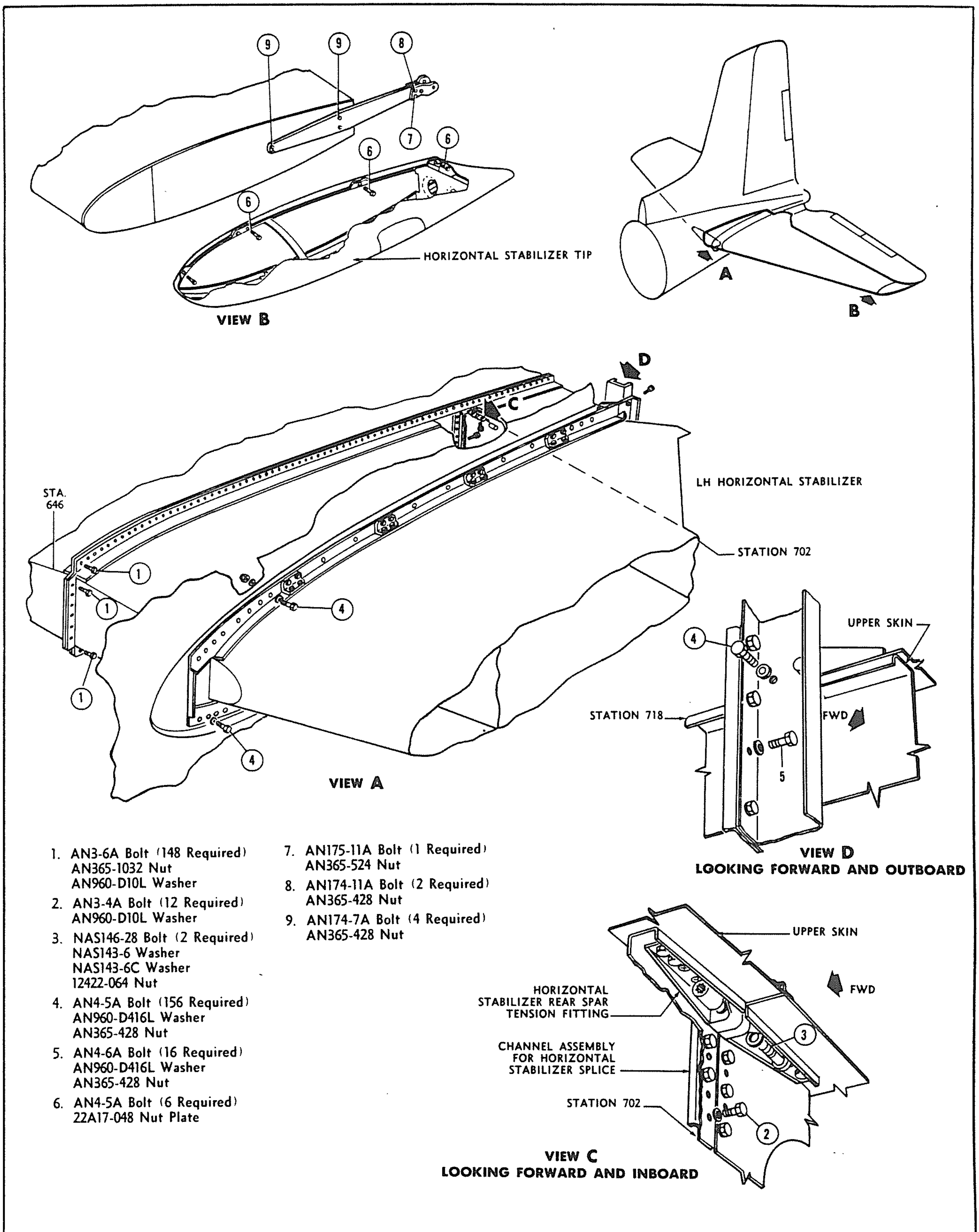
b. Remove the rudder trim control tab cable (see paragraph 2-215).

c. Working through the access door on the stabilizer tip *(see figure 1-9),* disconnect the antenna wire and pull it out through the hole in the de-icer shoe, breaking the plastic seal if necessary.

d. Remove the fairing strips and collar from the bottom end of the de-icer shoe.

e. Working through the access door in the dorsal fin forward of the vertical stabilizer *(see figure 1-9),* disconnect the de-icer shoe pipe.

f. Remove the bolts that attach the rear spar of the vertical stabilizer to the fuselage.



1. AN3-6A Bolt (148 Required)
   AN365-1032 Nut
   AN960-D10L Washer
2. AN3-4A Bolt (12 Required)
   AN960-D10L Washer
3. NAS146-28 Bolt (2 Required)
   NAS143-6 Washer
   NAS143-6C Washer
   12422-064 Nut
4. AN4-5A Bolt (156 Required)
   AN960-D416L Washer
   AN365-428 Nut
5. AN4-6A Bolt (16 Required)
   AN960-D416L Washer
   AN365-428 Nut
6. AN4-5A Bolt (6 Required)
   22A17-048 Nut Plate
7. AN175-11A Bolt (1 Required)
   AN365-524 Nut
8. AN174-11A Bolt (2 Required)
   AN365-428 Nut
9. AN174-7A Bolt (4 Required)
   AN365-428 Nut

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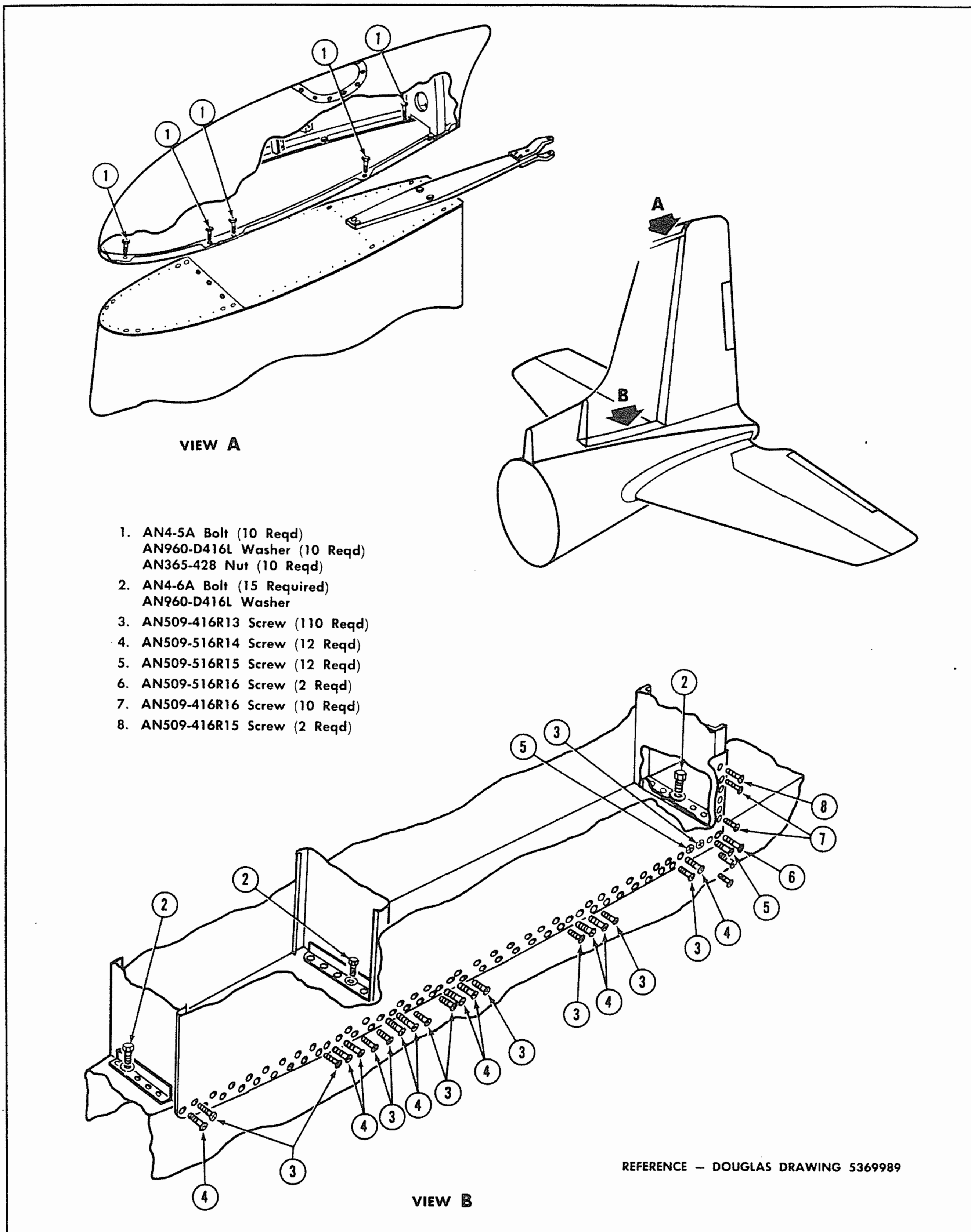
Figure 2-6. Horizontal Stabilizer Installation



1. AN4-5A Bolt (10 Reqd)
   AN960-D416L Washer (10 Reqd)
   AN365-428 Nut (10 Reqd)
2. AN4-6A Bolt (15 Required)
   AN960-D416L Washer
3. AN509-416R13 Screw (110 Reqd)
4. AN509-516R14 Screw (12 Reqd)
5. AN509-516R15 Screw (12 Reqd)
6. AN509-516R16 Screw (2 Reqd)
7. AN509-416R16 Screw (10 Reqd)
8. AN509-416R15 Screw (2 Reqd)

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Figure 2-7. Vertical Stabilizer Installation

g. Remove the machine screws that attach the stabilizer to the fuselage. (As these screws vary in length, it will be well to note where the various sizes are located.)

h. Tilt the stabilizer aft and lift it from the fuselage.

2-38. INSTALLATION OF VERTICAL STABILIZER.
(See figure 2-7.)

a. Mount the forward end first, then press the aft end into place. Strips of metal will be helpful in fitting the stabilizer skin over the fuselage skin. One screw may be attached on the forward end of the splice to serve as a fulcrum and guide the aft part into place.

b. Install all the machine screws that attach the stabilizer to the fuselage and the dorsal fin.

c. Install the bolts that attach the rear spar of the vertical stabilizer to the fuselage. To draw up these bolts it will be necessary to loosen the lower end of the cloth weather cover screwed to the left side of the spar.

d. Install the access door in the dorsal fin forward of the stabilizer, after the de-icer shoe pipe has been connected.

e. Install the collar and fairing strips at the bottom of the de-icer shoe.

f. Connect the antenna wire to the fitting inside the stabilizer tip, and weather-seal the hole in the de-icer shoe with dope, if necessary.

2-39. VERTICAL STABILIZER TIP.

2-40. REMOVAL OF VERTICAL STABILIZER TIP. Remove the cover of the access door on the stabilizer tip. Working through the access door, detach the antenna wire, and pull it through the hole in the de-icer shoe, breaking the seal if necessary. Remove the bolts that attach the tip to the stabilizer, and remove the tip.

2-41. INSTALLATION OF VERTICAL STABILIZER TIP. Reverse the removal procedure, weather-sealing the antenna wire hole in the de-icer shoe, if necessary.

2-42. EMPENNAGE BOLT TORQUE VALUES. For information on empennage bolt torque values, see paragraph 2-453.

2-43. SURFACE CONTROLS.

2-44. DESCRIPTION. Surface controls consist of all controlling units necessary for the operation of the ailerons, aileron tabs, elevators, elevator tabs, rudder, rudder tab, and wing flaps. All movable control surfaces, except the wing flaps, are controlled from the flight compartment through conventional two-way closed cable systems; dual controls are provided for the ailerons, elevators, and the rudder. Automatic pilot servo units are connected into the aileron, elevator, elevator trim tab, and rudder cable systems. The wing flaps, which also are controlled from the flight compartment, are operated hydraulically. A piano wire-operated wing flap position indicator is installed to the left of the main instrument panel in the flight compartment.

2-45. CONTROL CABLES. Aircraft cables are made from high-tensile-strength wires, which are grouped and formed into the strands that compose the cable. Generally, these strands are put through a preforming process, which forms them into the shape they will have in the finished cable. As a result of the preforming process, the cable does not have a tendency to fray or unlay when cut.

2-46. Cable construction is designated by numbers. The first number denotes the number of strands in the cable and the second denotes the number of wires in each strand. For example, a 7 x 19 cable is composed of 7 strands, with 19 wires in each strand. The diameter of a cable is the diameter of the circle that will enclose the cable. To obtain an accurate measurement of the diameter of a cable, the gage must be rotated completely around the cable.

2-47. Fittings are swaged onto cable ends with a swaging machine. The sleeve is pressed hard enough to cause the inside of the sleeve to take the shape of the cable, since the terminal is of softer material than the cable. After a terminal has been swaged to a cable, the terminal cannot be removed and used again.

2-48. SURFACE CONTROLS ILLUSTRATIONS AND CABLE CHARTS. A set of cable routing master drawings, showing cable routing at all major pulley brackets and fairleads, and at the firewall seals, is provided to facilitate finding the control system to which a particular cable applies. All cables affected are called out by run numbers. These run numbers and the systems to which they pertain are listed in the item list, sheet 1 of the illustration. The index, which precedes the item list, locates the various system illustrations.

2-49. The various control systems of the aircraft are illustrated individually, with complete information necessary to the text graphically presented on each series of drawings. A typical breakdown of such a series, listed as one figure number, consists of a key drawing of the entire system; details of the necessary controlling units; pulley brackets, fairleads, and firewall seals; rigging and adjustment diagram; cable chart; and cable assembly drawings. Controlling units are shown from a top or bottom view, depending upon the position from which maintenance work will be performed.

2-50. A cable chart contains the data necessary for the repair and replacement of all cables installed within the system represented. The cable reference number (Cable Ref. No.) on the chart is the same as the reference number on the illustrations where the cable is shown. Coding letters in the column headed "Type" correspond to the coding letters on the cable assembly drawings that follow the chart, and numbers in the column headed "Fittings" correspond to the fitting numbers on the cable assemblies.

2-51. INSPECTION OF CONTROLS. Inspection of the controls should be performed at regular intervals and also after exposure to severe operating conditions. In some cases, grease will have to be removed from cables for thorough inspection.

CAUTION

Never use degreaser on the control cables. Degreasing cuts the lubrication in the soft center of the cable, causing the cable to break down more readily through internal friction.

Cables should be relubricated immediately after inspection is completed *(see figure 1-42)*. A thorough inspection of the controls should include the following operations:

a. Check the security and condition of all pulley brackets.

b. Make certain that all pulley guard pins are in place.

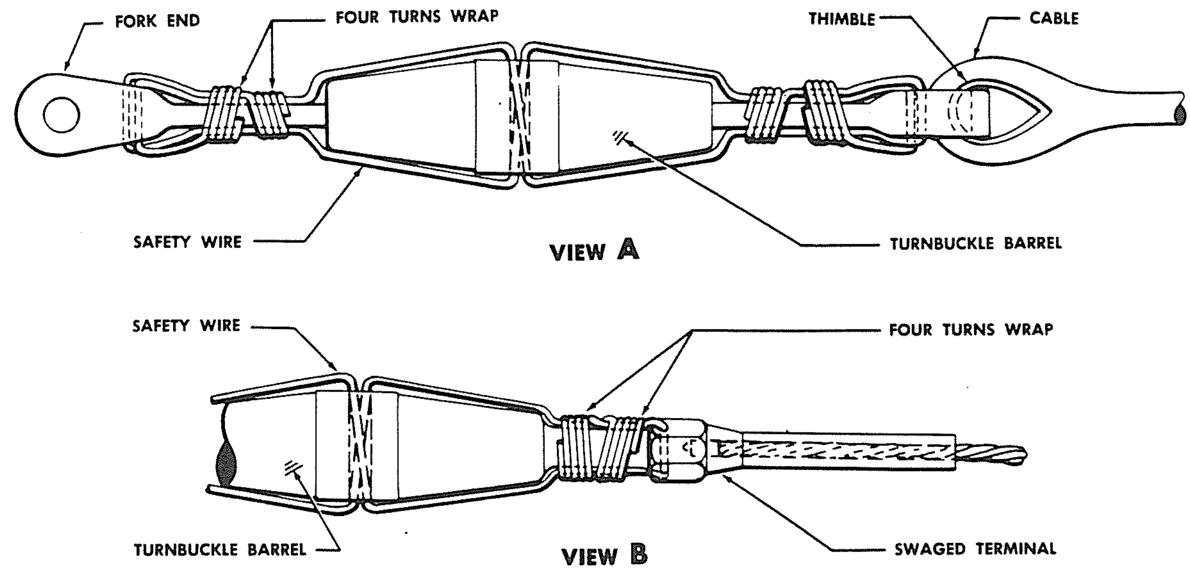
c. Inspect for broken or cracked pulleys and pulley flanges, frozen bearings and flat spots on pulley grooves. Replace any damaged parts.

d. See that all cotter pins are secure.

e. Inspect fairlead grommets and rubstrips and replace any that are damaged or worn excessively.

f. Inspect all turnbuckles to see that they are safetied *(see figure 2-8)*.

g. Examine the control cables for broken wires. A cable passing around a pulley weakens from fatigue, caused by bending, internal friction, and wearing of wires passing over one another. Whenever more than six wires are broken in any one inch length of cable, the cable should be replaced. Examine the cable for broken wires by running a cloth over it, thus preventing injury to the hands. Rust, dust, or dirt at the point of operation will tend to shorten the life of the cable. Before installing a new cable, it should be placed in a hydraulic jack, preloaded to 60 per cent of its breaking strength, and this tension held for one minute; this operation will prevent excessive cable stretching after installation.



TURNBUCKLE SAFETYING REQUIREMENTS

CABLE SIZE (INCH)	DIAMETER OF WIRE (INCH)	MATERIAL	SPECIFICATIONS
1/16	.032	Nickel-Copper Alloy (Monel)	Federal ZZ-N-281, Class A, Annealed
		Nickel-Chromium-Iron Alloy (Inconel)	Air Force-Navy AN-N-4, Cond. A, Annealed
		*Carbon Steel-Zinc Coated	AN-W-22
		Corrosion Resisting Steel	AN-W-24, Cond. A
3/32 or 1/8	.040	Nickel-Copper Alloy (Monel)	Federal ZZ-N-281, Class A, Annealed
		Nickel-Chromium-Iron Alloy (Inconel)	Air Force-Navy AN-N-4, Cond. A, Annealed
5/32 and greater	.047	Carbon-Steel-Zinc Coated	AN-W-22
		Corrosion Resisting Steel	AN-W-24, Cond. A

*Tolerance on diameter of .032 zinc coated carbon steel wire is plus or minus .001 inch.

Notes:

1. The assembled turnbuckle should show the last thread on each turnbuckle shank flush with the end of the barrel. This is considered the best position of the shanks relative to the barrel. However, not more than three threads may be exposed on the turnbuckle shank.
2. After the turnbuckle is adjusted to its best locking position, pass two safety wires through the hole in the center of the turnbuckle barrel, and bend the ends of the wire 90 degrees toward the ends of the turnbuckle, as shown in View A. Pass the ends of the wire through the hole in the turnbuckle eyes or between the jaws of the forked ends. Then bend the ends back toward the center of the turnbuckle and wrap each end four times around the shank, binding the wrapping wires in place, as shown in View A. When a swaged terminal is being safetied, pass a wire through the hole provided for this purpose in the terminal, loop over the free end of the other wire, and wrap both ends around the shank, as shown in View B.

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Figure 2-8. Turnbuckle Safetying Procedure

h. Operate each individual control system and check for free movement of controls, bellcranks, quadrants, pulleys, levers, and control surfaces; correct cable tension *(see figure 2-9)*; correct degree of travel and direction of movement in control units.

2-52. LUBRICATION REQUIREMENTS FOR SURFACE CONTROLS.

a. Remove excess lubrication and dirt from all the surface controls and cables, using a cloth moistened with solvent.

CAUTION

Do not use a saturated cloth on control cables, as excess solvent cuts the lubrication in the center of the cable, permitting rapid wear and corrosion.

b. Every 1000 hours, or as conditions warrant, perform the following:

c. Inspect the cable coatings on lengths running on pulleys, drums, and fairleads. Touch up where necessary with corrosion preventative compound CPC, sufficiently fluidized to permit penetration by the compound and to prevent the surface from becoming tacky.

d. Apply oil lubricant OGP to all cable clevis attach points.

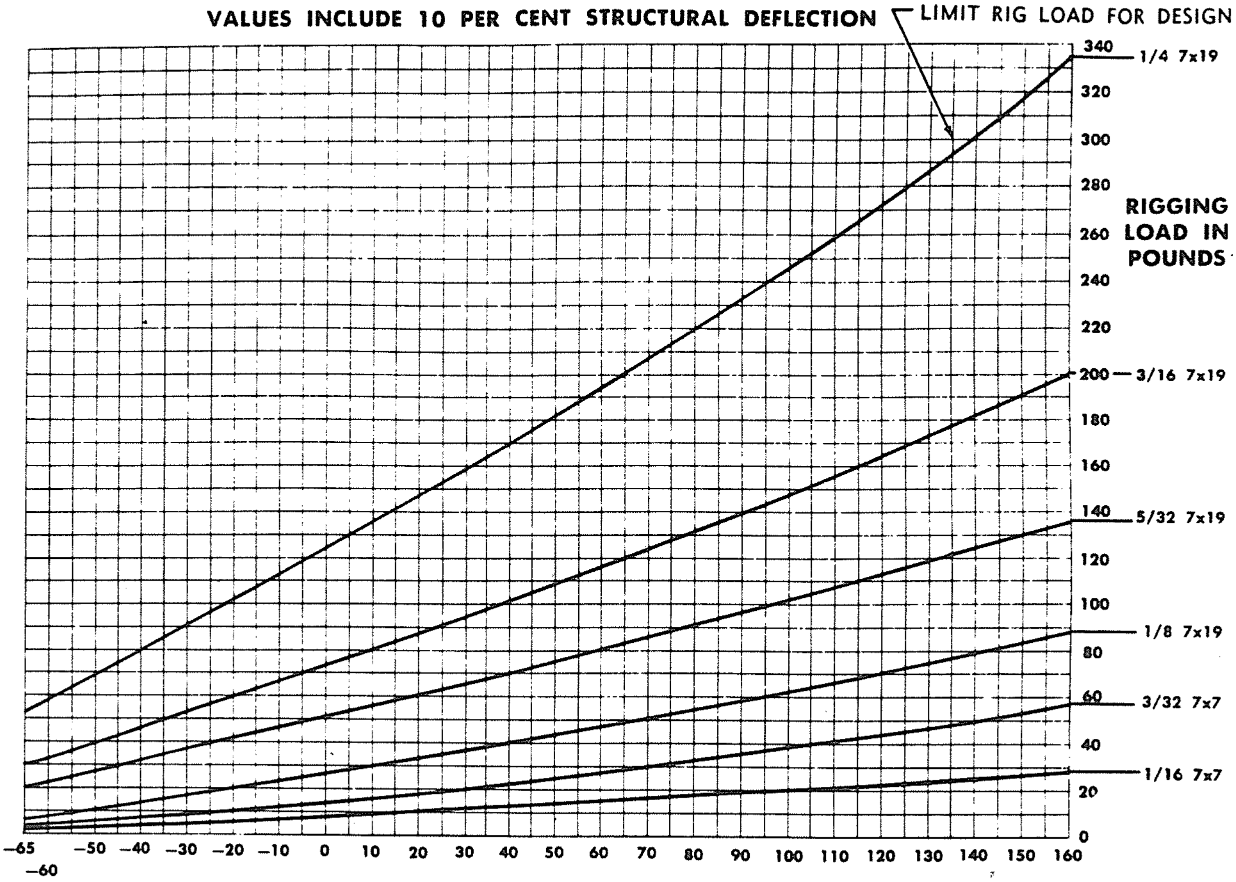
Note

For specifications and recommended lubrication procedures of other components, see figure 1-42.

2-53. TROUBLE SHOOTING OF SURFACE CONTROLS.

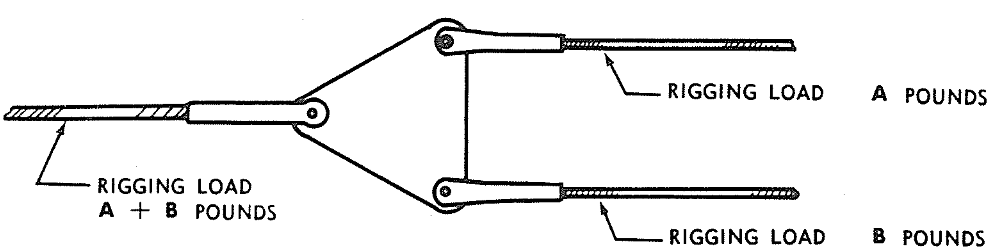
Whenever an individual control system is malfunctioning, it will be necessary to perform a thorough inspection of that particular system to determine the cause of improper performance. The trouble may be caused by an unsafetied turnbuckle, a broken pulley, a missing guard pin causing the cable to jump the pulley, or improper adjustment. The chart that follows lists some of the most common causes of malfunctioning. After eliminating the cause of improper performance, it is imperative that the adjustment of the entire system be checked.

Trouble	*Probable Cause*	*Remedy*
a. Controls feel sluggish. Movement of controls results in springy feeling. Surface lags behind control movement.	Cables are slack.	Rig cables to correct tension.
	Lost motion in rods or linkage.	Replace bolts or bushings, where necessary, in rods or linkage to eliminate play.
b. Controls are stiff. Surfaces do not return to neutral of their own accord in flight.	Tension in control cables too great.	Rig cables to correct tension.
	Fairleads or pulley brackets out of alignment.	Adjust fairleads or pulley brackets for correct alignment. *Maximum* allowable cable pull-off is two degrees.
	Ball bearings defective or frozen. If system is free from interference of foreign objects, tensions are satisfactory and all units are found in correct alignment, bearing failure is indicated.	Check each bearing in system and replace any which bind or show indications of failure. Relubricate bearings, where necessary.
	Bearings preloaded. Turn smoothly but with high friction.	Check for improper installation. Make certain that correct spacer or washer is used. Check control column head bearings and aileron crank bearings for tight bolts.
c. Trim control binds.	Tension in control cables too great.	Rig cables to correct tension.
	Fairleads or pulley brackets out of alignment.	Adjust fairleads or pulley brackets for correct alignment. *Maximum* allowable cable pull-off is two degrees.
	Tab drive mechanism needs lubrication.	Apply grease on threads of tab pushrods and drive drums *(see figure 1-42)*.
d. Tab creeping and/or acting erratically.	Excessive play in drive mechanism.	Check adjustment of drive mechanism; replace worn parts.
	Cables rigged too loosely.	Rig cables to correct tension. If all other causes are eliminated and tabs still creep, rerig cables to 1½ times correct tension.
	Loose flight compartment indicator.	Calibrate indicator. Adjust or replace worn parts.
	Distorted closing channel in front of tab.	Straighten channel.
	Tab drive mechanism out of alignment with tab pushrods.	Disconnect pushrods at tab to see if they have been sprung out of line. Correct alignment by shimming under drum assembly mounting bolts.
	Threads are binding in tab drive mechanism.	Lubricate threads. If condition is unrelieved, disconnect one rod and see if length is correct.



Notes:

1. Reliable tensiometers must be used for adjusting the rigging loads and the accuracy of the rigging load maintained within ±10% on 1/8-inch cables and larger and within ± 5 pounds on 3/32 and 1/16-inch cables.
2. The chart values are for each cable, whether single or double system. In cases where 2 cables join a third, the load on the third cable will be the sum of the loads in the first 2 cables, as rigged from the chart. See example below.
3. Reference — Douglas drawing 4155390.



EXAMPLE OF DOUBLE SYSTEM RIGGING LOADS

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Figure 2-9. Cable Rigging Tension Chart

Trouble	Probable Cause	Remedy
e. Controls have a tendency to foul momentarily or have an erratic motion.	Frayed cables.	Replace cables when there are more than six broken wires in any working area. A working area is that length of cable which passes over a pulley or through a fairlead.
	Broken pulley. Foreign objects, especially in vicinity of turnbuckles. Cable off pulley, or improperly routed.	Carefully inspect control system and eliminate cause of trouble.
f. Controls apparently out of adjustment.	Control stops incorrectly adjusted.	Adjust stops.
	Pushrods or other linkage bent by misuse.	Replace bent pushrods or linkage.
g. Excessive vibration on control surfaces and tabs.	Loose or worn hinges, bolts and or operating mechanism.	Tighten hinge bolts to proper torque. Check operating mechanism. Replace necessary parts.
	Loose fabric or inspection doors.	Repair fabric and tighten inspection doors.
	Improperly secured balance weight.	Secure balance weight.
	Improperly balanced surface or tab.	Rebalance.
h. Wing heaviness.	Aileron trim control indicator loose or bent.	Readjust neutral point of indicator and check for proper tab throws.
	Warped aileron.	Straighten or replace aileron.
	Misalignment of hinges.	Align hinges. Check vertical alignment of hinges.
	Flaps rigged out of line in UP position.	Rerig flaps.
i. Erratic aileron control forces in flight.	Misalignment of hinges, causing aileron warpage.	Align hinges. Check vertical alignment of hinges.
	Warped aileron.	Straighten or replace aileron.
	Improperly rigged control system.	Rerig system. Check all control components for proper location in neutral.
	Excessive control system friction.	Check complete system for causes of friction, such as improper cable pull-off, defective bearings, improper rigging.

2-54. CABLE ROUTING MASTER DRAWINGS FOR PULLEY BRACKETS, FAIRLEADS, AND FIREWALL SEALS. Figure 2-10 contains views of all major pulley brackets and fairleads used throughout the aircraft. Correct cable positions are shown.

2-55. PULLEYS AND PULLEY BRACKETS. Pulleys are used throughout the aircraft to change the direction of control cable routing. Brackets, fastened to the structure of the aircraft, support the pulleys in position. Guard pins are used to maintain proper cable position in regard to the pulleys.

2-56. REMOVAL OF PULLEYS AND PULLEY BRACKET. If control cables are routed through the pulley bracket assembly in such a manner that removal of the bracket attaching bolts and guard pins will not release the pulleys and bracket from the cables, the control cables must be released at the nearest turnbuckles and pulled through the bracket assembly prior to removing the bracket attaching bolts.

2-57. INSTALLATION OF PULLEYS AND PULLEY BRACKET. Place the pulley bracket, with pulleys attached, in position and bolt it to the structure of the aircraft. Route appropriate cables through the pulley bracket, fasten the turnbuckles, and secure the guard pins. Tension the cables in accordance with the Cable Rigging Tension Chart, figure 2-9, and safety the turnbuckles.

2-58. CABLE ROUTING THROUGH PULLEY BRACKETS. For routing of cables through major pulley brackets, see figure 2-10.

2-59. FAIRLEADS. Fairleads are installed throughout the aircraft to control the positions of the cables in straight runs and to provide cable passage through supporting structure.

2-60. REMOVAL OF FAIRLEADS. Release the necessary cables at the nearest turnbuckles, remove the fairlead grommets, and pull the cables through the fairlead. Remove the fairlead.

2-61. INSTALLATION OF FAIRLEADS. Secure the fairlead in the proper position and route the affected cables through the fairlead, replacing fairlead grommets. Fasten turnbuckles, check cable tension, and safety the turnbuckles.

2-62. CABLE ROUTING THROUGH FAIRLEADS. For routing of cables through the major fairleads see figure 2-10.

2-63. RUBSTRIPS. Rubstrips are provided at various locations to prevent the cables from rubbing on structural members, thus preventing excessive cable wear.

2-64. REMOVAL OF RUBSTRIPS. Some rubstrips may be readily removed without affecting the cables for which they are provided. Others can be removed only after disconnecting the cables at the nearest turnbuckles and pulling the cables through the rubstrips.

2-65. INSTALLATION OF RUBSTRIPS. Install the rubstrip in position. If appropriate, route the cable through the rubstrip and fasten the turnbuckles. Check cable tension and safety the turnbuckles.

(Continued on Page 98)

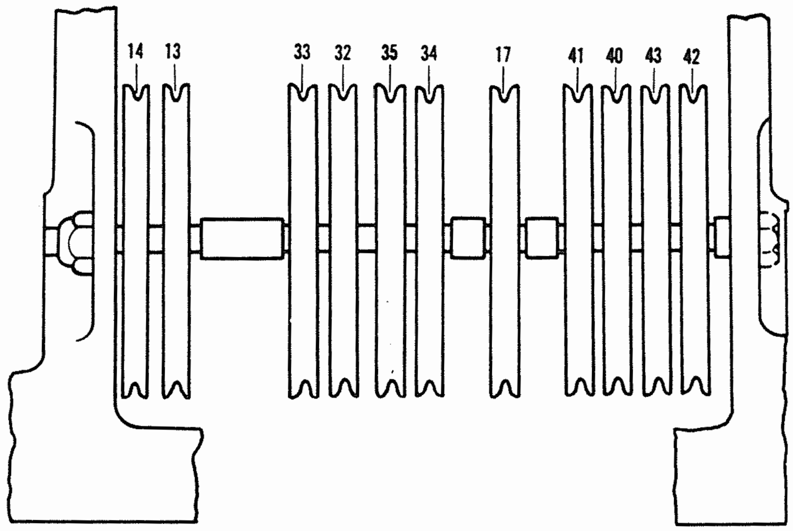
Index

The following list of illustrations includes only those systems which make use of the fairleads and pulley brackets shown in these master drawings.

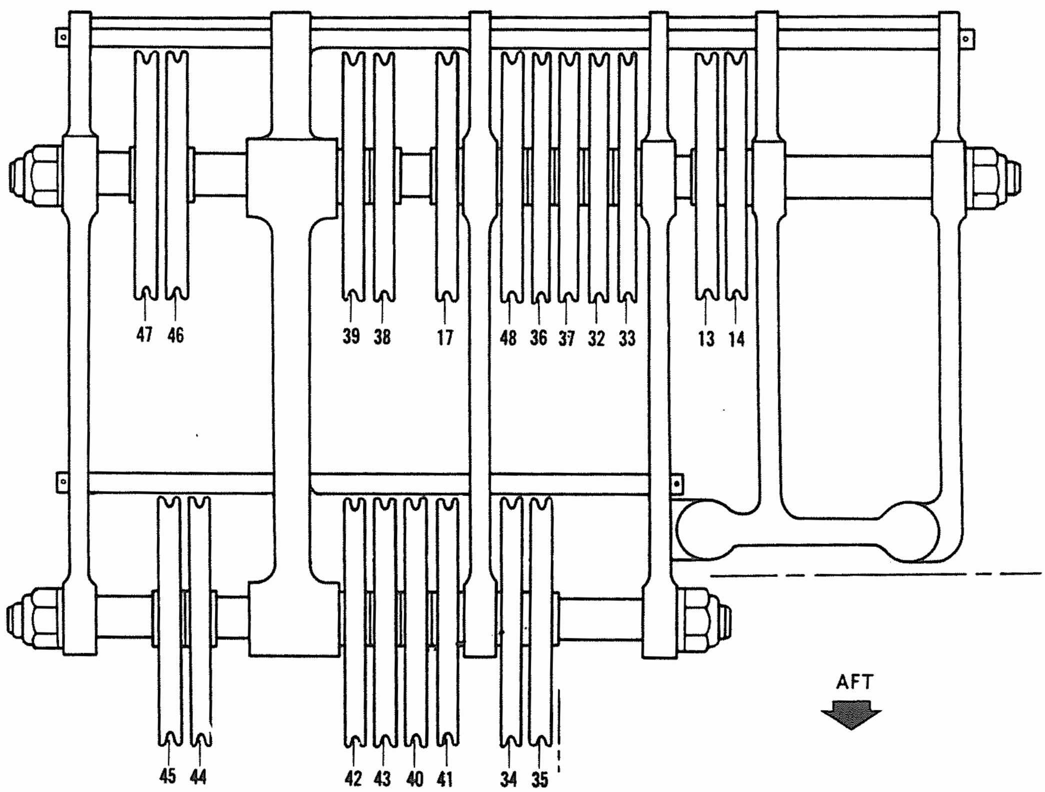
Key to Figure 2-10

1. Left Aileron Trailing Edge Up
2. Right Aileron Trailing Edge Up
3. Aileron Trim Control, Right Aileron Tab Trailing Edge Up
4. Aileron Trim Control, Right Aileron Tab Trailing Edge Down
5. Rudder Trailing Edge Left
6. Rudder Trailing Edge Right
7. Rudder Trim Control, Rudder Tab Trailing Edge Left
8. Rudder Trim Control, Rudder Tab Trailing Edge Right
9. Elevator Trailing Edges Up (Left Upper Cables)
10. Elevator Trailing Edges Down (Left Lower Cables)
11. Elevator Trailing Edges Up (Right Upper Cables)
12. Elevator Trailing Edges Down (Right Lower Cables)
13. Elevator Trim Tab Trailing Edges Up
14. Elevator Trim Tab Trailing Edges Down
15. Mechanical Safety Latch, Lock
16. Mechanical Safety Latch, Unlock
17. Tail Gear Control
18. Surface De-Icing, On
19. Surface De-Icing, Off
20. Fire Emergency Shutoff Open, Left Engine
21. Fire Emergency Shutoff Closed, Left Engine
22. Fire Emergency Shutoff Open, Right Engine
23. Fire Emergency Shutoff Closed, Right Engine
24. Fuel Tank Selector Valve On, Left Center Wing
25. Fuel Tank Selector Valve Off, Left Center Wing
26. Fuel Tank Selector Valve On, Left Outer Wing
27. Fuel Tank Selector Valve Off, Left Outer Wing
28. Fuel Tank Selector Valve On, Right Center Wing
29. Fuel Tank Selector Valve Off, Right Center Wing
30. Fuel Tank Selector Valve On, Right Outer Wing
31. Fuel Tank Selector Valve Off, Right Outer Wing
32. Propeller Pitch Increase, Left Engine
33. Propeller Pitch Decrease, Left Engine
34. Propeller Pitch Increase, Right Engine
35. Propeller Pitch Decrease, Right Engine
36. Throttle Advance, Left Engine
37. Throttle Close, Left Engine
38. Throttle Advance, Right Engine
39. Throttle Close, Right Engine
40. Mixture Enriching, Left Engine
41. Mixture Leaning, Left Engine
42. Mixture Enriching, Right Engine
43. Mixture Leaning, Right Engine
44. Carburetor Air Hot, Left Engine
45. Carburetor Air Cold, Left Engine
46. Carburetor Air Hot, Right Engine
47. Carburetor Air Cold, Right Engine
48. Automatic Pilot Release
49. Automatic Pilot Elevator Trim Tab Servo, Tab Trailing Edges Up
50. Automatic Pilot Elevator Trim Tab Servo, Tab Trailing Edges Down

Figure 2-10 (Sheet 1 of 10 Sheets). Cable Routing Master Drawings — Index and Item List



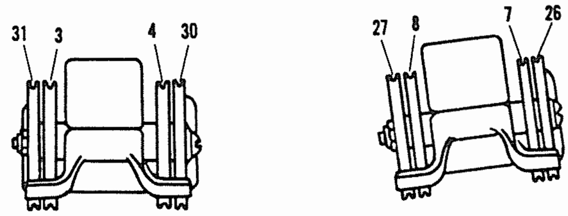
PULLEYS IN CONTROL PEDESTAL
LOOKING FORWARD



PULLEY BRACKET
STATION 10 (UNDER CONTROL PEDESTAL)
BOTTOM VIEW

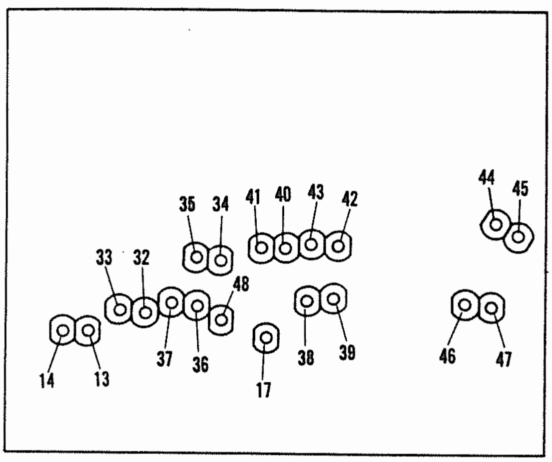
1,171

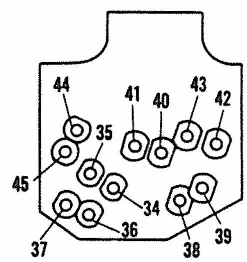
Figure 2-10 (Sheet 2 of 10 Sheets). Cable Routing Master Drawings — Pulley Brackets, Control Pedestal and Station 10



PULLEY BRACKETS, STATION 10 — LOOKING AFT

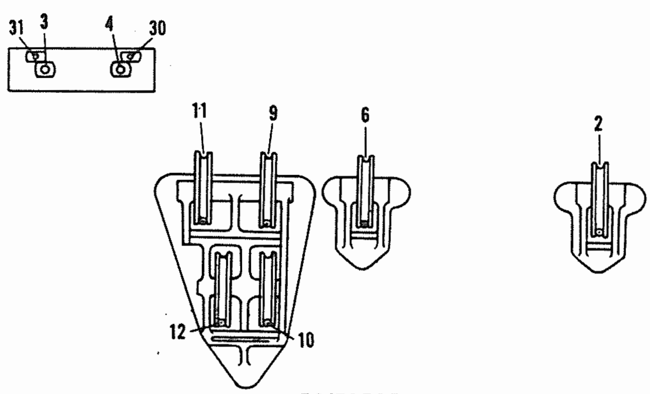
PULLEY BRACKETS, STATION 26 — TOP VIEW





FAIRLEAD, STATION 64
LOOKING FORWARD

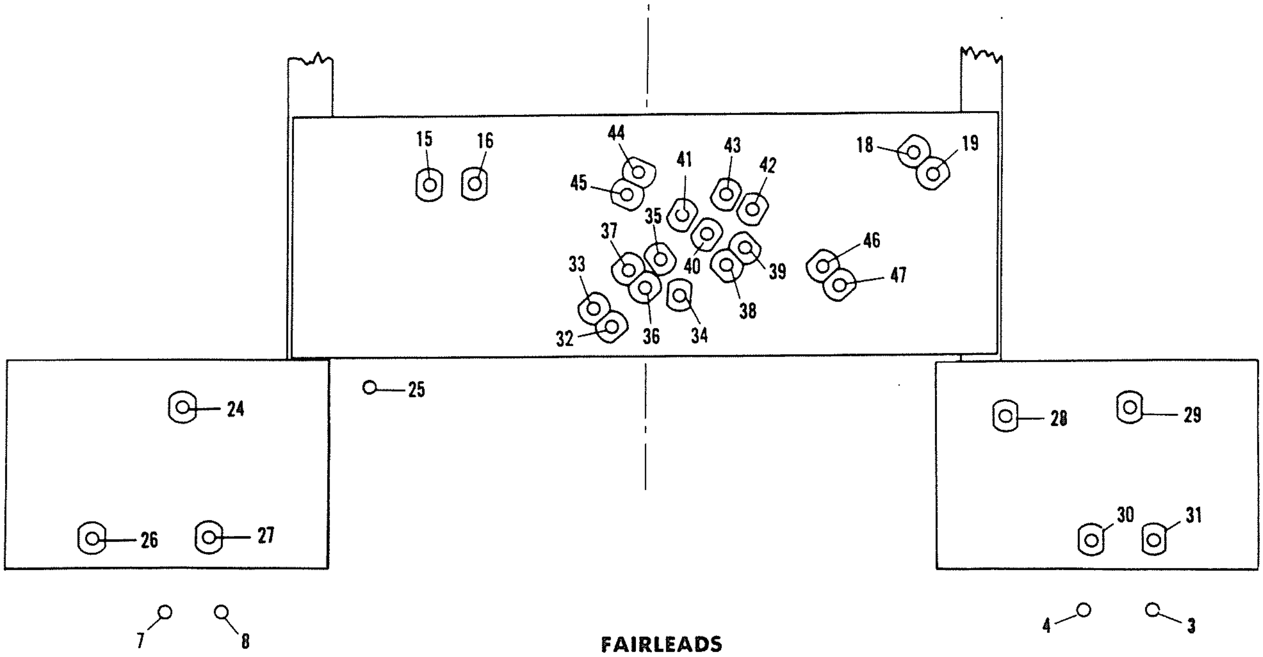
FAIRLEAD, STATION 24 — LOOKING FORWARD



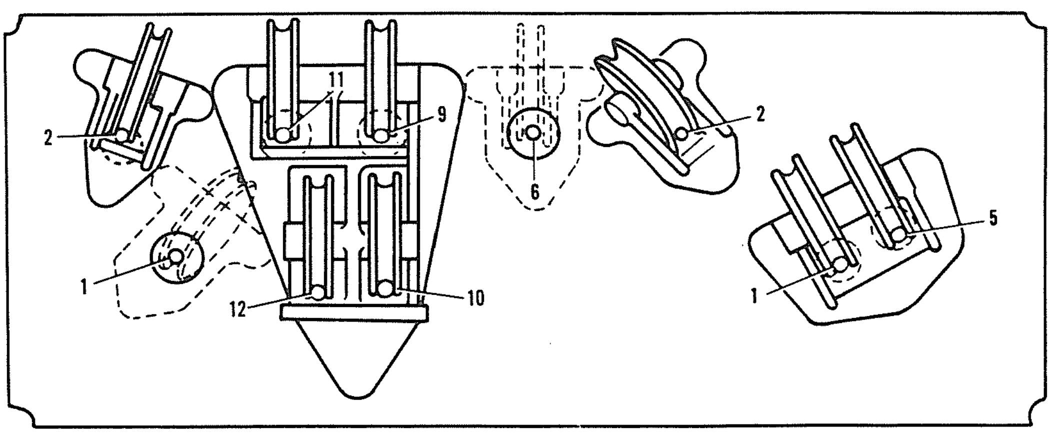
FAIRLEAD AND PULLEY BRACKET
STATION 37 — LOOKING AFT

1,172

Figure 2-10 (Sheet 3 of 10 Sheets). Cable Routing Master Drawings — Pulley Brackets, Stations 10 and 26; Fairleads, Stations 24 and 64; Fairlead and Pulley Brackets, Station 37



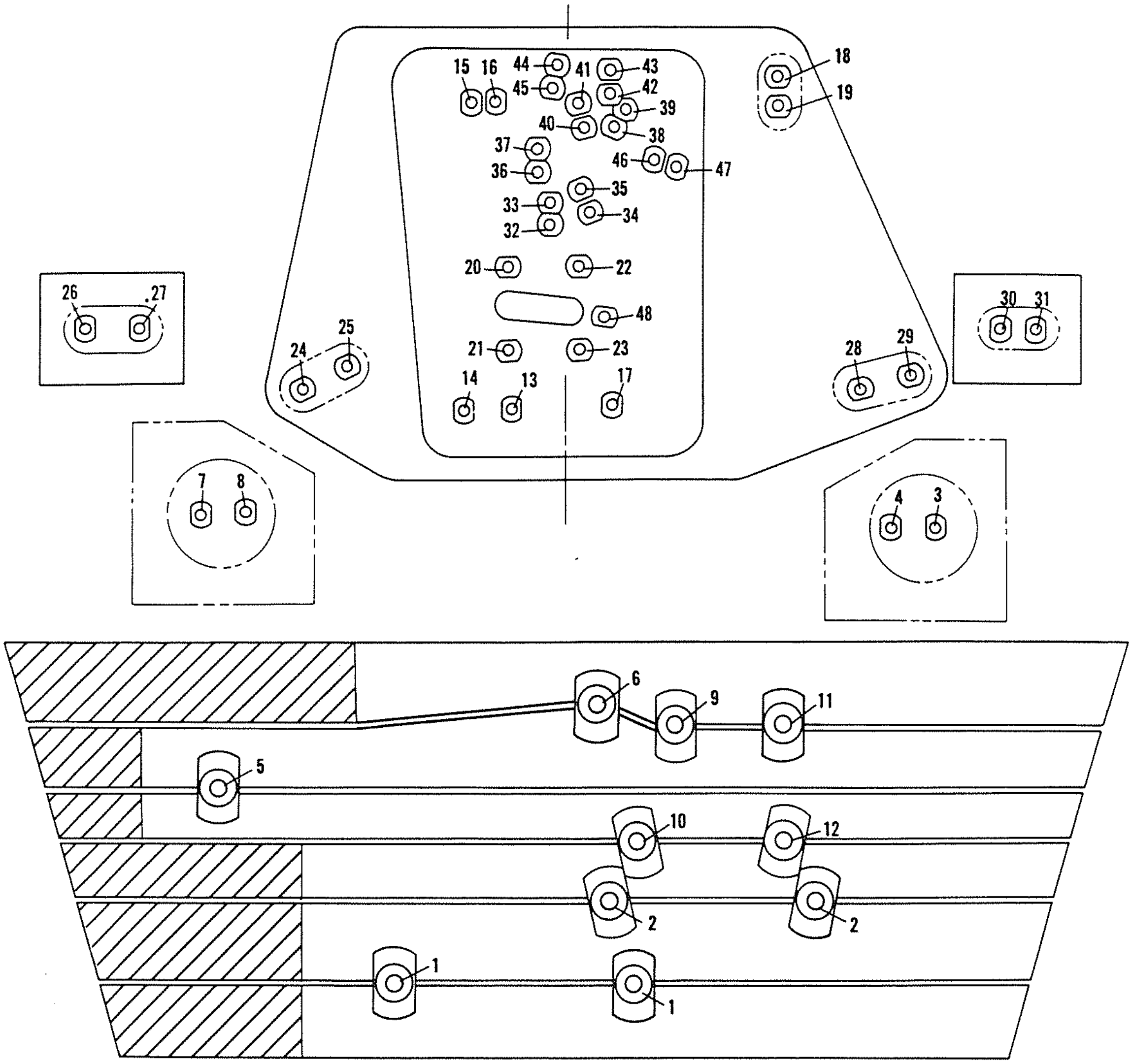
FAIRLEADS
STATION 97
LOOKING FORWARD



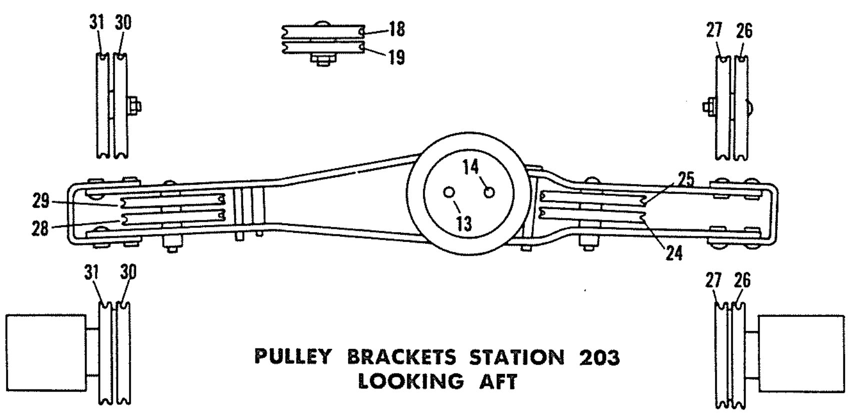
PULLEY BRACKETS
STATION 97
LOOKING AFT

1,173

Figure 2-10 (Sheet 4 of 10 Sheets). Cable Routing Master Drawings — Fairleads and Pulley Brackets, Station 97



FAIRLEADS, STATION 156 — LOOKING FORWARD

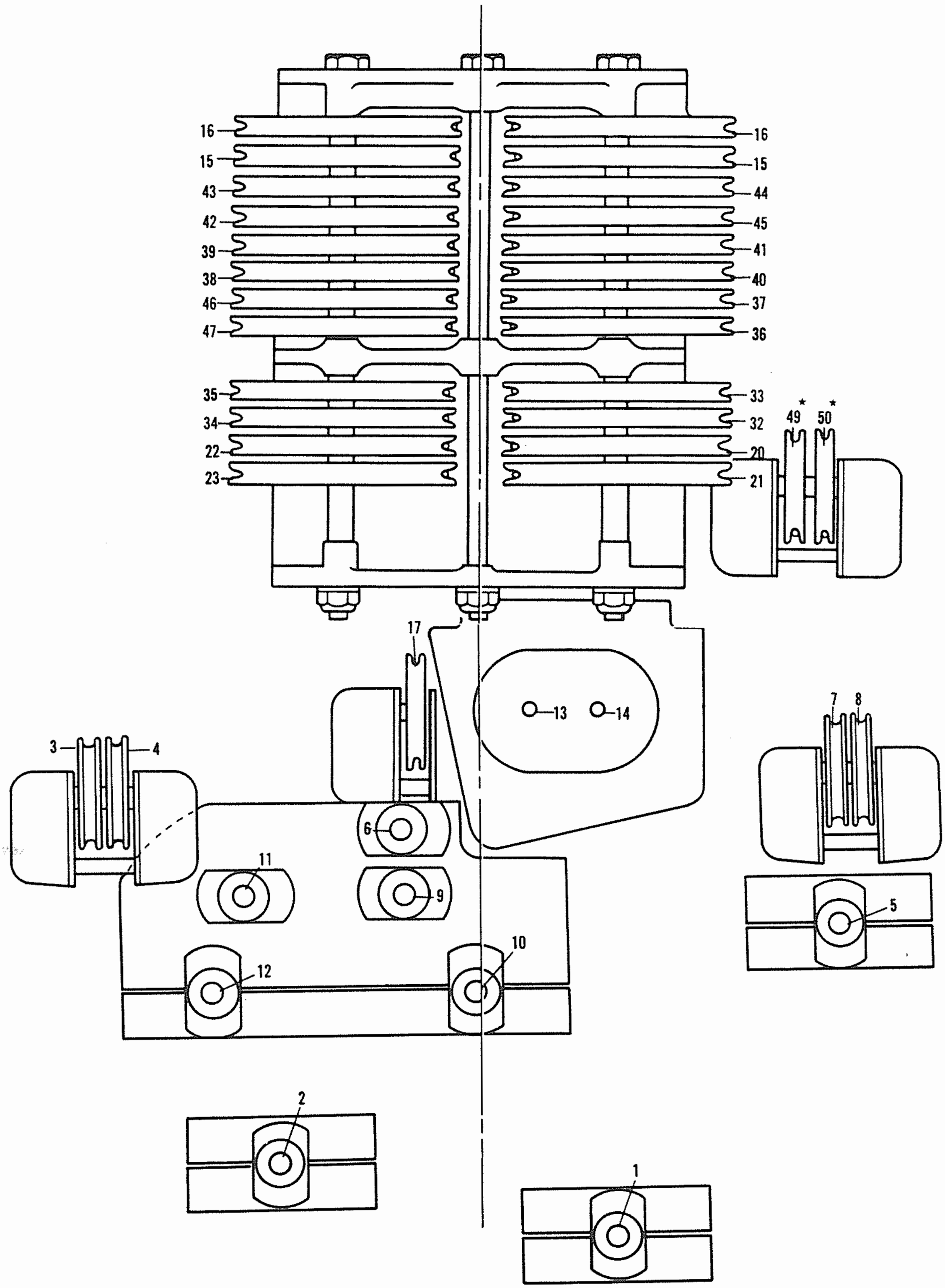


PULLEY BRACKETS STATION 203
LOOKING AFT

SHADED PORTION OF FAIRLEAD EFFECTIVE ON AIRCRAFT A, B, AND 1 THROUGH 28.

1,174

Figure 2-10 (Sheet 5 of 10 Sheets). Cable Routing Master Drawings—Fairleads, Station 156; Pulley Brackets, Station 203

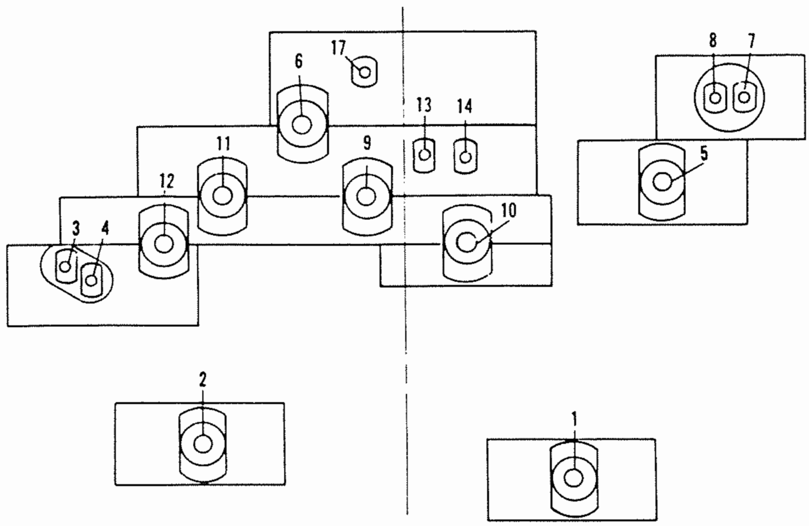


FAIRLEADS AND PULLEY BRACKETS
STATION 222 (FRONT SPAR)—LOOKING AFT

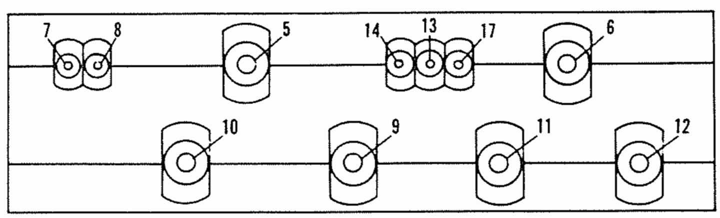
*Aircraft C, D, 9, and subsequent (cables only)

1,175

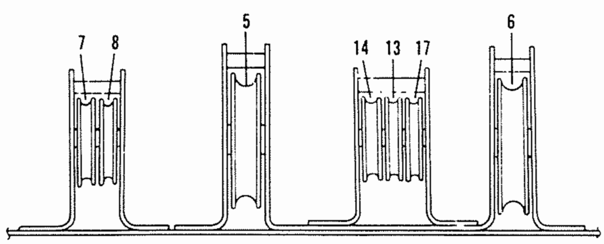
Figure 2-10 (Sheet 6 of 10 Sheets). Cable Routing Master Drawings — Fairleads and Pulley Brackets, Station 222



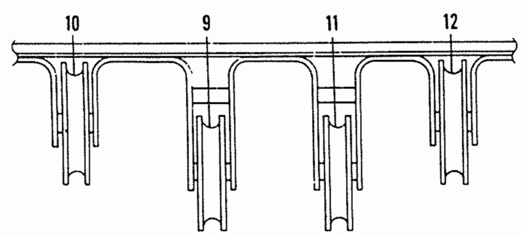
FAIRLEADS, STATION 258 (CENTER SPAR) — LOOKING AFT



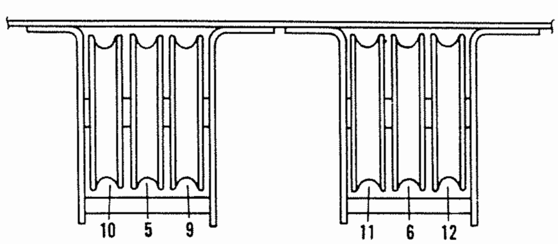
FAIRLEAD, STATION 324 -LOOKING FORWARD



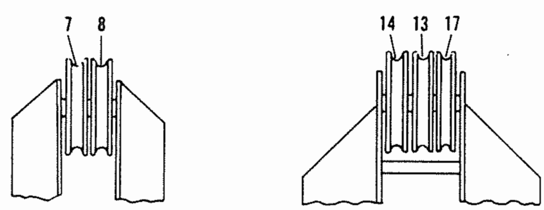
PULLEY BRACKETS, STATION 333
(LOOKING FORWARD)



PULLEY BRACKETS, STATION 351
(LOOKING FORWARD)



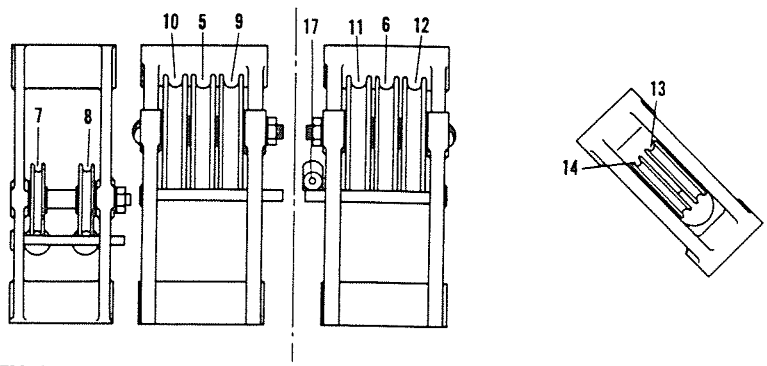
PULLEY BRACKETS
STATION 450 — LOOKING FORWARD



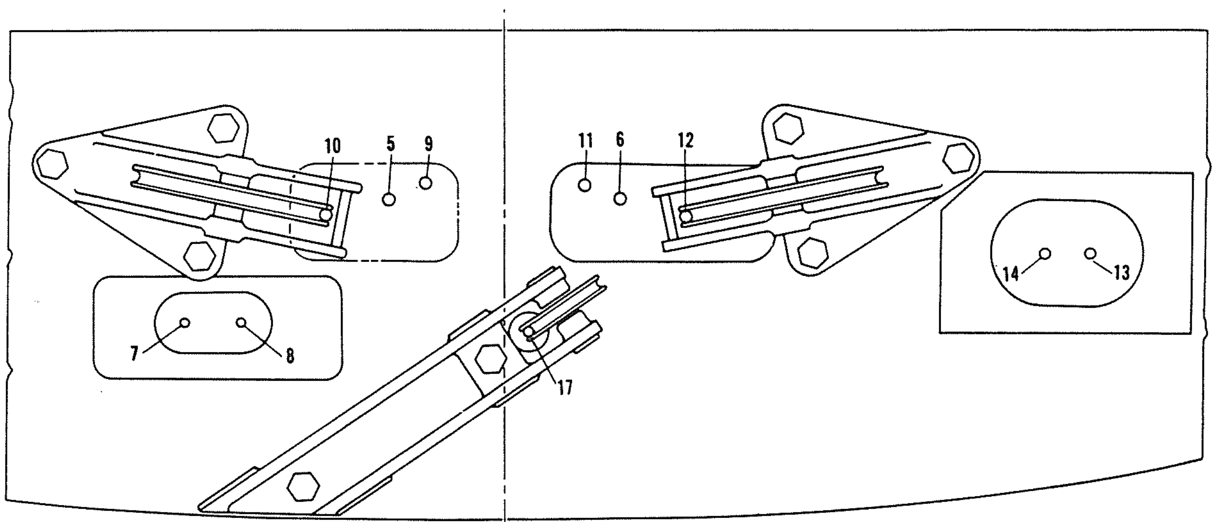
PULLEY BRACKETS
STATION 465 — LOOKING FORWARD

1,176

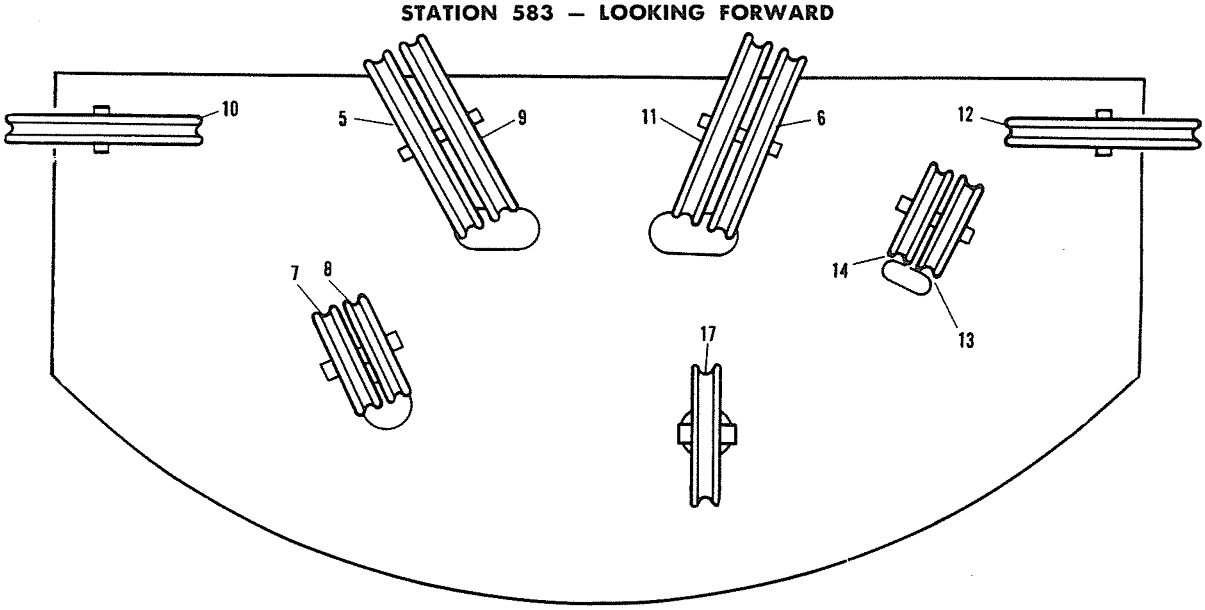
Figure 2-10 (Sheet 7 of 10 Sheets). Cable Routing Master Drawings — Fairleads, Stations 258 and 324; Pulley Brackets, Stations 333, 351, 450, and 465



PULLEY BRACKETS, STATION 538 — LOOKING FORWARD



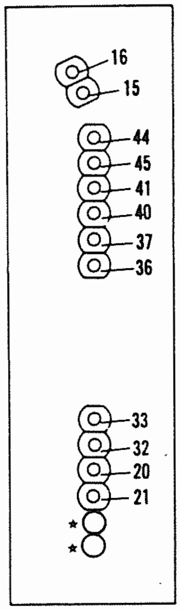
FAIRLEADS AND PULLEY BRACKETS
STATION 583 — LOOKING FORWARD



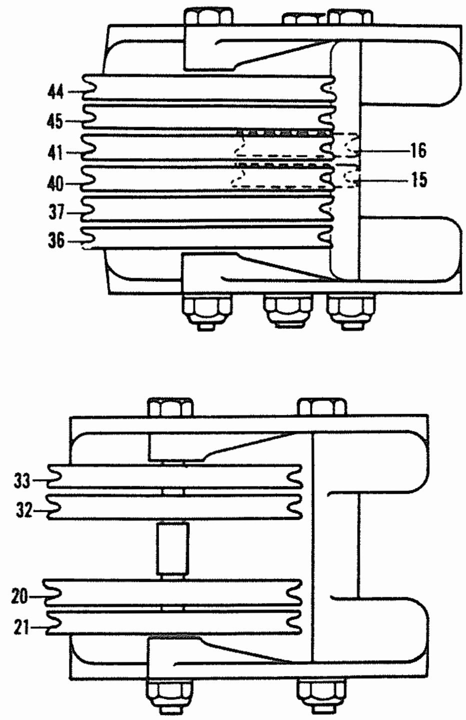
PULLEY BRACKETS, STATION 623 — LOOKING FORWARD

1,177

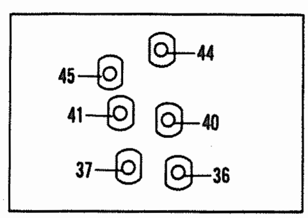
Figure 2-10 (Sheet 8 of 10 Sheets). Cable Routing Master Drawings — Pulley Brackets, Stations 538 and 623; Fairleads and Pulley Brackets, Station 583



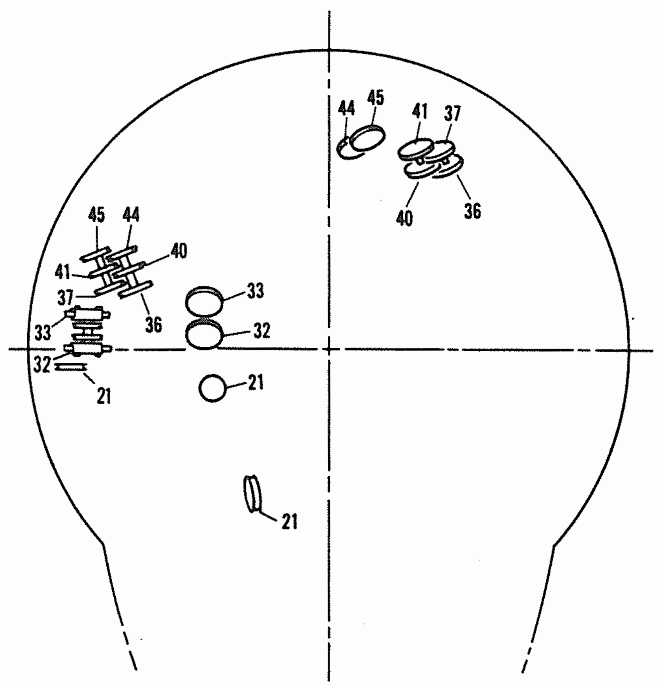
FAIRLEAD
LH WING STATION 45
LOOKING INBOARD



PULLEY BRACKETS
ON FRONT SPAR IN LH NACELLE
LOOKING AFT



FAIRLEAD
LH NACELLE
STATION 200 — LOOKING AFT

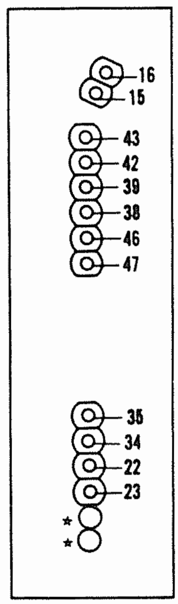


*Not used

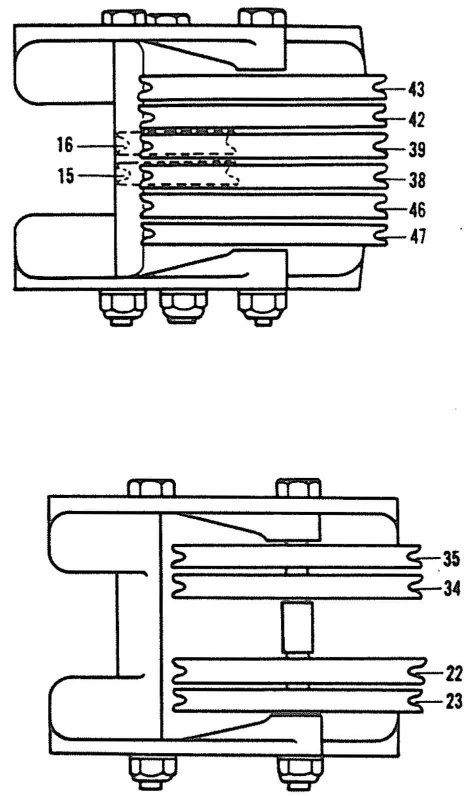
PULLEY BRACKETS
LH NACELLE FIREWALL
LOOKING AFT

1,178

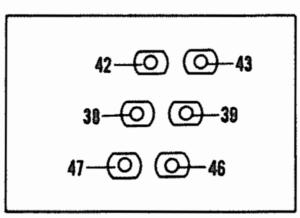
Figure 2-10 (Sheet 9 of 10 Sheets). Cable Routing Master Drawings — Fairleads, Left Wing Station 45 and Left Nacelle Station 200; Pulley Brackets, Left Nacelle Front Spar and Left Nacelle Firewall



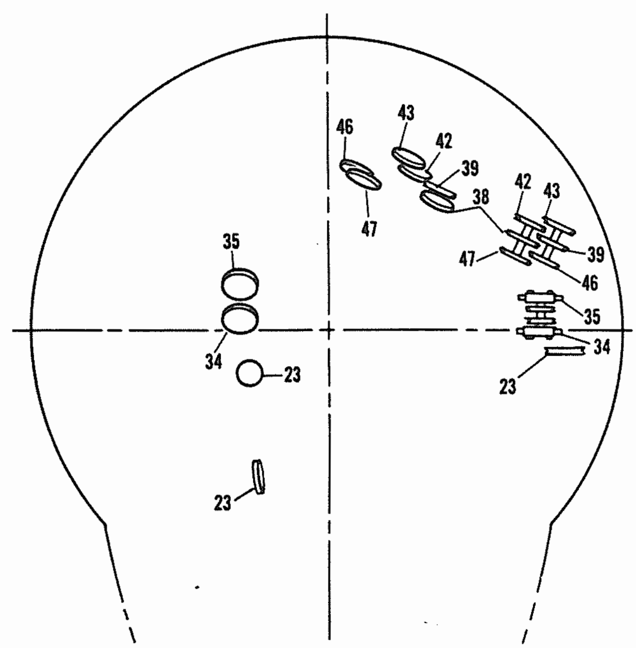
FAIRLEAD
RH WING STATION 45
LOOKING INBOARD



PULLEY BRACKETS
ON FRONT SPAR IN RH NACELLE
LOOKING AFT



FAIRLEAD
RH NACELLE
STATION 200 — LOOKING AFT



PULLEY BRACKETS
RH NACELLE FIREWALL
LOOKING AFT

*Not used

1,179

Figure 2-10 (Sheet 10 of 10 Sheets). Cable Routing Master Drawings — Fairleads, Right Wing Station 45 and Right Nacelle Station 200; Pulley Bracket, Right Nacelle Front Spar and Right Nacelle Firewall

(Continued from Page 87)

2-66. AILERONS. The two ailerons consist of aluminum-alloy frame construction covered with doped fabric. Each aileron is attached to the wing outer panel by seven hinge fittings and is controlled by a pushrod extending aft from each of the two aileron wing bellcranks. The left aileron contains a geared tab; the right aileron is provided with a geared tab that is also trim controlled.

2-67. REMOVAL OF AILERON.
(See figure 2-11.)

a. Clamp the aileron tab in neutral and remove the screws from the aileron hinge cut-out covers.

b. Disconnect the forward aileron tab control rod from the crank assembly on the aileron spar.

c. Detach the two aileron control rods from the ailerons by removing the aileron clevis bolts.

d. Remove the stop nuts from the eyebolt studs at each hinge station.

e. Remove the aileron by pulling it aft.

2-68. BALANCING OF AILERON.
(See figure 2-12.)

a. Remove the aileron from the aircraft (see paragraph 2-67).

b. Mount the aileron with the bottom surface up on the aileron balancing stands. Check the hinge bearings for binding and adjust the stands so that the surface swings freely.

Note

The location of the hinge line is extremely critical. The attaching eyebolts must be snug so that the hinge line (relative to the surface) is the same as the hinge line when the surface is installed on the aircraft.

c. Place a 1-pound weight on the aileron at station 178.57 (in line with the inboard hinge point). Move the weight along this station line until the aileron bottom surface is horizontal. Measure the distance from the tab hinge center line to the center of the weight. This distance must be between $7\frac{1}{16}$ and $17\frac{1}{16}$ inches for the aileron to be correctly balanced.

d. If the distance is more than $17\frac{1}{16}$ inches, add 0.63 pounds in the bay between stations 126 and 132. Repeat step c.

e. If the distance is still more than $17\frac{1}{16}$ inches, add 1.0 pounds in the bay between stations 132 and 139 and repeat step c.

f. If the distance is less than $7\frac{1}{16}$ inches, remove 0.63 pounds from the bay between stations 126 and 132. Repeat step c.

g. If the distance is still less than $7\frac{1}{16}$ inches, remove the total weight from the bay between stations 126 and 132 and repeat step c.

h. If the distance continues to be less than $7\frac{1}{16}$ inches, remove 0.8 pounds from the bay between stations 132 and 139 and again repeat step c.

Note

When the aileron has been thrown out of balance because of a weight change, adjust the counterbalance weight which is opposite or nearest to the change. When the aileron has been thrown out of balance due to repairs made in a number of places or when new covering, paint, dope, etc., is applied, balance the aileron in accordance with the steps outlined above.

2-69. AILERON HINGE WEAR TOLERANCES. Due to the close tolerance fits of the aileron hinge bolts and bearings, the maximum wear limit is 0.001 inch in excess of the manufacturing tolerance between the two parts.

2-70. INSTALLATION OF AILERON.
(See figure 2-11.)

a. Support the aileron in position and secure the eyebolts at each hinge station with the stop nuts.

b. Connect the two aileron control rods to the ailerons by installing the aileron clevis bolts.

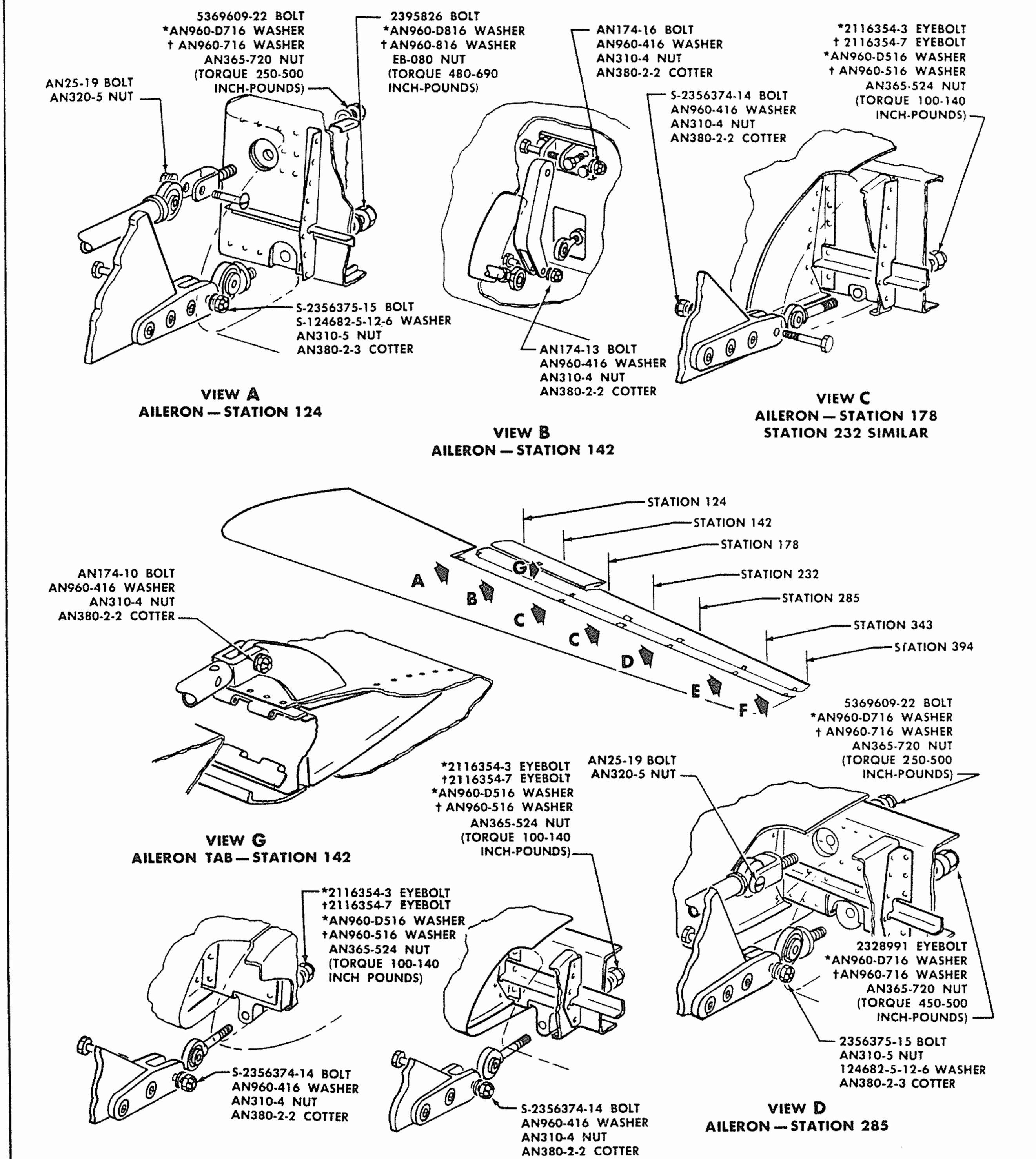
c. Connect the forward aileron tab control rod to the crank assembly on the aileron spar.

d. Replace the aileron hinge cut-out covers.

e. Adjust the aileron control system (see paragraph 2-90) and the aileron geared tab control system (see paragraph 2-99).

2-71. AILERON CONTROL SYSTEM. *(See figure 2-14.)* The two aileron control surfaces function simultaneously in opposite directions and are controlled by dual control wheels mounted on the control columns in the flight compartment. The control wheel motion is transmitted to the aileron cables by means of a sprocket and chain attached to the shaft of each wheel. The aileron cables are attached to the chain in the upper elbow of each column, routed through the shafts, over pulleys at the base of the column, and then aft through the fuselage to the aileron cross just aft of the rear spar in the center wing. The cables from each wheel are joined by links at station 175 in the fuselage. Two-way cable systems, extending outboard through the wings, transmit movement of the aileron cross to the wing bellcranks. The bellcranks are linked to the ailerons by means of connecting rods. Stops are located in the control columns and in the inboard wing bellcrank brackets. An automatic pilot servo unit is connected to the left wing DOWN cable (left aileron UP) at station 180.

(Continued on Page 109)



*Aircraft A, B, and 1 through 8
†Aircraft C, D, 9, and subsequent

Notes:

1. For torque values other than noted, see paragraph 2-453.
2. Reference: Douglas Drawings 5367131, 5369608, 5369609 and 5369729, and 5371403.

1,180

Figure 2-11. Installation of Aileron and Aileron Tab

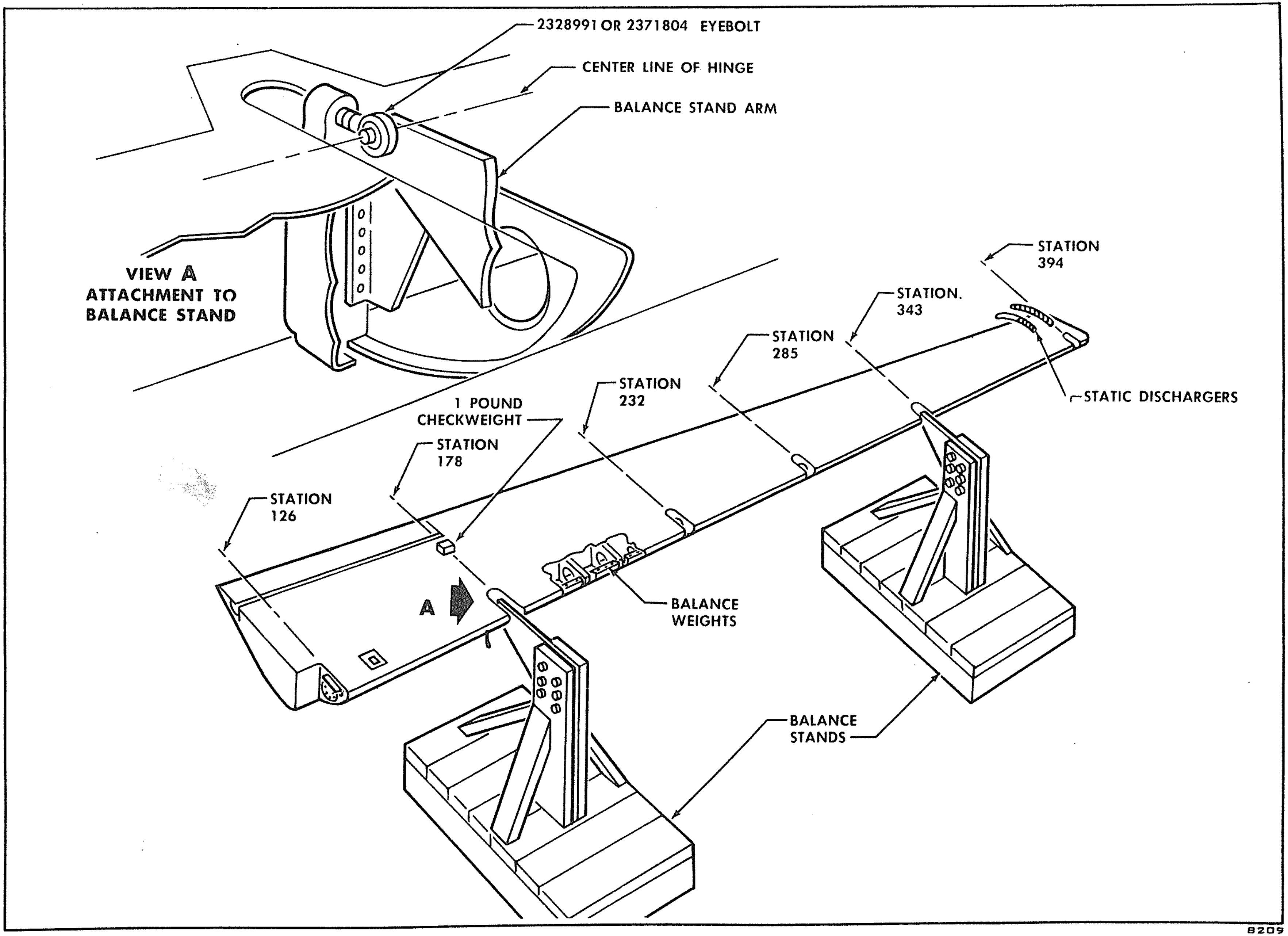


Figure 2-12. Balancing Aileron

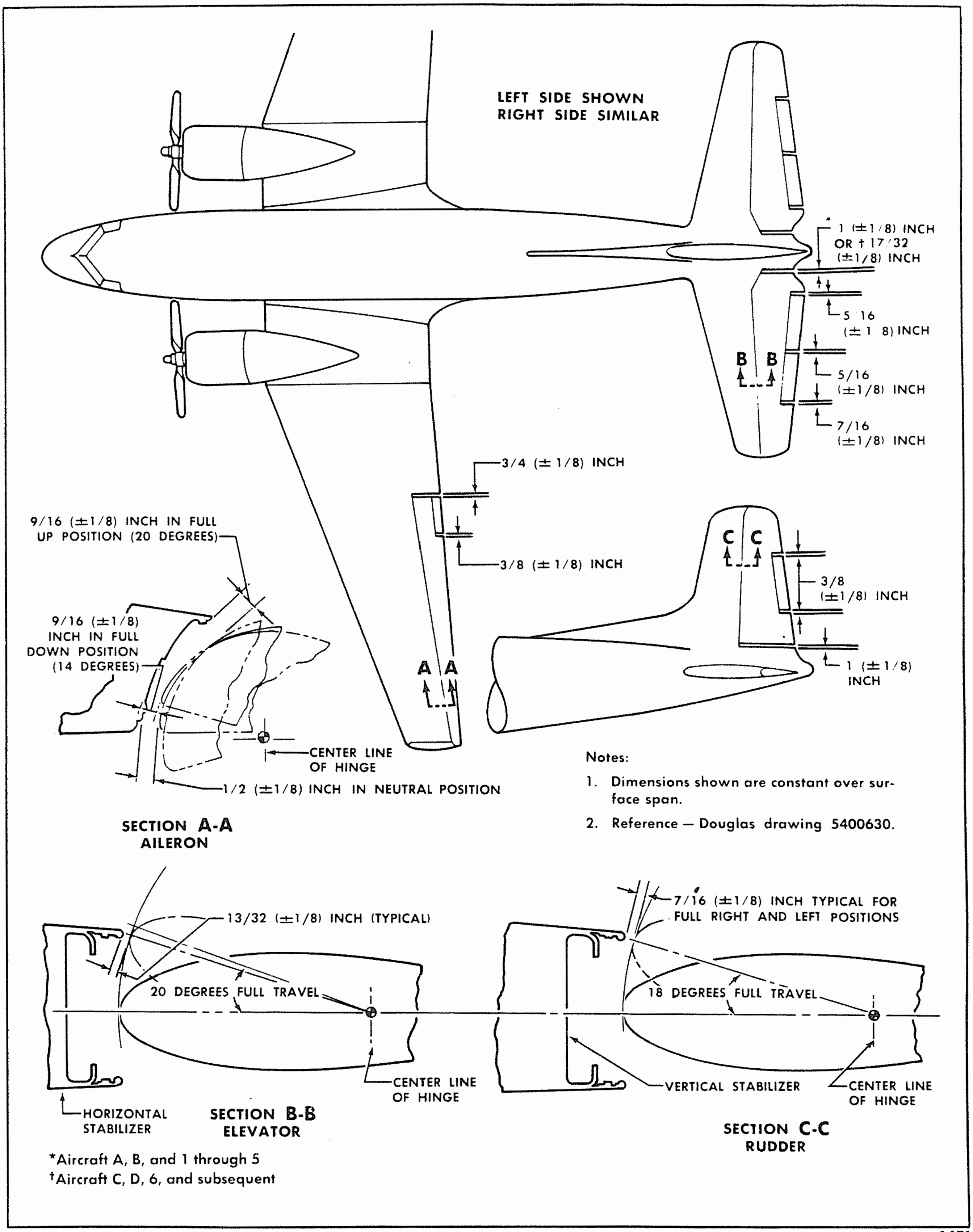


Figure 2-13 (Sheet 1 of 2 Sheets). Control Surface Clearance Chart

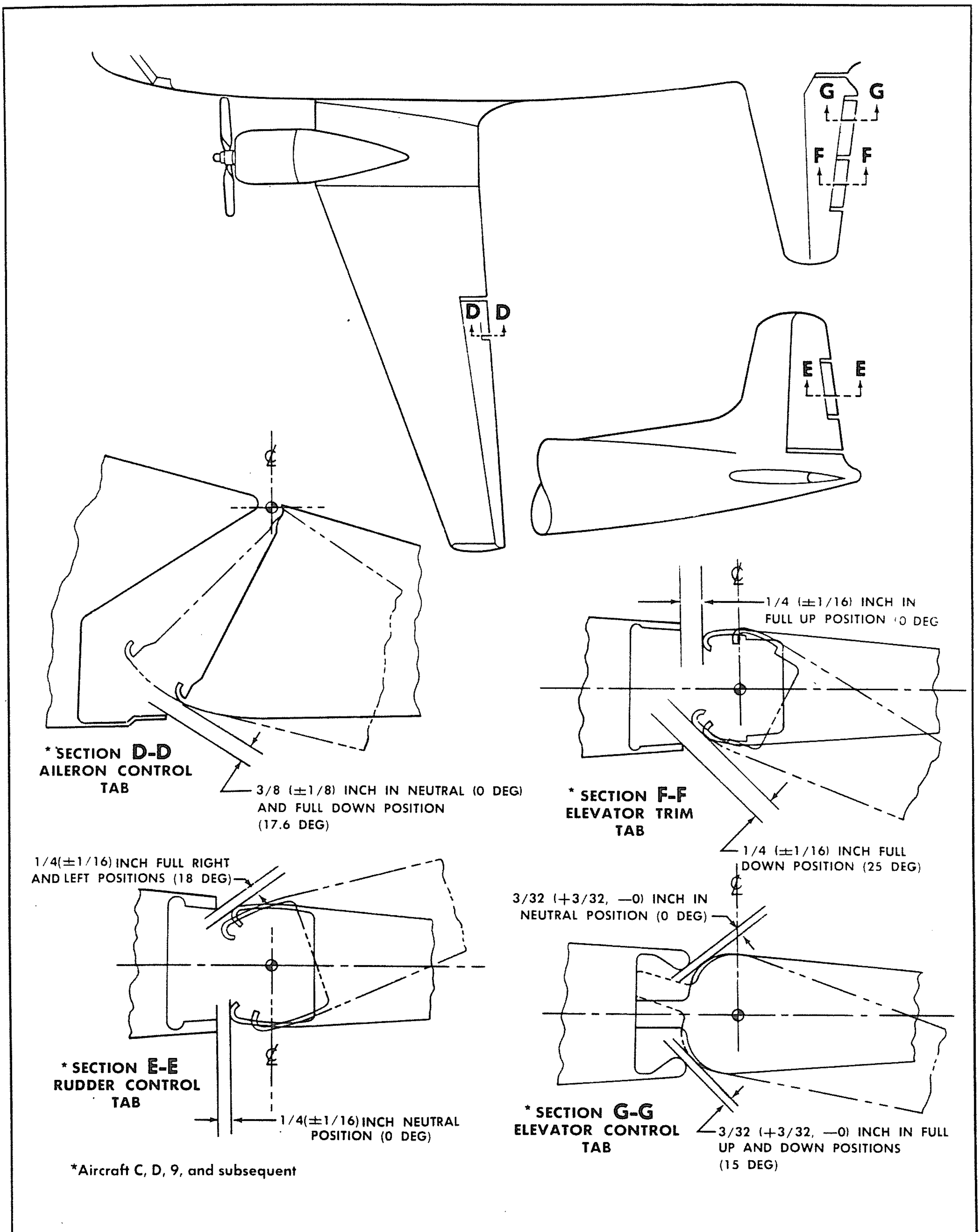


Figure 2-13 (Sheet 2 of 2 Sheets). Control Surface Clearance Chart

Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 5.
2. For cable assembly details, see Sheets 6 and 7.
3. Tension cables in accordance with the Cable Rigging Tension Chart, Figure 2 9
4. Tighten all nuts and bolts in accordance with torque values shown in Torque Value Chart, paragraph 2 453
5. Reference—Douglas drawings 5371403, 5372511, 5372516, 5372517, 5372518, and 5398929.

A—Fairlead and Pulley Brackets—Station 37,	see Sheet 2
B—Pulley Brackets—Station 97,	see Sheet 2
C—Fairleads—Station 156,	see Sheet 2
D—Automatic Pilot Rigging,	see Sheet 1
E—Fairleads and Pulley Brackets—Station 222,	see Sheet 3
F—Fairleads—Station 258,	see Sheet 3
G—Aileron Cross—Rear Spar,	see Sheet 1
H—Inboard Wing Bellcrank,	see Sheet 4
I—Geared Tab Mechanism,	see Sheet 4
J—Outboard Wing Bellcrank,	see Sheet 4

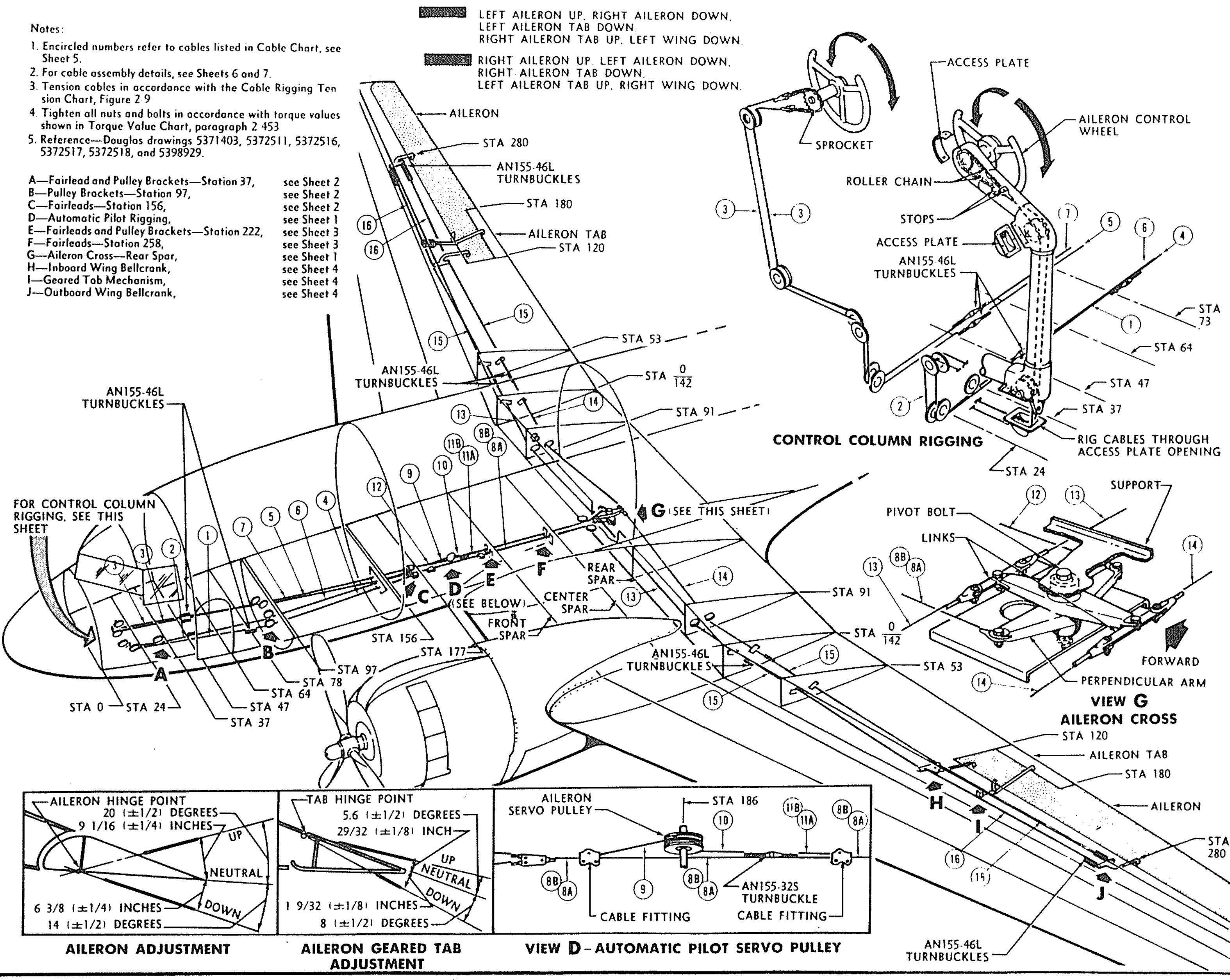
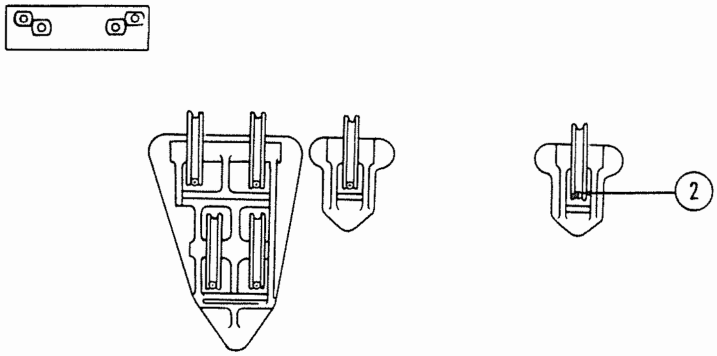
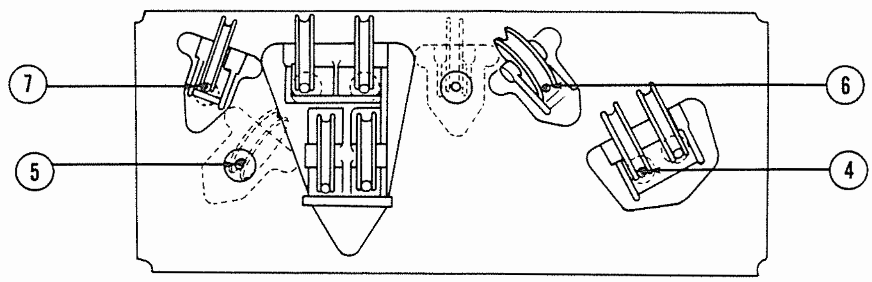


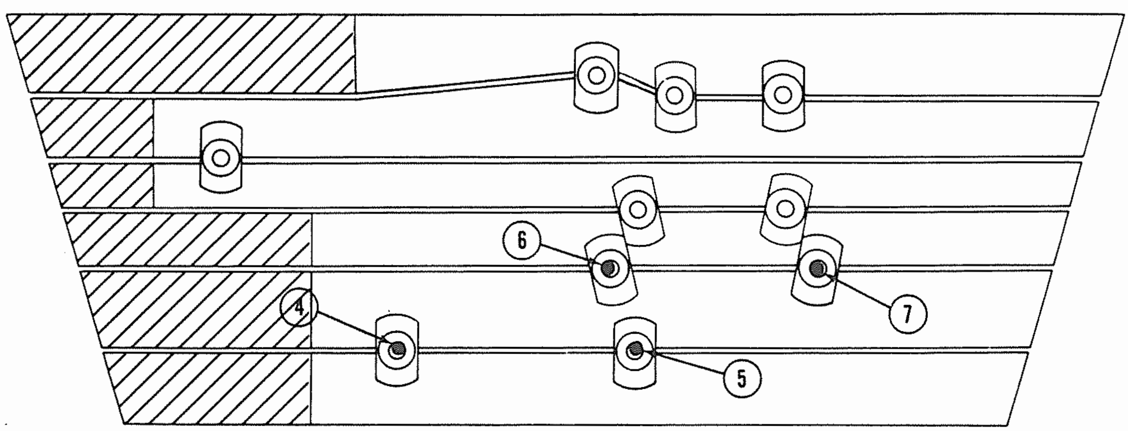
Figure 2-14 (Sheet 1 of 7 Sheets). Aileron and Aileron Geared Tab Control System — Key Drawing



VIEW A — FAIRLEAD AND PULLEY BRACKETS
AT STATION 37 — LOOKING AFT



VIEW B — PULLEY BRACKETS
AT STATION 97 — LOOKING AFT



VIEW C — FAIRLEAD AT
STATION 156 — LOOKING FORWARD

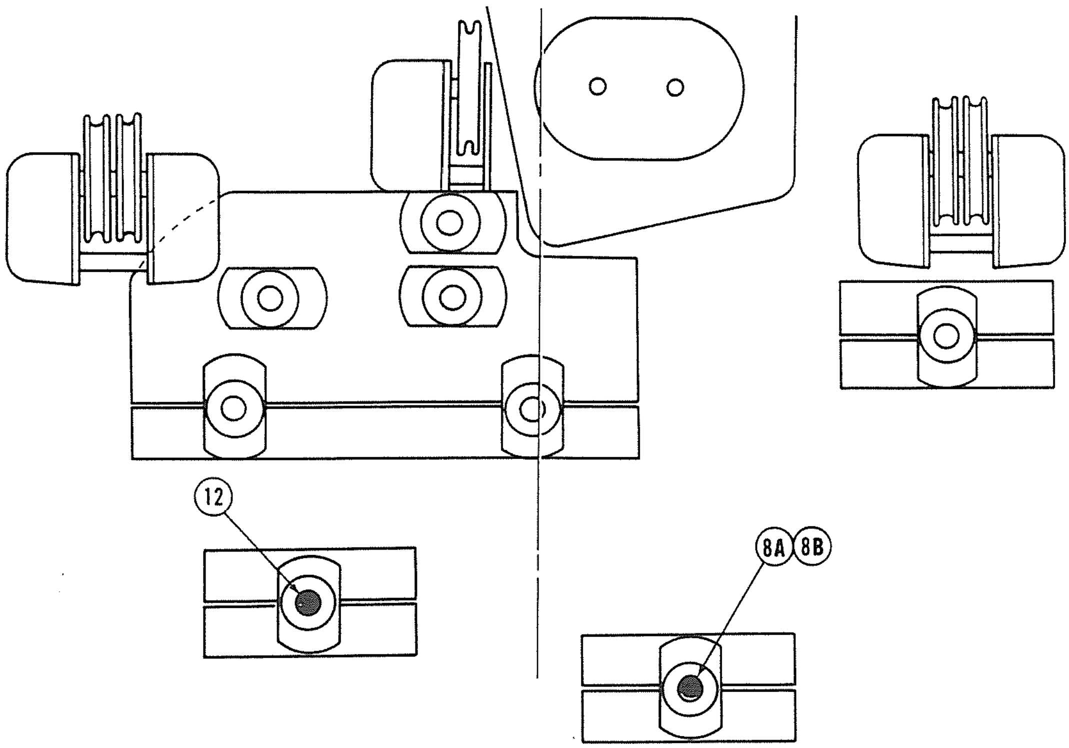
LEFT AILERON UP, RIGHT AILERON DOWN,
LEFT AILERON TAB DOWN,
RIGHT AILERON TAB UP, LEFT WING DOWN

RIGHT AILERON UP, LEFT AILERON DOWN,
RIGHT AILERON TAB DOWN,
LEFT AILERON TAB UP, RIGHT WING DOWN

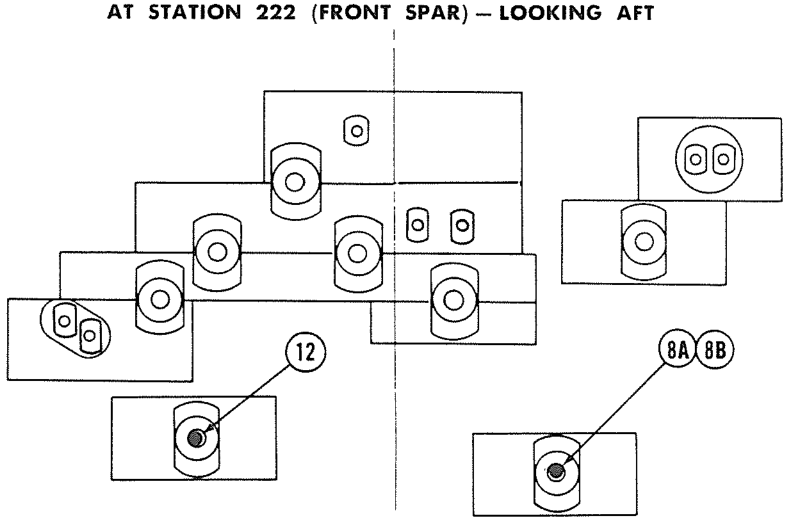
SHADED PORTION OF FAIRLEAD EFFECTIVE
ON AIRCRAFT A, B, AND 1 THROUGH 28.

1,183

Figure 2-14 (Sheet 2 of 7 Sheets). Aileron and Aileron Geared Tab Control System — Fairlead and Pulley Brackets, Station 37; Pulley Brackets, Station 97; Fairleads, Station 156



VIEW E — FAIRLEADS AND PULLEY BRACKETS
AT STATION 222 (FRONT SPAR) — LOOKING AFT



VIEW F FAIRLEADS AT STATION 258
(CENTER SPAR) LOOKING AFT

LEFT AILERON UP, RIGHT AILERON DOWN,
LEFT AILERON TAB DOWN,
RIGHT AILERON TAB UP, LEFT WING DOWN.

RIGHT AILERON UP, LEFT AILERON DOWN,
RIGHT AILERON TAB DOWN,
LEFT AILERON TAB UP, RIGHT WING DOWN.

1,184

Figure 2-14 (Sheet 3 of 7 Sheets). Aileron and Aileron Geared Tab Control System — Fairleads and Pulley Brackets, Station 222; Fairleads, Station 258

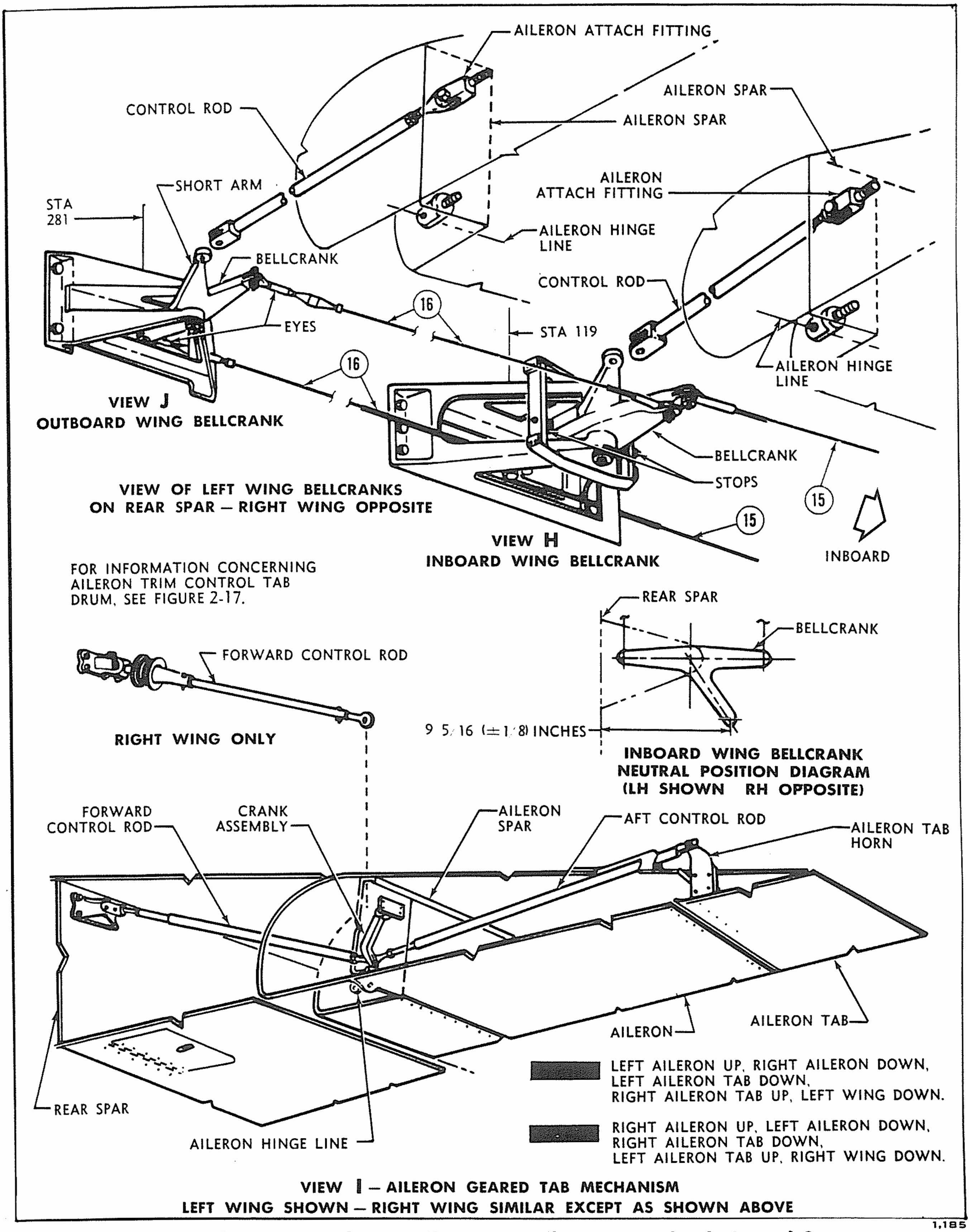


Figure 2-14 (Sheet 4 of 7 Sheets). Aileron and Aileron Geared Tab Control System — Wing Bellcranks, Geared Tab Mechanism

AILERON AND AILERON GEARED TAB CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	No. REQD.	TYPE	CABLE LENGTH (L_1) (L_2)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)	(5)
1	2135993-4 LH Ail. Up	1	A	123 7/8	3/16 dia 7 x 19 flex	2135994	AN669L6 RH			
2	2135993-6 LH Ail. Down	1	B	93 3/8	3/16 dia 7 x 19 flex	2135994	AN669S6 LH			
3	2135993-2 LH Ail. Up LH Ail. Down	2	A	95 7/8	3/16 dia 7 x 19 flex	2135994	AN669L6 RH			
4	4398907-509 LH Ail. Up	1	C	86	3/16 dia 7 x 19 flex	AN669L6 LH	AN668-6			
5	4398907-505 LH Ail. Up	1	C	115 1/2	3/16 dia 7 x 19 flex	AN669L6 LH	AN668-6			
6	4398907-507 LH Ail. Down	1	D	122 1/2	3/16 dia 7 x 19 flex	AN669L6 RH	AN668-6			
7	4398907-503 LH Ail. Down	1	C	113 3/4	3/16 dia 7 x 19 flex	AN669L6 LH	AN668-6			
*8A	4398907-513 LH Ail. Up	1	E	(L_1) 127 1/4 (L_2) 6 1/16 (L_3) 22 3/4	3/16 dia 7 x 19 flex	AN668-6	AN663-6	AN663-6	AN663-6	
†8B	4398907-559 LH Ail. Up	1	E	(L_1) 127 1/4 (L_2) 6 1/16 (L_3) 33 3/4	3/16 dia 7 x 19 flex	AN668-6	AN663-6	AN663-6	AN668-6	
9	4398907-519 Ail. Servo Bus	1	F	11 1/4	1/8 dia 7 x 19 flex	2391676	AN664-4	AN664-4		
10	4398907-525 Ail. Servo Jumper	1	G	14 1/2	1/8 dia 7 x 19 flex	AN664-4	AN669S4 LH			
*11A	4398907-523 Ail. Servo Jumper	1	H	6 3/4	1/8 dia 7 x 19 flex	AN669S4 RH	AN664-4	2391676		
†11B	4398907-561 Ail. Servo Jumper	1	H	17 3/4	1/8 dia 7 x 19 flex	AN669S4 RH	AN664-4	2391676		
12	4398907-511 LH Ail. Down	1	I	125 1/4	3/16 dia 7 x 19 flex	AN668-6	AN668-6			
13	3391412-525 LH Ail. Up RH Ail. Up	2	J	167 5/8	3/16 dia 7 x 19 flex	AN667-6	AN669L6 RH			
14	3391412-527 LH Ail. Down RH Ail. Down	2	J	166 5/8	3/16 dia 7 x 19 flex	AN667-6	AN669L6 RH			
15	3391412-529 LH & RH Ail. Up LH & RH Ail. Down	4	C	91 1/4	3/16 dia 7 x 19 flex	AN669L6 LH	AN668-6			
16	3391412-531 LH & RH Ail. Up LH & RH Ail. Down	4	K	159 1/4	3/16 dia 7 x 19 flex	AN668-6	AN669L6 LH			

*Aircraft A, B, and 1 through 52.
†Aircraft C, D, 53, and subsequent.

1,186

Figure 2-14 (Sheet 5 of 7 Sheets). Aileron and Aileron Geared Tab Control System — Cable Chart

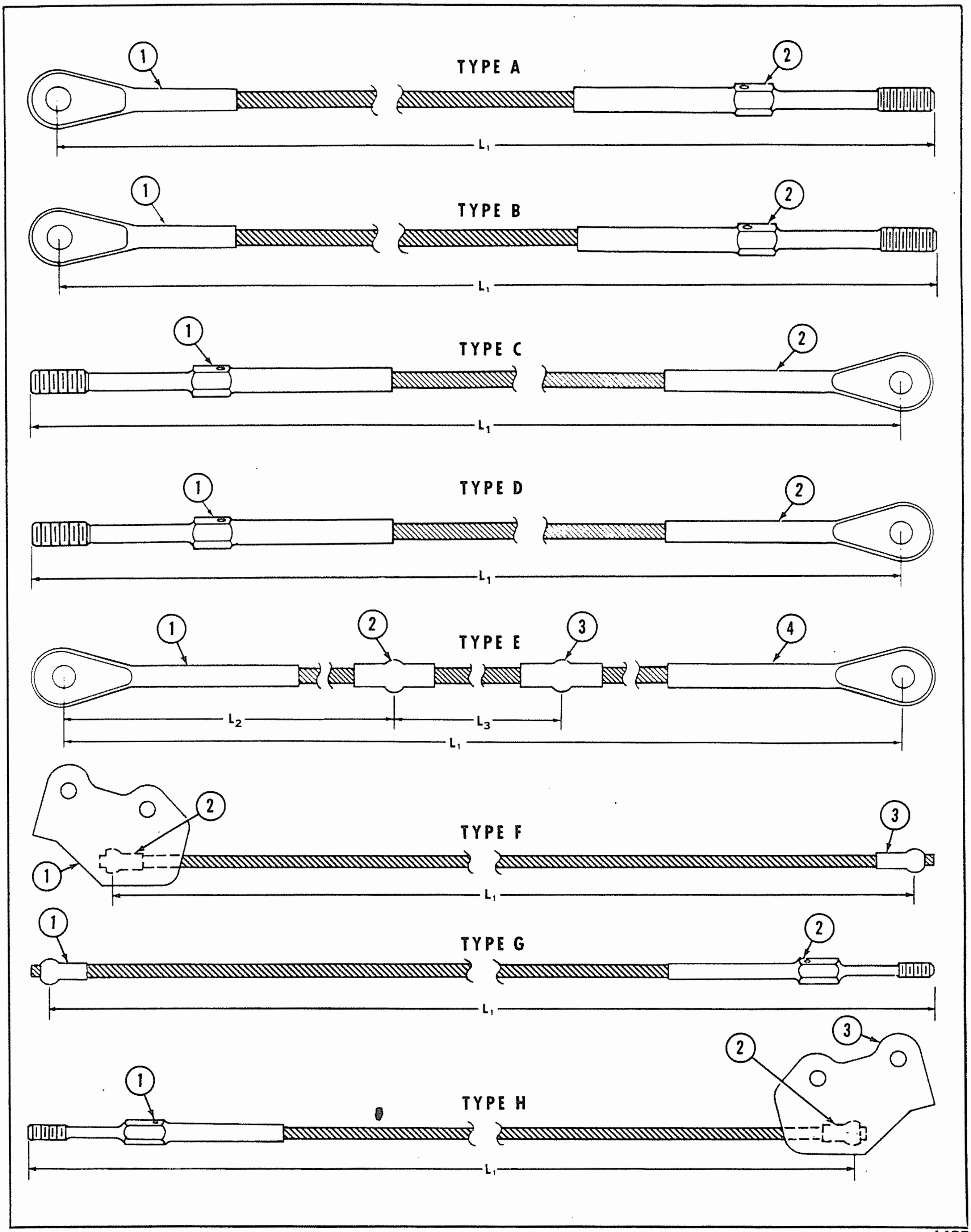


Figure 2-14 (Sheet 6 of 7 Sheets). Aileron and Aileron Geared Tab Control System — Cable Assemblies

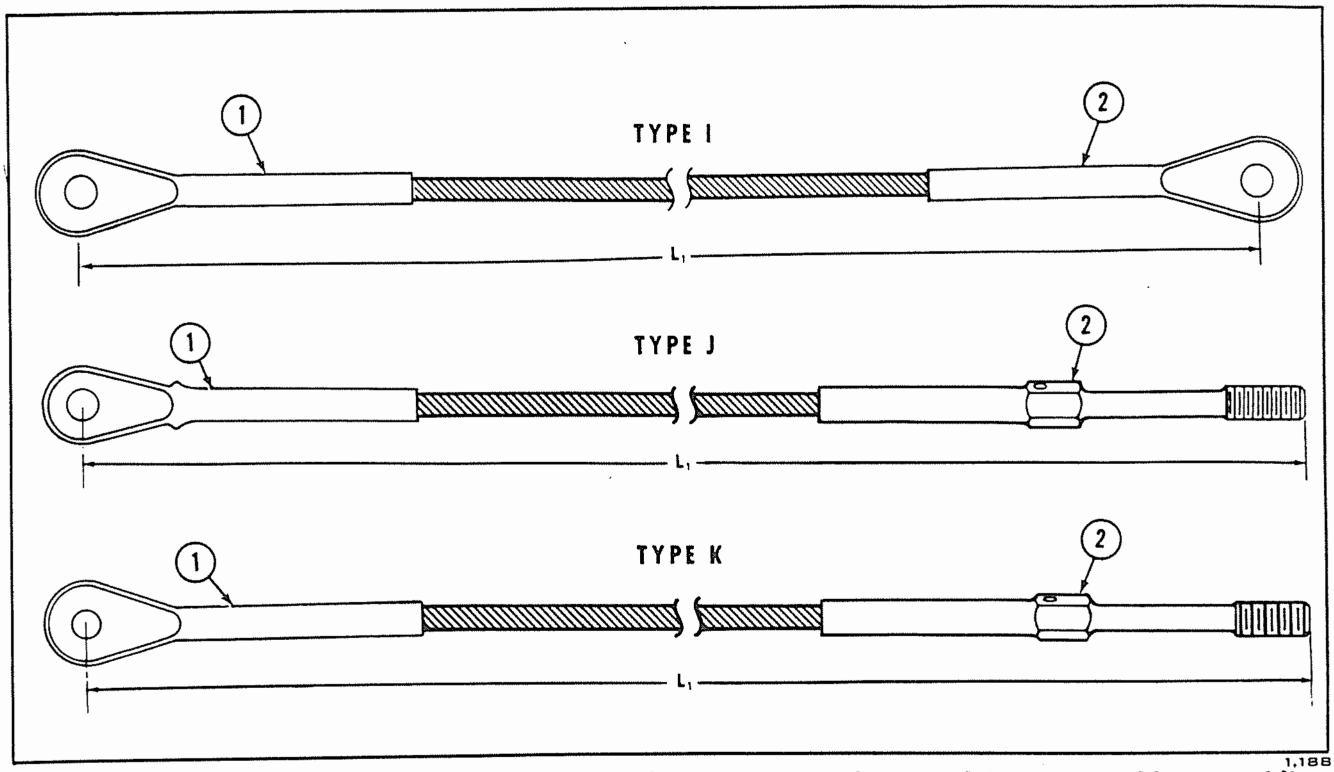


Figure 2-14 (Sheet 7 of 7 Sheets). Aileron and Aileron Geared Tab Control System — Cable Assemblies

(Continued from Page 98)

2-72. AILERON CONTROL WHEEL ASSEMBLIES AND ROLLER CHAINS. *(See figure 2-14.)* An aileron control wheel is provided on each of the control columns in the flight compartment; each wheel rotates a shaft containing a sprocket at the forward end. A roller chain, to which two stops are attached, passes over the sprocket and connects to the forward ends of the aileron control cables, thus transmitting the movement of the control wheel to the system. Travel of the control wheel is limited by the roller chain stops striking a slotted plate in the control column elbow.

2-73. REMOVAL OF AILERON CONTROL WHEEL ASSEMBLY AND ROLLER CHAIN.

a. To remove the aileron control wheel assembly, place the ailerons in neutral and install surface locks on the ailerons. Remove the access door on the under side of the aircraft and disconnect aileron cables at approximately station 64. Remove the plate screwed to the top of the control column. Cut the safety wire and release the six bolts from the cap through which the wheel shaft passes. Lift the roller chain until it is free of the sprocket and remove the complete wheel and sprocket assembly by pulling aft on the wheel.

CAUTION

Do not allow the chain and cable to drop down in the central column.

b. To remove the roller chain, lift the chain until the stops strike the stop plate, and give the chain a quarter turn so that the stops can be drawn through the stop plate opening. Fasten the cables to the control head so that they will not drop down into the column when they are disconnected from the chain. Disconnect the chain from the forward ends of the control cables. Lift the chain out of the control column head.

2-74. MINOR REPAIR AND REPLACEMENT OF AILERON CONTROL WHEEL ASSEMBLIES AND ROLLER CHAINS. Replace worn or damaged sprockets, bearings, pulleys, or roller chains.

2-75. INSTALLATION OF AILERON CONTROL WHEEL ASSEMBLY AND ROLLER CHAIN.

a. Connect the roller chains to the forward ends of the control cables and lower the cables and chains into the control columns, giving the chains a quarter turn in order that the stops will pass through the stop plate opening.

CAUTION

Do not allow the chain and cable to drop down into the control column.

b. Install the complete wheel and sprocket assemblies, keeping the open sections of the control wheels in a straight line across the top. A board attached across the wheels may be used to lock the wheels in

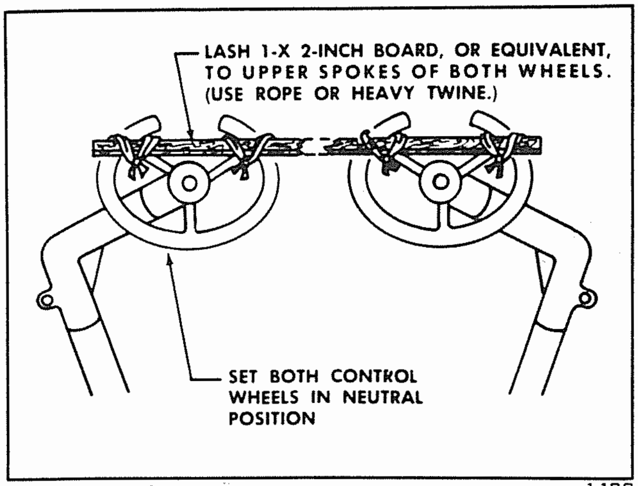


Figure 2-15. Locking Aileron Control Wheels in Neutral

this position *(see figure 2-15).* Fit the roller chains onto the sprocket teeth to allow rotation of the control wheels, approximately 180 degrees in either direction from neutral, before the roller chain stops contact the stop plates.

c. Install the six bolts in the caps through which the wheel shafts pass. Lock with safety wire.

d. Install the plates on the top of the control columns.

e. Connect the aileron control cables at approximately station 64 and tension the cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

f. Remove the surface locks and adjust the aileron control system (see paragraph 2-90). Safety the turnbuckles.

g. Install the access door on the under side of the aircraft.

2-76. AILERON AUTOMATIC PILOT PULLEY. The aileron automatic pilot pulley is located just to the left of the center line of the aircraft below the floor at station 186. For further information of the automatic pilot pulley, see paragraph 6-82.

2-77. AILERON CROSS. *(See figure 2-16.)* The aileron cross, located on the center line of the aircraft just aft of the rear spar, consists of two arms in the shape of a cross, one perpendicular to the aircraft center line for attachment of the forward control cables, the other parallel with the center line for attachment of the control cables leading to the bellcranks in the outer wing panels. The cross is actuated by the forward aileron control cables and transmits this movement through two-way cable systems to the aileron wing bellcranks.

2-78. REMOVAL OF AILERON CROSS.

a. Position the ailerons in neutral and clamp surface locks on the ailerons.

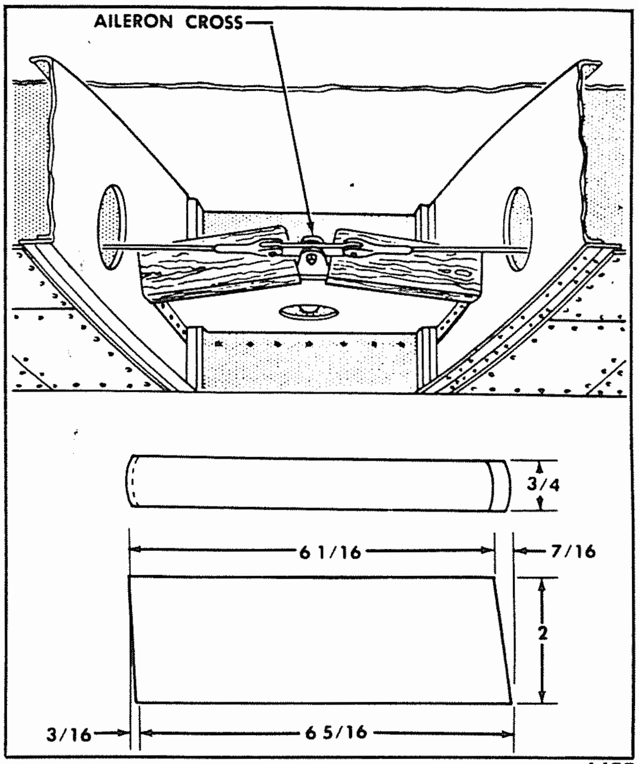


Figure 2-16. Locking Aileron Cross in Neutral

b. Remove the access door on the under side of the fuselage and loosen the four aileron cable turnbuckles at approximately station 64.

c. Remove the access door on the under side of the fuselage and loosen the turnbuckle connecting the aileron servo jumper cables at approximately station 193.

d. Open the aileron cross access door and disconnect the forward aileron control cables from the perpendicular arm of the aileron cross.

e. Open the access door and loosen the turnbuckles in the aileron cross-to-inboard bellcrank cables at station 27 in the outer wing panels.

f. Disconnect the cross-to-bellcrank cables from the links at the aileron cross.

g. Remove the aileron cross pivot bolt.

h. Pull the aileron cross aft and out of the aircraft.

2-79. MINOR REPAIR AND REPLACEMENT OF AILERON CROSS. Replace worn or broken bearings or links.

2-80. INSTALLATION OF AILERON CROSS.

a. Place the aileron cross in position with the ends of the longer arms pointing forward and aft, and install the aileron cross pivot bolt.

b. Connect the cross-to-bellcrank cables to the links at the aileron cross, and the forward aileron control cables to the perpendicular arm of the cross.

c. Secure the aileron cross in the neutral position. This may be accomplished by inserting two blocks of wood between the aft end of the horn arm and the corners of the lower web *(see figure 2-16).*

d. Tension the aileron forward and cross-to-bellcrank cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

e. Remove the surface locks.

f. Adjust the aileron control system (see paragraph 2-90).

g. Safety the turnbuckles.

h. Install the access doors.

2-81. AILERON WING BELLCRANKS. *(See figure 2-14.)* Two aileron wing bellcranks are provided, just aft of the rear spar, in each wing outer panel; one at station 118 and the other at station 280. The bellcranks are actuated by the control cables, which connect the inboard cranks with the aileron cross and transmit movement to the ailerons by means of control rods. A separate two-way cable system in each wing connects the inboard and outboard bellcranks, causing them to operate simultaneously.

2-82. MINOR REPAIR AND REPLACEMENT OF AILERON WING BELLCRANKS. Replace any worn or broken bearings.

2-83. REMOVAL OF INBOARD AILERON WING BELLCRANK.

a. Position the ailerons in neutral and clamp surface locks on the ailerons.

b. Open access door and loosen the turnbuckles in the cross-to-bellcrank cables at station 27 in the wing outer panels.

c. Open the access door and loosen the turnbuckles in the cables, connecting the two bellcranks at station 273, just inboard of the outboard bellcrank.

d. Disconnect the four cables from the inboard bellcrank.

e. Remove the aileron control rod from the short arm of the bellcrank.

f. Remove the bellcrank pivot bolt from the bracket and remove the inboard aileron wing bellcrank.

2-84. INSTALLATION OF INBOARD AILERON WING BELLCRANK.

a. Place the inboard aileron wing bellcrank in position in the bracket and install the bellcrank pivot bolt.

b. Connect the aileron control rod to the short arm of the bellcrank.

c. Attach the four cables to the bellcrank.

d. Tension the control cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

e. Remove the surface locks.

f. Adjust the aileron control system (see paragraph 2-90).

g. Safety the turnbuckles.

h. Install the access doors.

2-85. REMOVAL OF OUTBOARD AILERON WING BELLCRANK.

a. Position the ailerons in neutral and clamp surface locks on the ailerons.

b. Open the access door and loosen the turnbuckles in the cables, connecting the two bellcranks at station 273, just inboard of the outboard crank.

c. Disconnect the two cables from the outboard bellcrank.

d. Remove the aileron control rod from the short arm of the bellcrank.

e. Remove the bellcrank pivot bolt from the bracket and remove the outboard aileron wing bellcrank.

2-86. INSTALLATION OF OUTBOARD AILERON WING BELLCRANK.

a. Place the outboard aileron wing bellcrank in position and install the bellcrank pivot bolt.

b. Connect the aileron control rod to the short arm of the bellcrank.

c. Connect the two control cables to the bellcrank.

d. Tension the control cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

e. Remove the surface locks.

f. Adjust the aileron control system (see paragraph 2-90).

g. Safety the turnbuckles and install the access doors.

2-87. AILERON CONTROL CABLES. *(See figure 2-14.)* Two 2-way cable systems, originating in the control columns, extend aft through the fuselage to station 177, forward of the automatic pilot servo, where they are interconnected, and extend from there to the aileron cross as one 2-way system. Separate two-way cables extend from the aileron cross, outboard through the wings, to the inboard and outboard aileron wing bellcranks.

2-88. REMOVAL OF AILERON CONTROL CABLES.

a. Place the ailerons in neutral and install surface locks on the ailerons.

b. Remove surface controls access doors on the under side of the aircraft. Disconnect the aileron control cables at approximately station 64. Thread the eight cable ends for removal.

c. Remove the pulleys from the brackets, forward of the turnbuckles, at station 24.

d. Remove the pulleys, located beneath the cover plates, from the two elbows of the control columns.

e. Remove necessary guard pins.

f. Remove the plates screwed to the tops of the control columns.

g. Cut the safety wires and release the bolts from the caps through which the wheel shafts pass.

h. Lift each roller chain until it is free of the sprocket and remove the complete wheel and sprocket assembly by pulling aft on the wheel.

CAUTION

Do not allow the chain and cable to drop down in the control column.

i. Lift each chain until the stops strike the stop plate, give the chain a quarter turn so that the stops can be drawn through the stop plate opening, and pull the chains and cables out of the control columns.

j. Disconnect the roller chains from the cables.

k. Disconnect the aft portions of the forward aileron cables from the links at station 175 and thread the aft sections of the cable.

l. Remove pulley guard pins and grommets and pull the center sections of the forward cables aft. Remove the cables.

m. Loosen the turnbuckle connecting the aileron servo jumper cables at approximately station 193. Release the cables from the aileron automatic pilot pulley. Unfasten the bus cable fittings and remove the aileron servo cables from the aircraft.

n. Disconnect the forward cables from the perpendicular arm of the aileron cross.

o. Remove fairleads and guard pins, and draw the aft sections of the forward aileron cables aft.

p. Open the access doors and disconnect the turnbuckles in the cross-to-bellcrank cables at station 27 in the wing outer panels. Thread the cable ends.

q. Disconnect the cables from the links at the aileron cross, remove guard pins and draw the inboard sections of the cables inboard and out of the aircraft.

r. Disconnect the outboard sections of the cross-to-bellcrank cables from the inboard bellcranks, and draw the cables out of the aircraft.

s. Open access doors and loosen the turnbuckles at station 273, in the wing outer panels, just inboard of the outboard bellcranks.

t. Disconnect the cables from the bellcranks and thread and remove the cables.

2-89. INSTALLATION OF AILERON CONTROL CABLES.

a. Be sure that the ailerons are locked in neutral with the surface locks.

b. Adjust the aileron control rods, if necessary, so that the long arms of the wing bellcranks are perpendicular to the rear spars.

c. Secure the aileron cross in the neutral position with the ends of the perpendicular arm pointing aft. This may be accomplished by inserting two blocks of wood between the aft end of the horn arm and the corners of the lower web *(see figure 2-16).*

d. Place the automatic pilot in the engaged position.

e. Install the outboard aileron control cables between the inboard and outboard aileron wing bellcranks.

f. Route the outboard sections of the cross-to-bellcrank cables through the wings and connect them to the inboard bellcrank.

g. Connect the inboard sections of the cross-to-bellcrank cables to the links at the aileron cross and route them outboard through the wings. Connect the cables with turnbuckles at outer wing station 27.

h. Attach the aft sections of the forward aileron cables to perpendicular arm of the aileron cross and route the cables forward through the aircraft, replacing necessary fairleads and guard pins.

i. Install the aileron servo cables at the aileron automatic pilot pulley. Connect the bus cable fittings to the aileron control cable.

j. Install the center sections of the forward aileron cables and route them forward through the aircraft, replacing pulley guard pins and fairlead grommets. Connect the aft sections of the aileron cables to the links at station 175.

k. Connect the roller chains to the forward sections of the aileron cables.

l. Route the forward sections of the aileron cables through the control columns and aft through the fuselage.

CAUTION

Do not allow the chain and cable to drop down in the control column.

m. Install the aileron control wheel and sprocket assemblies. When replacing the aileron control wheels, place the open sections of the wheels in a straight line across the top. A board attached across the wheels may be used to lock the wheels in this position.

n. Position the chain assemblies on the control wheel sprockets to allow rotation of the control wheels, approximately 180 degrees in either direction from neutral, before the roller chain stops contact the stop plates.

o. Tighten the bolts in the caps through which the wheel shafts pass and install safety wire. Install the plates on the tops of the control columns.

p. Install the pulleys in the two elbows of the control columns.

q. Install the pulleys in the brackets at station 24.

r. Connect the aileron control cables with turnbuckles at approximately station 64.

s. Install necessary guard pins.

t. Tension all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

u. Remove all locking devices.

v. Adjust the aileron control system (see paragraph 2-90).

w. Safety all turnbuckles.

x. Roll out system to check for adequate clearance.

y. Install all access doors.

2-90. ADJUSTMENT OF AILERON CONTROL SYSTEM.

a. Place the aileron control wheels in the neutral position (open sections of wheels are in a straight line across the top as shown in figure 2-15). Position the aileron trim control handle, on the right side of the control pedestal, so that the trim indicator reads zero degrees. The ailerons should be faired with the outer wing trailing edges. The aileron tabs should be faired with the aileron trailing edges.

b. Clamp the aileron tabs in the faired position and disconnect and remove the forward tab control rods.

Note

With the ailerons in the faired position and the tabs clamped in the faired position, mark points in line on the inboard corners of the aileron tabs and on the outboard corners of the trailing sections of the outer wing panels for measuring aileron travel. Measure between these points when the ailerons are in the extreme UP or DOWN position.

c. Check the tension of all cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*.

d. Turn the aileron control wheels clockwise approximately 180 degrees from the neutral (right aileron UP). Adjust the stop on the right wing inboard bellcrank bracket to allow an UP throw, for the right aileron trailing edge, of 20 ($\pm\frac{1}{2}$) degrees or $9\frac{1}{16}$ ($\pm\frac{1}{4}$) inches from neutral, measured at the inboard corner of the aileron tab. Adjust the stop on the left wing inboard bellcrank bracket to allow a DOWN throw, for the left aileron trailing edge, of 14 ($\pm\frac{1}{2}$) degrees or $6\frac{3}{8}$ ($\pm\frac{1}{4}$) inches from neutral, measured in the same manner.

e. Turn the aileron control wheels counterclockwise approximately 180 degrees from neutral (left aileron UP). Adjust the stop on the left wing inboard bellcrank bracket to allow an UP throw, for the left aileron trailing edge, of 20 ($\pm\frac{1}{2}$) degrees or $9\frac{1}{16}$ ($\pm\frac{1}{4}$) inches from neutral, measured at the inboard corner of the aileron tab. Adjust the stop on the right wing inboard bellcrank bracket to allow a DOWN throw, for the left aileron trailing edge, of 14 ($\pm\frac{1}{2}$) degrees or $6\frac{3}{8}$ ($\pm\frac{1}{4}$) inches from neutral, measured in the same manner.

f. Turn the aileron control wheels back and forth to make certain that the ailerons move in opposite directions.

g. If correct movements or distances are not achieved, it will be necessary to check the rigging, as it is possible to rig differential throw into the system unless the wing bellcranks are in a true neutral position (see paragraph 2-81).

h. Safety the turnbuckles.

i. For additional information on rigging the aileron control system, see paragraph 2-231.

2-91. AILERON GEARED TABS. An aileron geared tab, constructed of conventional metal ribs and skin, is attached by a piano-type hinge to the inboard end of each aileron to provide aerodynamic boost. The aileron tabs are actuated by the control rods of the geared tab mechanisms. The right aileron tab is also actuated by a trim control mechanism, attached to the forward end of the right geared tab mechanism, for trimming purposes.

2-92. REMOVAL OF AILERON GEARED TAB. *(See figure 2-11.)*

a. Disconnect the aft aileron tab control rod from the tab horn.

b. Remove the tab hinge pin.

c. Remove the aileron tab.

2-93. BALANCING OF AILERON GEARED TAB. Balancing of the aileron geared tab is not required.

2-94. INSTALLATION OF AILERON GEARED TAB. *(See figure 2-13.)*

a. Place the aileron geared tab in position and install the tab hinge pin.

b. Connect the aft aileron tab control rod to the tab horn.

c. Tension the cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

d. Check the adjustment of the aileron geared tab control system (see paragraph 2-99).

e. Roll out system to check for proper routing and clearance.

2-95. AILERON GEARED TAB CONTROL SYSTEM. *(See figure 2-14.)* When functioning as geared tabs, the two aileron geared tab control surfaces function simultaneously in opposite directions to each other and in opposite directions to the ailerons to which they are attached. Operation of the geared tabs is controlled by the movement of the ailerons, which is transmitted by the geared tab mechanisms to the tabs.

2-96. AILERON GEARED TAB MECHANISM. *(See figure 2-14.)* Each of the two geared tab mechanisms consists of two control rods connected by a crank at the aileron spar. The forward control rod of the left tab mechanism is secured to a bracket on the rear spar of the left wing outer panel at station 146; the forward control rod of the right tab mechanism terminates at the aileron trim control drum, installed on the aft face of the rear spar of the right wing outer panel at station 146. Both aft control rods connect with the horns of the geared tabs. Movement of the ailerons is transmitted through the geared tab mechanisms to the aileron tabs to provide aerodynamic boost for the ailerons.

2-97. REMOVAL OF AILERON GEARED TAB MECHANISM.

a. Place the ailerons in neutral. Position the aileron trim control handle, on the aft face of the control pedestal, so that the trim indicator reads zero degrees.

b. Clamp the aileron tabs in the faired position.

c. Disconnect the forward control rod from the rear spar bracket (left wing outer panel) or from the aileron trim control tab drum (right wing outer panel) and from the mechanism crank on the aileron spar. Remove the control rod.

d. Disconnect the aft control rod from the mechanism crank and from the geared tab horn and remove.

e. Remove the bolt securing the mechanism crank to the bracket on the aileron spar and remove the crank.

2-98. INSTALLATION OF AILERON GEARED TAB MECHANISM.

a. Bolt the aileron geared tab mechanism crank in position at the bracket on the aileron spar.

b. Connect the aft control rod to the geared tab horn and to the mechanism crank.

c. Connect the forward control rod to the rear spar bracket (left wing outer panel), or to the aileron trim control tab drum (right wing outer panel), and to the mechanism crank at the aileron spar.

d. Release the aileron tab clamps.

e. Adjust the aileron geared tab control system (see paragraph 2-99).

2-99. ADJUSTMENT OF AILERON GEARED TAB CONTROL SYSTEM.

a. Check the adjustment of the aileron control system (see paragraph 2-90).

b. Place the aileron control wheels, on the control columns, in the neutral position (open sections of wheels are in a straight line across the top). Position the aileron trim control handle, on the right side of the control pedestal, so that the trim indicator reads zero degrees. The ailerons should be faired with the outer wing trailing edges. The aileron tabs should be faired with the aileron trailing edges.

Note

With the ailerons and the aileron tabs in the faired position, mark points in line on the outboard corners of the aileron tabs and on the inboard corners of the trailing sections of the ailerons for measuring aileron geared tab travel. Measure between these points when the aileron tabs are in the extreme UP or DOWN position.

c. Check the tension of all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

d. Set the left forward geared tab control rod so that one-half of the threads of the rod engage with the tab mechanism bracket on the rear spar; set the right forward tab control rod so that one-half of the threads of the rod engage with the aileron trim control drum. With the forward control rods in this position, adjust the aft tab control rods to position the tabs in neutral.

e. Turn the aileron control wheels clockwise as far as they will go (right aileron UP, right aileron geared tab DOWN). The right aileron will go UP. Check for a DOWN throw, for the right aileron geared tab trailing edge, of 8 ($\pm\frac{1}{2}$) degrees or $1\frac{9}{32}$ ($\pm\frac{1}{8}$) inches from the faired position (neutral), measured at the outboard corner of the aileron tab. Check for an UP throw, for the left aileron geared tab trailing edge, of 5.6 ($\pm\frac{1}{2}$) degrees or $\frac{29}{32}$ ($\pm\frac{1}{8}$) inch from the faired position, measured in the same manner.

f. Turn the aileron control wheels counterclockwise as far as they will go (left aileron UP, left aileron geared tab DOWN). The left aileron will go UP. Check for a DOWN throw, for the left aileron geared tab trailing edge, of 8 ($\pm\frac{1}{2}$) degrees or $1\frac{9}{32}$ ($\pm\frac{1}{8}$) inches from the faired position, measured at the outboard corner of the aileron tab. Check for an UP throw, for the right aileron geared tab trailing edge, of 5.6 ($\pm\frac{1}{2}$) degrees or $\frac{29}{32}$ ($\pm\frac{1}{8}$) inch from the faired position, measured in the same manner.

g. If these throws are not achieved, it will be necessary to adjust the forward aileron geared tab control rods for correct throw, adjust the tabs to the faired position with the aft control rods, and check for correct throws of the tabs according to steps d and e, preceding.

h. Safety the turnbuckles.

i. For further information on rigging the aileron geared tab control system, see paragraph 2-231.

2-100. AILERON TRIM CONTROL SYSTEM. *(See figure 2-17.)* The right aileron tab functions both as a geared tab and as a trim control tab. When the tab is used for correcting lateral trim, it is controlled by the aileron trim control handle, on the aft face of the control pedestal, through a cable drum and two-way cable system that terminates at the trim control tab drum, located on the aft face of the rear spar at station 146 in the right wing outer panel. Movement is transmitted by the control drum, through the geared tab mechanism, to the aileron tab. Cable stops, attached to the aileron trim control cables, limit the movement of the cables by striking a fixed fairlead at station 156 in the fuselage. They are not adjustable.

2-101. AILERON TRIM CONTROL HANDLE ASSEMBLY AND DRUM. *(See figure 2-17.)* An aileron trim control handle protrudes from the right aft face of the control pedestal and is accessible to both pilots. It is attached to a shaft, extending through the pedestal, on which the trim control cable drum is installed. The drum is actuated by the aileron trim control handle assembly and transmits movement, through the two-way cable system, to the aileron trim control tab drum in the right outer wing panel. An aileron trim control indicator, located below the control handle, records the movement of the cable drum by means of a gearing arrangement.

2-102. REMOVAL OF AILERON TRIM CONTROL HANDLE ASSEMBLY AND DRUM.

a. Remove the round inspection plate from the right side of the control pedestal.

b. Tape the cables at the drums in the pedestal and right outer wing to prevent them from unwrapping.

c. Remove the access doors on the under side of the aircraft and disconnect the aileron trim control cables at station 117. Thread the forward cable ends.

d. Remove the necessary pulleys from the pulley brackets to release the aileron trim control cables between station 117 and the pedestal.

e. Release and remove the aileron trim control shaft collar on the forward side of the pedestal.

f. Loosen the cable drum from the shaft. Hold the cable drum and draw the shaft, with the control handle attached, aft and out of the pedestal.

g. Remove the cable drum and cable from the pedestal.

2-103. MINOR REPAIR AND REPLACEMENT OF AILERON TRIM CONTROL HANDLE ASSEMBLY AND DRUM. Replace the spur gear if the teeth are worn or broken. Replace bushings if worn or defective.

2-104. INSTALLATION OF AILERON TRIM CONTROL HANDLE ASSEMBLY AND DRUM.

a. Place the aileron control wheels, on the control columns, in the neutral position.

b. Clamp the right aileron tab in the faired position.

c. Route the forward trim control cables down through the pedestal and aft, replacing necessary pulleys. Attach cables at station 117 but do not tighten.

d. Remove tape from cables and drums.

e. Set the aileron trim control indicator in neutral.

f. Place the trim control drum so that the teeth of the spur gear on the drum engage with the teeth on the top of the aileron indicator shaft gear (indicator at neutral) and slide the trim control shaft, with handle attached, through the pedestal and cable drum into position in the forward face of the pedestal.

g. Secure the shaft in position by attaching the collar on the forward end of the shaft.

h. Secure the cable drum to the shaft.

i. Tension all cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*.

j. Unclamp the right aileron tab and adjust the aileron trim control system (see paragraph 2-116).

k. Safety the turnbuckles.

l. Replace the access doors on the under side of the aircraft and the round inspection plate on the right side of the pedestal.

2-105. AILERON TRIM CONTROL INDICATOR ASSEMBLY. *(See figure 2-17.)* An aileron trim control indicator assembly, provided to record the movement applied to the right aileron tab by the trim control handle, is located on the right aft face of the control pedestal just below the handle.

2-106. REMOVAL OF AILERON TRIM CONTROL INDICATOR ASSEMBLY.

a. Remove the aileron trim control handle assembly and drum (see paragraph 2-102).

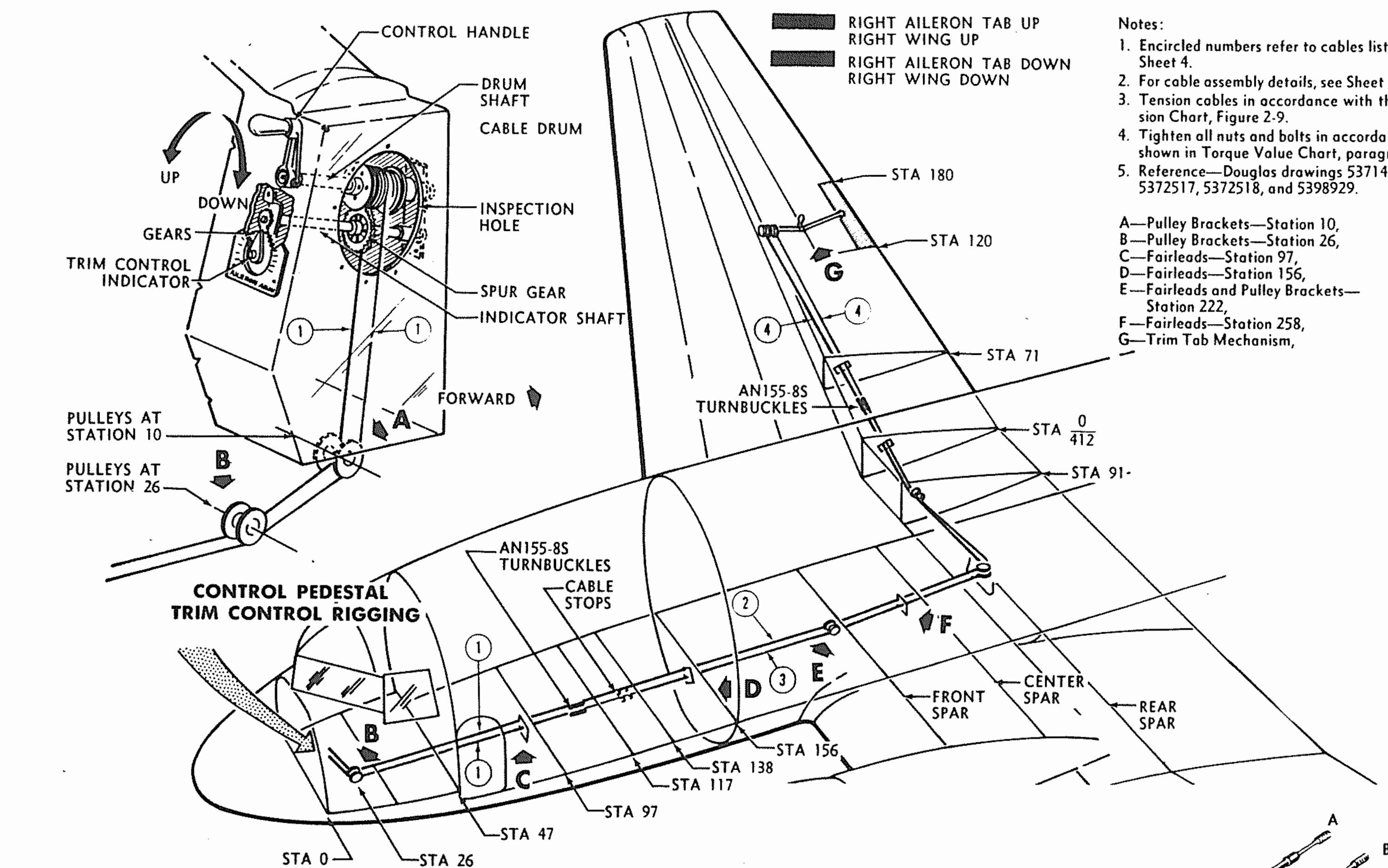
b. Release the taper pin from the indicator shaft.

c. Release the screws from the indicator bracket and remove the indicator.

d. Work the indicator shaft and gear out of the pedestal.

2-107. MINOR REPAIR AND REPLACEMENT OF AILERON TRIM CONTROL INDICATOR ASSEMBLY. Replace gears if teeth are worn or broken.

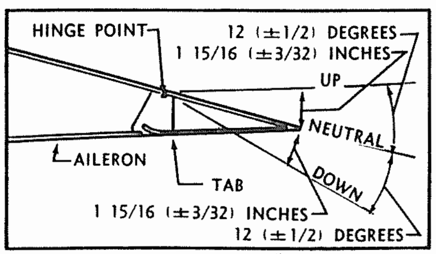
(Continued on Page 120)



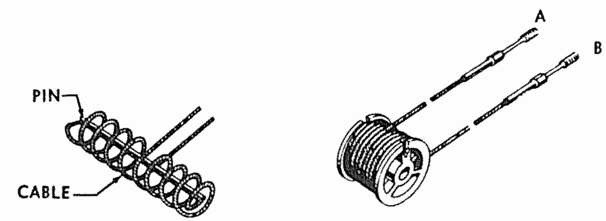
Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 4.
2. For cable assembly details, see Sheet 4.
3. Tension cables in accordance with the Cable Rigging Tension Chart, Figure 2-9.
4. Tighten all nuts and bolts in accordance with torque values shown in Torque Value Chart, paragraph 2-453.
5. Reference—Douglas drawings 5371403, 5372511, 5372516, 5372517, 5372518, and 5398929.

A—Pulley Brackets—Station 10,	see Sheet 2.
B—Pulley Brackets—Station 26,	see Sheet 2.
C—Fairleads—Station 97,	see Sheet 2.
D—Fairleads—Station 156,	see Sheet 2.
E—Fairleads and Pulley Brackets—Station 222,	see Sheet 3.
F—Fairleads—Station 258,	see Sheet 3.
G—Trim Tab Mechanism,	see Sheet 4.



RIGHT AILERON TAB TRIM CONTROL ADJUSTMENT



AILERON TRIM CONTROL DRUM WRAPPING PROCEDURE (TYPICAL FOR BOTH DRUMS)

1. Insert cable and pin. Wrap cable on drum from each end to center with equal number of turns on each side.
2. Wrap cable drums so that equal lengths project at A and B.

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Figure 2-17 (Sheet 1 of 4 Sheets). Aileron Trim Control System — Key Drawing

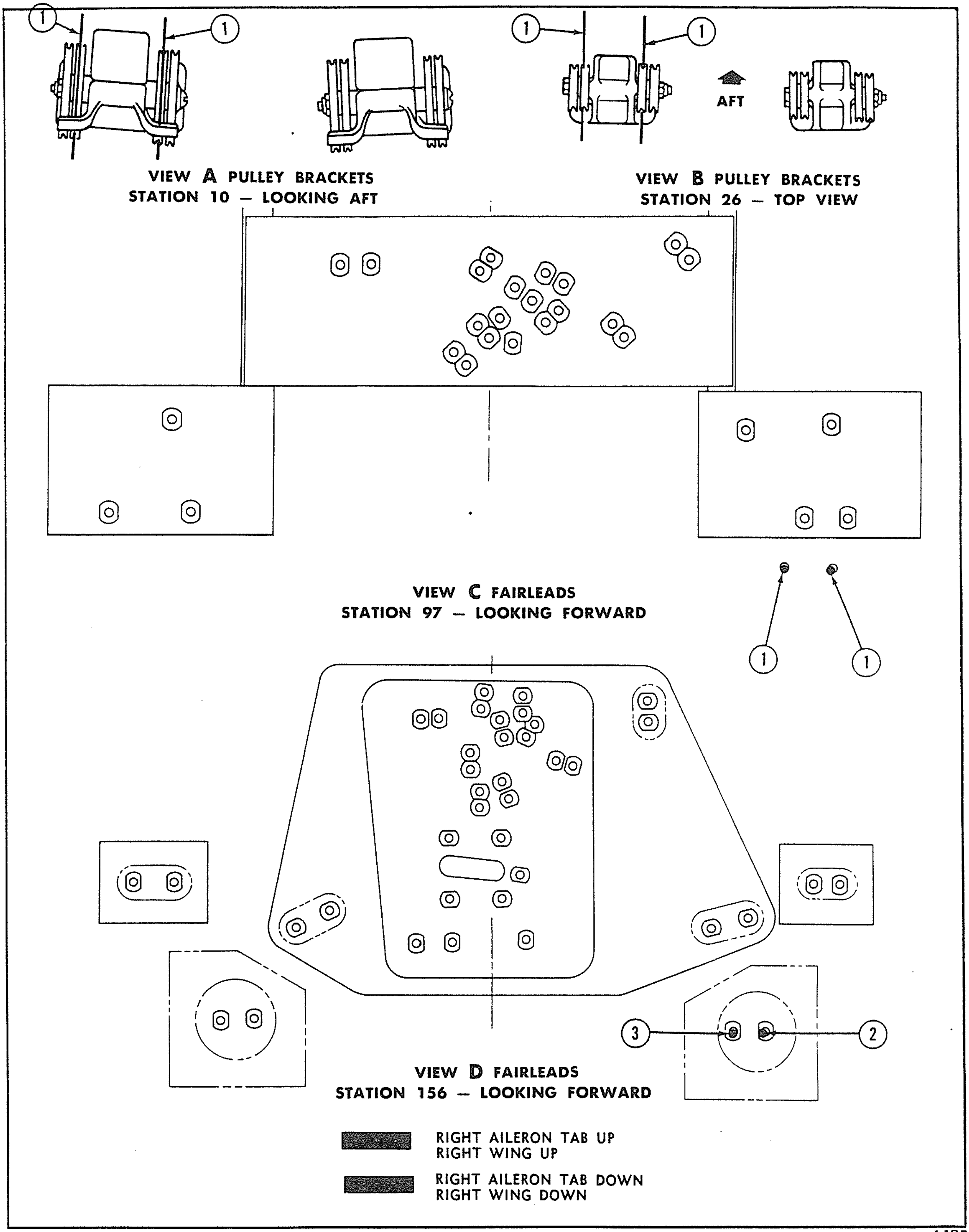
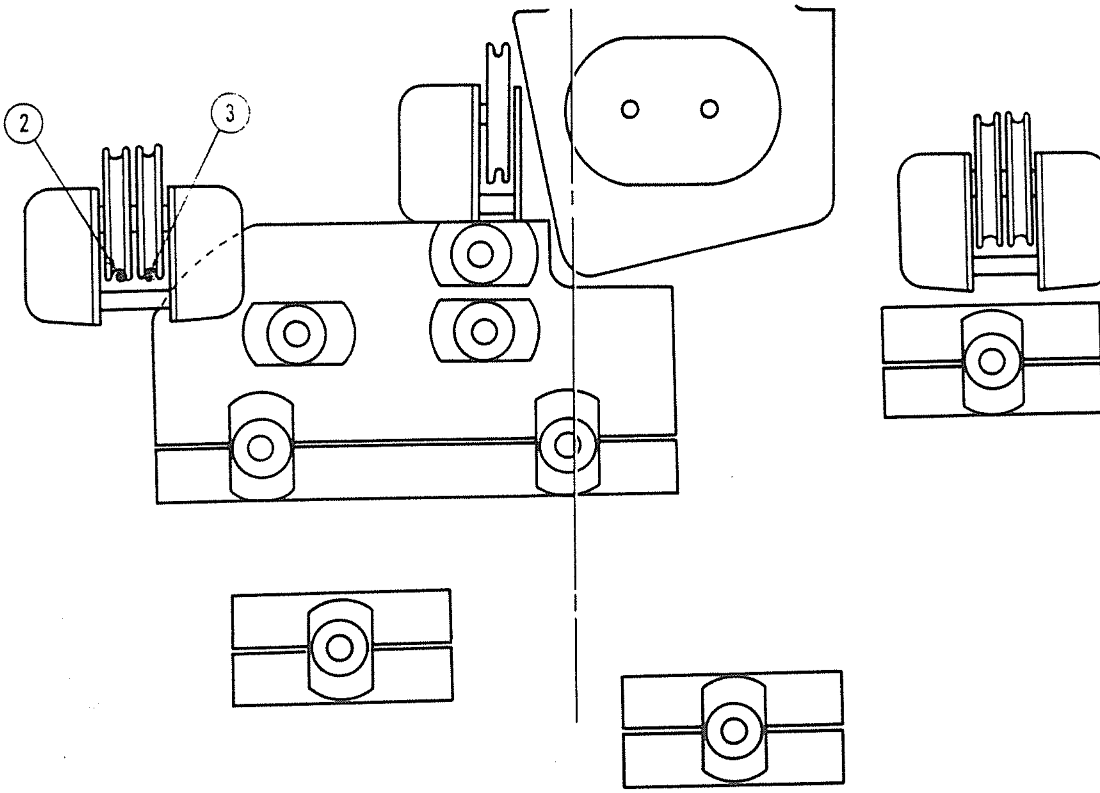
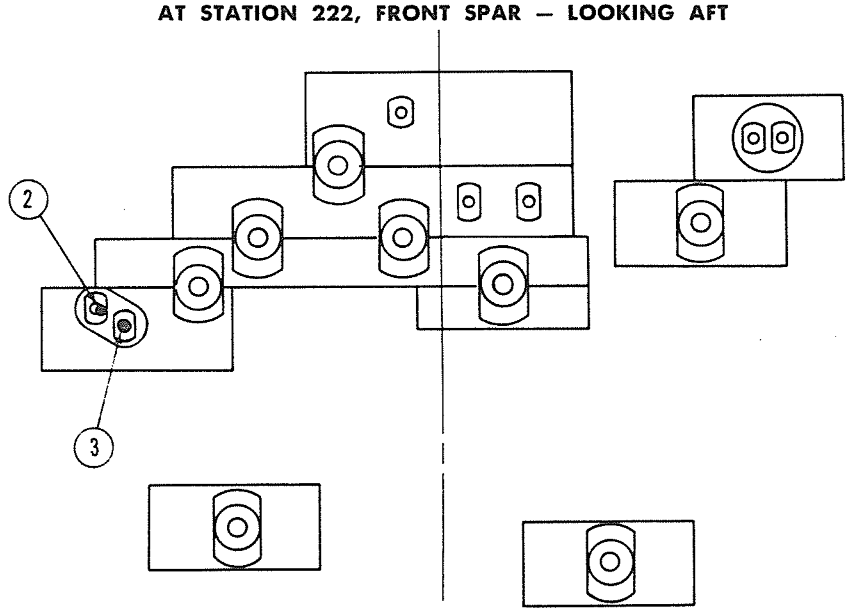


Figure 2-17 (Sheet 2 of 4 Sheets). Aileron Trim Control System — Pulley Brackets, Stations 10 and 26; Fairleads, Stations 97 and 156



VIEW E FAIRLEADS AND PULLEY BRACKETS,
AT STATION 222, FRONT SPAR — LOOKING AFT



VIEW F FAIRLEADS AT STATION 258
CENTER SPAR — LOOKING AFT

RIGHT AILERON TAB UP
RIGHT WING UP

RIGHT AILERON TAB DOWN
RIGHT WING DOWN

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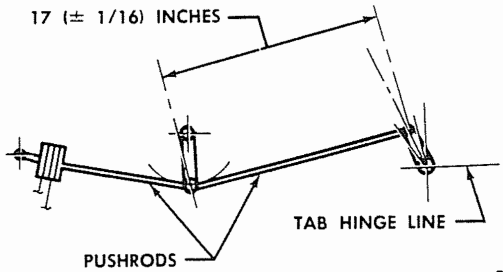
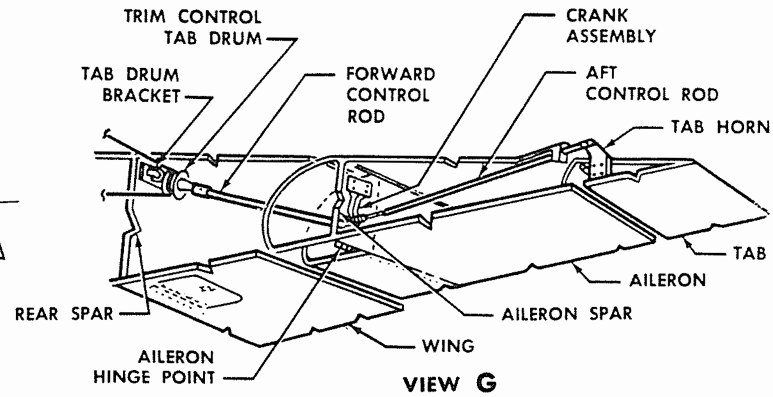
Figure 2-17 (Sheet 3 of 4 Sheets). Aileron Trim Control System — Fairleads and Pulley Brackets, Station 222; Fairleads, Station 258



RIGHT WING TRIM TAB PUSHRODS NEUTRAL POSITION DIAGRAM

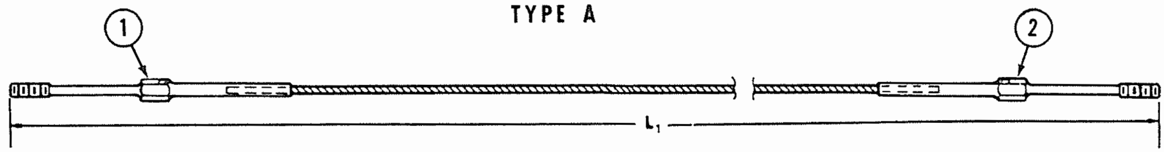


VIEW G
AILERON TRIM CONTROL TAB DRUM AND GEARED TAB MECHANISM

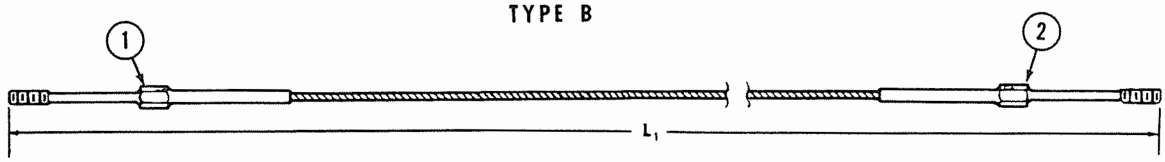
AILERON TRIM CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L1)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)
1	2115350-8 RH Tab Up RH Tab Down	1	A	125 3/4	1/16 dia 7x7 flex	S-2049220 -8DS-2R	S-2049220 -8DS-2R		
2	3391412-593 RH Tab Up	1	B	358 7/8	1/16 dia 7x7 flex	AN669S2 LH	AN669S2 LH		
3	3391412-595 RH Tab Down	1	C	358 7/8	1/16 dia 7x7 flex	AN669S2 LH	AN669S2 RH		
4	3391412-591 RH Tab Up RH Tab Down	1	C	312 1/4	1/16 dia 7x7 flex	AN669S2 LH	AN669S2 RH		

TYPE A



TYPE B



TYPE C

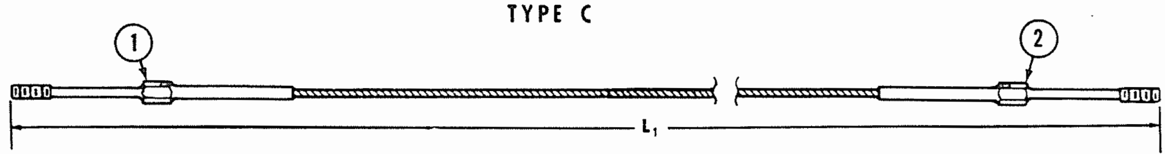


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Figure 2-17 (Sheet 4 of 4 Sheets). Aileron Trim Control System — Trim Tab Drive Mechanism, Cable Chart and Cable Assemblies

(Continued from Page 115)

2-108. INSTALLATION OF AILERON TRIM CONTROL INDICATOR ASSEMBLY.

a. Work the indicator shaft and gear into the control pedestal.

b. Install the indicator and tighten the screws on the indicator bracket.

c. Install the taper pin in the indicator shaft.

d. Install the aileron trim control handle assembly and drum (see paragraph 2-104).

2-109. AILERON TRIM CONTROL TAB DRUM. *(See figure 2-17.)* The aileron trim control tab drum is installed on the aft face of the rear spar of the right outer wing panel at station 145. The drum is actuated by the two-way cable system and transmits movement through the geared tab mechanism to the right aileron tab.

2-110. REMOVAL OF AILERON TRIM CONTROL TAB DRUM.

a. Place the ailerons in neutral. Position the aileron trim control handle, on the aft face of the control pedestal, so that the trim indicator reads zero degrees.

b. Clamp the right aileron tab in the faired position.

c. Tape the cables and drums in wing and pedestal to prevent them from unwrapping.

d. Open the access door and disconnect the aileron trim control cables at station 25 in the right wing outer panel. Pull the cable ends outboard to the drum.

e. Disconnect the forward control rod from the geared tab mechanism crank on the right aileron spar and from the aileron trim control tab drum. Remove the control rod.

f. Remove the drum from the bracket on the rear spar.

2-111. MINOR REPAIR AND REPLACEMENT OF AILERON TRIM CONTROL TAB DRUM. Replace worn or damaged bearings, bushings, oil seals, and washers.

2-112. INSTALLATION OF AILERON TRIM CONTROL TAB DRUM.

a. Install the trim control tab drum on the bracket on the rear spar.

b. Connect the forward control rod to the geared tab mechanism crank on the right aileron spar and to the aileron trim control tab drum.

c. Route the control cables inboard to station 25 and connect the cables with turnbuckles.

d. Tension the aileron trim control cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

e. Remove the right aileron tab surface lock.

f. Adjust the aileron trim control system (see paragraph 2-116).

g. Adjust the aileron geared tab control system (see paragraph 2-99).

h. Safety the turnbuckles at station 25 in the right wing outer panel.

2-113. AILERON TRIM CONTROL CABLES. A two-way cable system originating at the aileron trim control drum in the pedestal extends aft through the fuselage to the center wing and outboard to the aileron trim control tab drum on the rear spar of the right outer wing panel. Cable stops, attached to the aileron trim control cables, limit the movement of the cables by striking a fixed fairlead at station 156 in the fuselage. The stops are not adjustable *(see figure 2-17)*.

2-114. REMOVAL OF AILERON TRIM CONTROL CABLES.

a. Place the aileron control wheels in neutral. If necessary, adjust the aileron trim control handle on the pedestal until the right aileron tab is in the faired position.

b. Clamp the right aileron tab in the faired position.

c. Tape cables and drums in both pedestal and right wing to prevent them from unwrapping.

d. Remove the aileron trim control drum from the pedestal (see paragraph 2-102). Unwrap the forward cable from the drum.

e. Open the access door and disconnect the aileron trim control cables at station 25 in the right outer wing panel. Thread the cables. .

f. Remove the necessary fairlead grommets and pulley guard pins and draw the intermediate control cables inboard and forward. Remove the cables through the access door on the under side of the fuselage forward of station 156.

g. Draw the aileron trim control tab drum cable ends outboard to the drum.

h. Disconnect the forward control rod from the geared tab mechanism crank on the right aileron spar and from the aileron trim control tab drum. Remove the control rod.

i. Remove the drum from the bracket on the rear spar. Unwrap the cable from the drum.

2-115. INSTALLATION OF AILERON TRIM CONTROL CABLES.

a. Place the aileron control wheels in neutral and clamp the right aileron tab in the faired position.

b. Wrap the aft aileron trim control cable on the trim control tab drum *(see figure 2-17)*. Bolt the drum to the bracket on the rear spar at station 145 in the right wing outer panel.

c. Connect the forward end of the geared tab mechanism forward control rod to the trim control tab drum. Bolt the aft end of the rod to the mechanism crank on the aileron spar.

d. Route the tab drum cable ends inboard through the wing.

e. Insert the intermediate control cables through the access door forward of station 156 in the fuselage and route the appropriate cable ends aft to the center wing and outboard through the right wing to station 25 in the outer wing panel Replace grommets and guard pins. Connect cables at station 25 in the outer wing panel.

f. Wrap the forward aileron trim control cable on the trim control drum and tape *(see figure 2-17).*

g. Route the forward trim control cables down through the pedestal and aft, replacing necessary pulleys. Attach cables at station 117, but do not tighten.

h. Set the aileron trim control indicator in neutral.

i. Place the trim control drum so that the teeth of the spur gear on the drum engage with the teeth on the top of the aileron indicator shaft gear (indicator at neutral) and slide the trim control shaft, with handle attached, through the pedestal and cable drum into position in the forward face of the pedestal.

j. Secure the shaft in position by attaching the collar on the forward end of the shaft.

k. Secure the cable drum to the shaft.

l. Tension all cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

m. Remove tape from cables and drums.

n. Unclamp the right aileron tab.

o. Adjust the aileron trim control system (see paragraph 2-116).

p. Check the adjustment of the aileron geared tab control system (see paragraph 2-99).

q. Safety the turnbuckles.

r. Replace the access doors and the round inspection plate on the right side of the pedestal.

2-116. ADJUSTMENT OF AILERON TRIM CONTROL SYSTEM.

a. Place the aileron control wheels, on the control columns, in the neutral position. Position the aileron trim control handle, on the right side of the control pedestal, so that the trim indicator reads zero degrees (neutral). The aileron tabs should be faired with the ailerons.

Note

With the aileron tabs in the faired position, mark points in line on the outboard corner of the right aileron tab and on the inboard corner of the trailing section of the right aileron for measuring the aileron tab travel. Measure between these points when the tab is in the extreme UP or DOWN position.

b. Check the tension of all cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

c. Retain the aileron control wheels in neutral position and turn the aileron trim control handle in a clockwise direction until the indicator on the control pedestal reads full RIGHT WING DOWN. The right aileron tab should be DOWN 12 ($\pm\frac{1}{2}$) degrees or $1\frac{15}{16}$ ($\pm\frac{3}{32}$) inches from the faired position (neutral), measured at the outboard corner of the aileron tab.

d. Retain the aileron control wheels in neutral position and turn the aileron trim control handle in a counterclockwise direction until the indicator on the control pedestal reads full LEFT WING DOWN. The right aileron tab should be UP 12 ($\pm\frac{1}{2}$) degrees or $1\frac{15}{16}$ ($\pm\frac{3}{32}$) inches from the faired position, measured at the outboard corner of the aileron tab.

e. Safety the turnbuckles.

f. If correct throws are not achieved for the right aileron tab it will be necessary to check both the rigging of the aileron trim control system *(see figure 2-17),* and the adjustment of the aileron geared tab control system (see paragraph 2-99).

g. For further information on rigging the aileron trim control system, see paragraph 2-231.

2-117. ELEVATORS. The two elevators are of all-metal aluminum-alloy construction with fabric covering, and are statically and dynamically balanced by lead weights installed in the leading edges. Each elevator is supported at the horizontal stabilizer by four hinge fittings and rigidly connected to the elevator torque tube at the inboard edge. A trim tab and a spring control tab and mechanism are incorporated in each elevator.

2-118. REMOVAL OF ELEVATOR.
(See figure 2-18.)

a. Set the elevators in neutral and clamp the tabs in the faired position.

b. Remove the bolt and disconnect the elevator trim tab forward control rod from the crank at the elevator spar.

c. Disconnect the elevator spring control tab torque tube from the spring tab mechanism and structure support in the elevator.

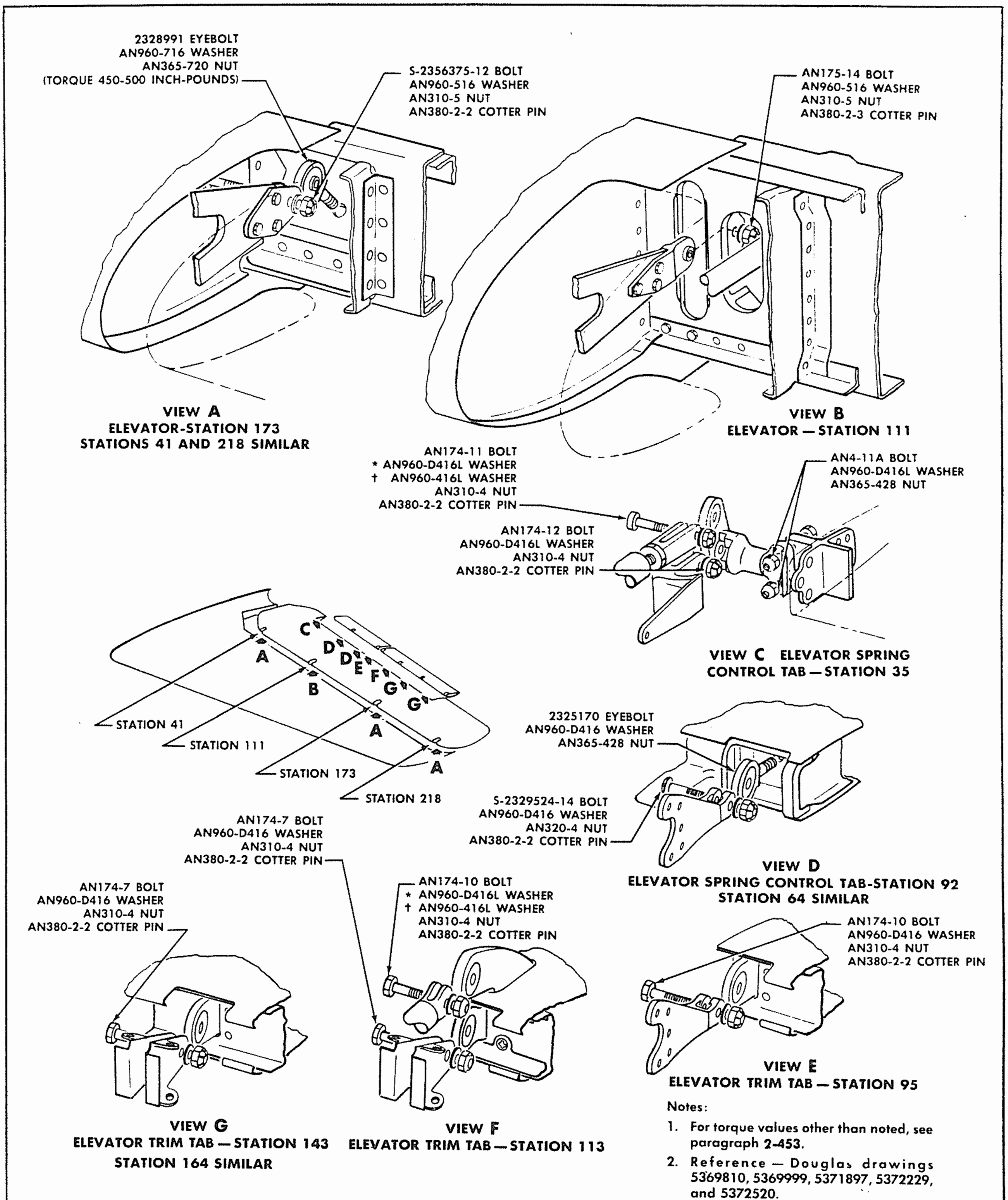
d. Support the elevator and remove the elevator torque tube attaching bolts.

e. Remove the stop nuts from the eyebolt studs at the hinge stations, accessible through access doors in the elevator.

f. Remove the elevator.

2-119. BALANCING OF ELEVATOR (PREFERRED METHOD).
(See figure 2-19.)

a. Remove the elevator from the aircraft (see paragraph 2-118).



Notes:

1. For torque values other than noted, see paragraph 2-453.
2. Reference — Douglas drawings 5369810, 5369999, 5371897, 5372229, and 5372520.

*Aircraft A, B, and 1 through 9.
†Aircraft C, D, 10, and subsequent.

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Figure 2-18. Installation of Elevator and Elevator Tabs

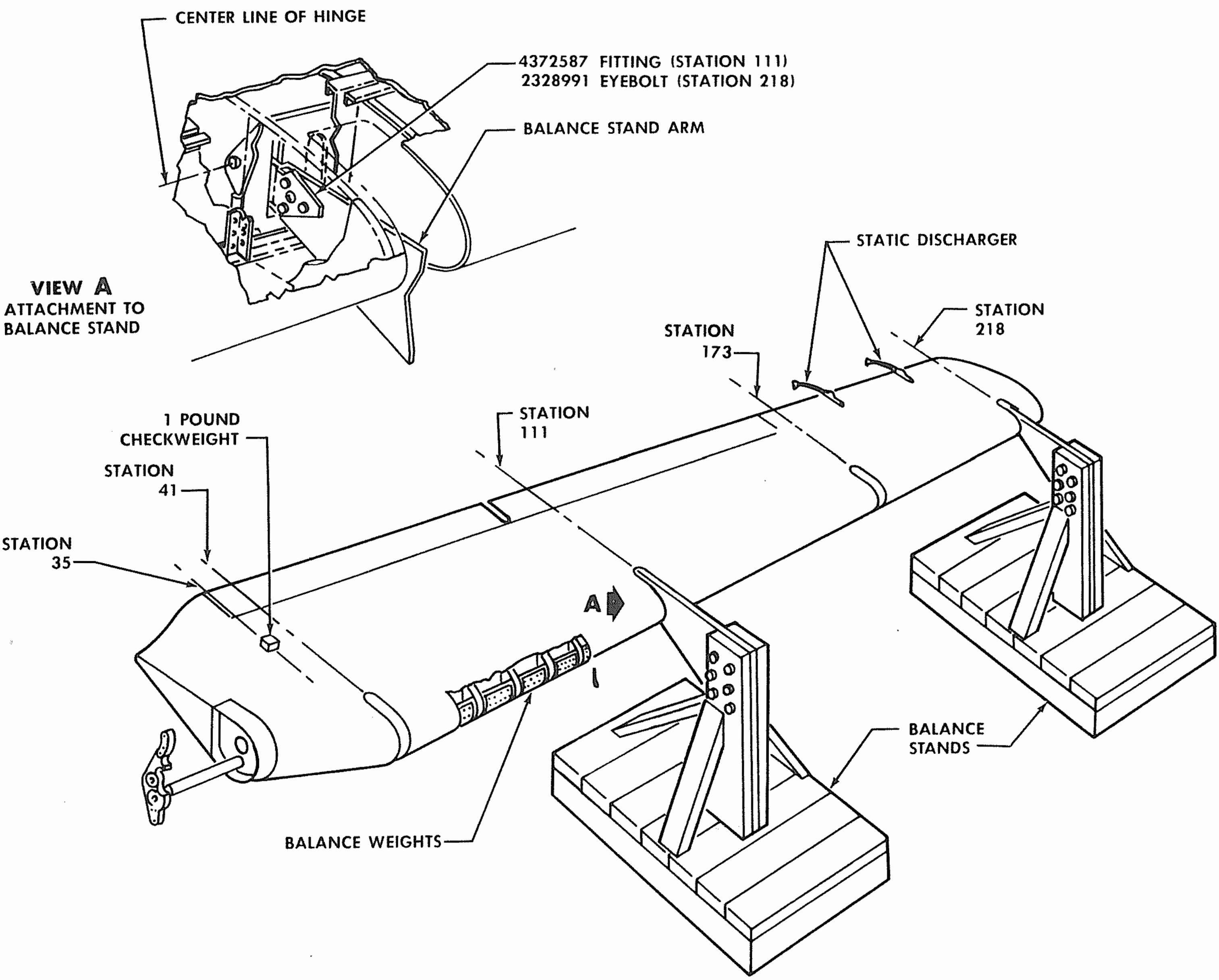


Figure 2-19. Balancing Elevator

b. Mount the elevator, with the tab torque tube attached, on the elevator balancing stands. Do not include the elevator torque tube. Check the hinge bearings for binding and adjust the stands so that the surface swings freely.

Note

The location of the hinge line is extremely critical. The attaching eyebolts must be snug so that the hinge line (relative to the surface) is the same as the hinge line when the surface is installed on the aircraft.

c. Place a 1-pound weight on the elevator at station 35.9 (opposite the inboard end of the control tab). Move the weight along this station line until the elevator is horizontal. Measure the distance from the extreme trailing edge of the elevator to the center of the weight. This distance must be between 18⅝ and 31⅛ inches for the elevator to be correctly balanced.

d. If the distance is more than 31⅛ inches, but less than 36⅛ inches, add 0.4 pound in the bay between stations 115 and 124 and repeat step c.

e. If the distance is more than 36⅛ inches, but less than 41⅛ inches, add 0.8 pound in the bay between stations 115 and 124 and repeat step c.

f. If the distance is more than 41⅛ inches, but less than 46⅛ inches, accomplish step e and add 0.38 pound in the bay between stations 102 and 110 and repeat step c.

g. If the distance is less than 18⅝ inches, but more than 13⅝ inches, remove 0.4 pound from the bay between stations 115 and 124 and repeat step c.

h. If the distance is less than 13⅝ inches, but more than 8⅝ inches, remove 0.8 pound from the bay between stations 115 and 124 and repeat step c.

i. If the distance is less than 8⅝ inches, but more than 3⅝ inches, accomplish step h and remove 0.38 pound from the bay between stations 102 and 110 and repeat step c.

Note

When the elevator has been thrown out of balance because of a weight change, adjust the counterbalance weight which is opposite or nearest to the change. When the elevator has been thrown out of balance due to repairs made in a number of places or when new covering, paint, dope, etc., is applied, balance the elevator in accordance with the steps outlined above.

2-120. BALANCING OF ELEVATOR (ALTERNATE METHOD).

a. Remove the bolts that attach the elevator tab torque tube to the elevator tab bellcrank.

b. Unbolt the elevator torque tube from the elevators.

c. Balance the elevators independently of each other in accordance with the balancing procedure outlined in paragraph 2-119, steps c through i.

Note

Keep the elevator trim tab in the neutral position while balancing the elevators on the aircraft.

2-121. ELEVATOR HINGE WEAR TOLERANCES. Due to the close tolerance fits of the elevator hinge bolts and bearings, the maximum wear limit is 0.001 inch in excess of the manufacturing tolerance between the two parts.

2-122. INSTALLATION OF ELEVATOR. *(See figures 2-18 and 2-20.)*

a. Support the elevator in position and install the eyebolts in the hinges.

b. Bolt the elevator torque tube to the elevator.

c. Connect the elevator spring control tab torque tube to the spring tab mechanism and structure support in the elevator.

d. Attach the elevator trim tab forward control rod to the crank at the elevator spar.

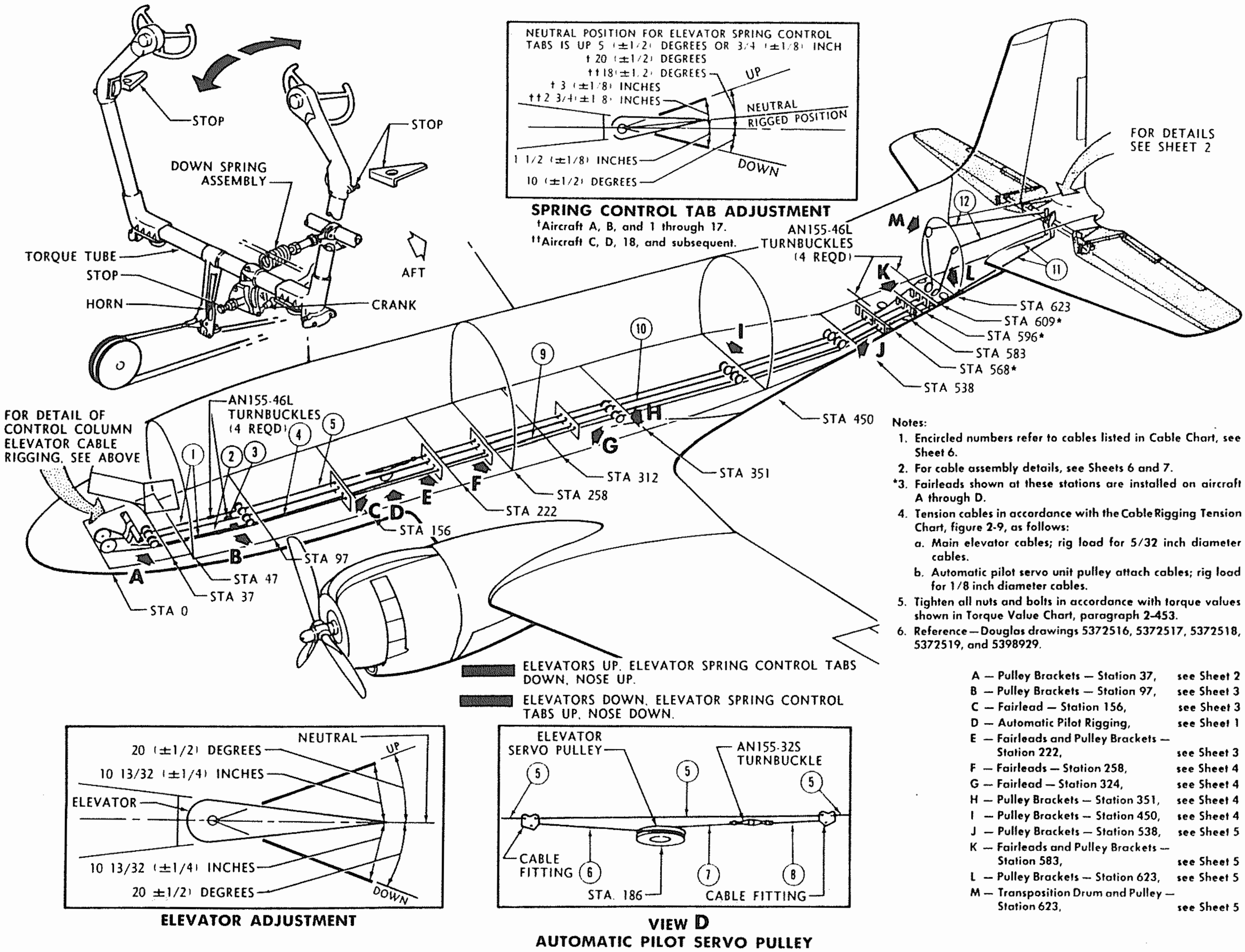
e. Unclamp the elevator tabs.

f. Adjust the elevator control system (see paragraph 2-140).

2-123. ELEVATOR CONTROL SYSTEM. *(See figure 2-20.)* The two elevator surfaces function simultaneously in the same directions and are controlled by forward and aft movement of the dual control columns in the flight compartment. Elevator movement is limited by adjustable stops mounted on the tail section structure; adjustable stops on the elevator torque tube at the elevator-operating cranks limit the movement of the spring control tabs. Separate two-way cable systems connect the control columns to the two elevator-operating cranks on the spring control tab torque tubes in the tail section. Control column travel is limited by adjustable stops at the upper elbows of the columns and on the fuselage structure. A down spring assembly attached to the control column torque tube assists in providing symmetrical forces for extreme cg conditions. An automatic pilot servo unit is connected to the right (pilot's) nose DOWN cable (elevator DOWN) at station 179.

2-124. CONTROL COLUMNS. *(See figure 2-20.)* The dual control columns located in the flight compartment are connected at their bases by a torque tube to effect identical movement. The columns form the initial controlling units of the elevator and elevator spring control tab control system. A horn projecting

(Continued on Page 132)



Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 6.
2. For cable assembly details, see Sheets 6 and 7.
*3. Fairleads shown at these stations are installed on aircraft A through D.
4. Tension cables in accordance with the Cable Rigging Tension Chart, figure 2-9, as follows:
   a. Main elevator cables; rig load for 5/32 inch diameter cables.
   b. Automatic pilot servo unit pulley attach cables; rig load for 1/8 inch diameter cables.
5. Tighten all nuts and bolts in accordance with torque values shown in Torque Value Chart, paragraph 2-453.
6. Reference—Douglas drawings 5372516, 5372517, 5372518, 5372519, and 5398929.

A — Pulley Brackets — Station 37,	see Sheet 2
B — Pulley Brackets — Station 97,	see Sheet 3
C — Fairlead — Station 156,	see Sheet 3
D — Automatic Pilot Rigging,	see Sheet 1
E — Fairleads and Pulley Brackets — Station 222,	see Sheet 3
F — Fairleads — Station 258,	see Sheet 4
G — Fairlead — Station 324,	see Sheet 4
H — Pulley Brackets — Station 351,	see Sheet 4
I — Pulley Brackets — Station 450,	see Sheet 4
J — Pulley Brackets — Station 538,	see Sheet 5
K — Fairleads and Pulley Brackets — Station 583,	see Sheet 5
L — Pulley Brackets — Station 623,	see Sheet 5
M — Transposition Drum and Pulley — Station 623,	see Sheet 5

Figure 2-20 (Sheet 1 of 7 Sheets). Elevator and Elevator Spring Control Tab Control System — Key Drawing

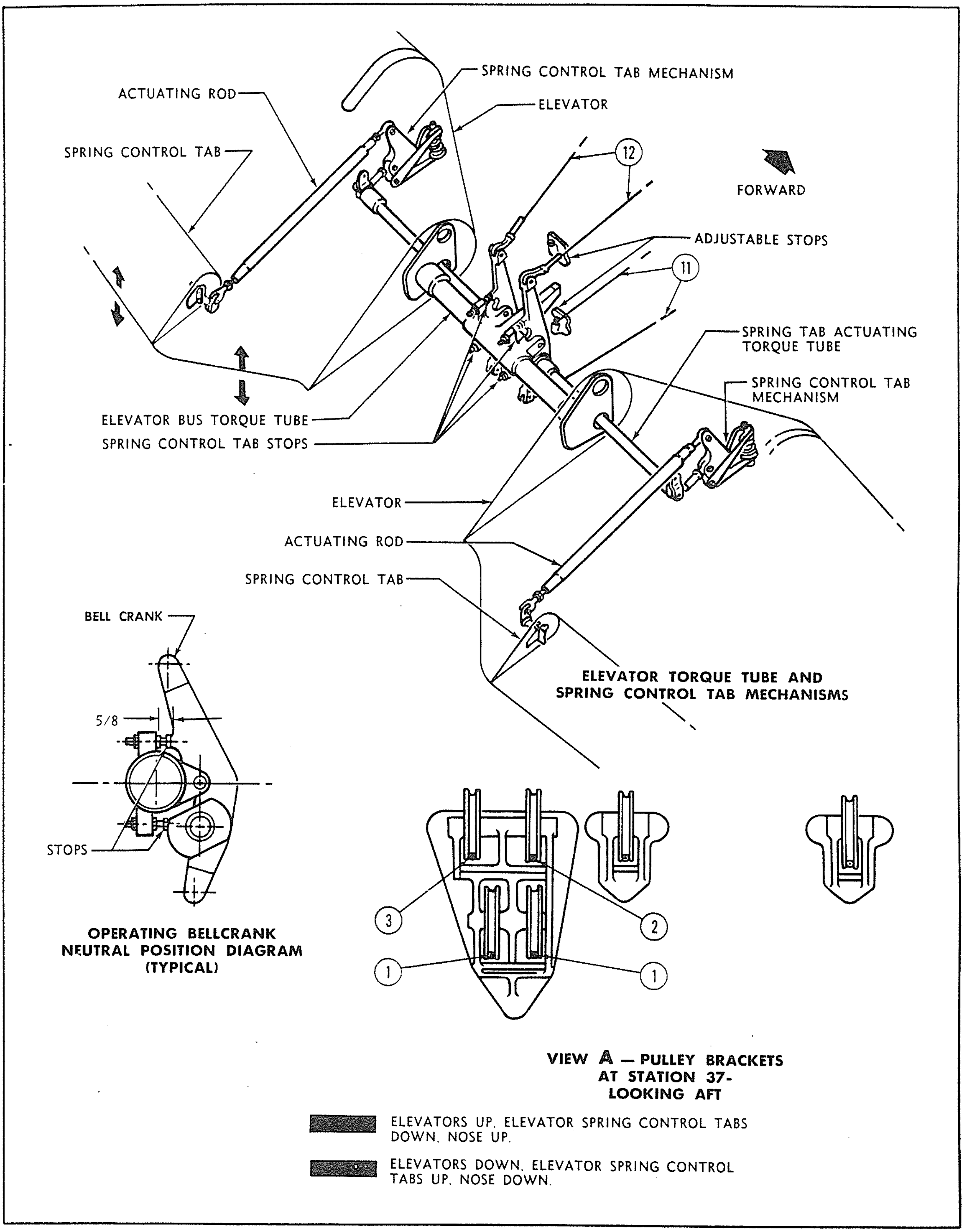
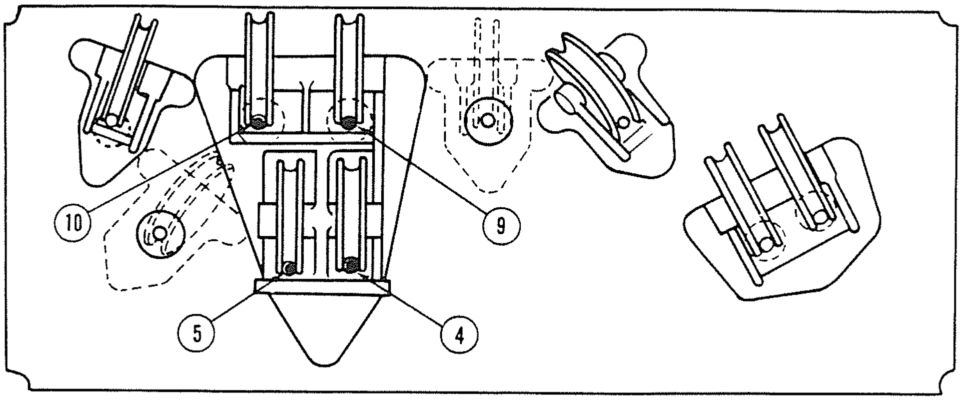
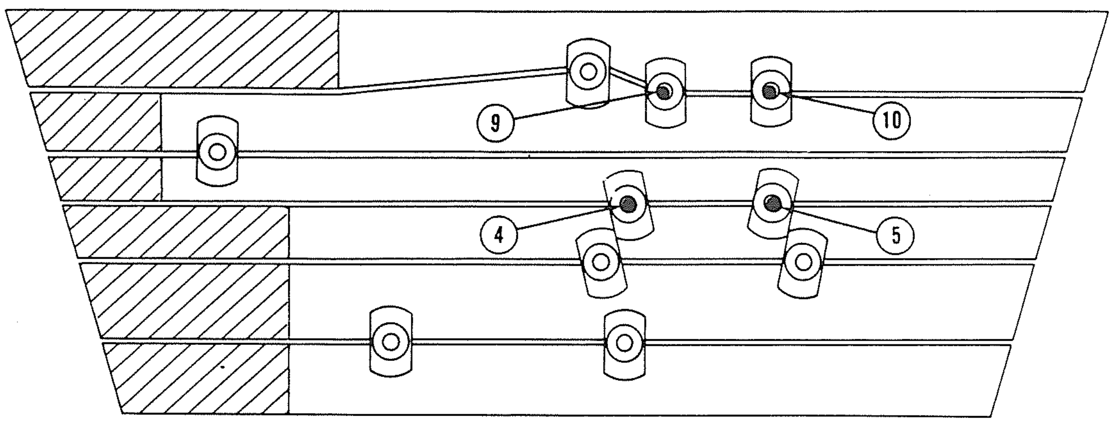


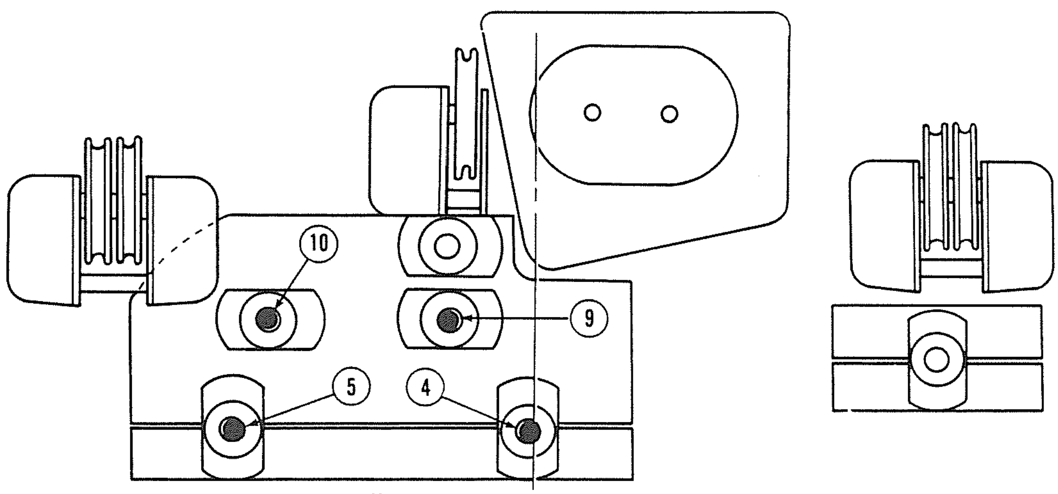
Figure 2-20 (Sheet 2 of 7 Sheets). Elevator and Elevator Spring Control Tab Control System — Torque Tube, Pulley Brackets, Station 37



VIEW B — PULLEY BRACKETS
AT STATION 97—LOOKING AFT



VIEW C — FAIRLEAD AT
STATION 156—LOOKING FORWARD



VIEW E — FAIRLEADS AND PULLEY BRACKETS
AT STATION 222, FRONT SPAR—LOOKING AFT

ELEVATORS UP, ELEVATOR SPRING CONTROL TABS DOWN, NOSE UP
ELEVATORS DOWN, ELEVATOR SPRING CONTROL TABS UP, NOSE DOWN
SHADED PORTION OF FAIRLEAD EFFECTIVE ON AIRCRAFT A. B. AND 1 THROUGH 28.

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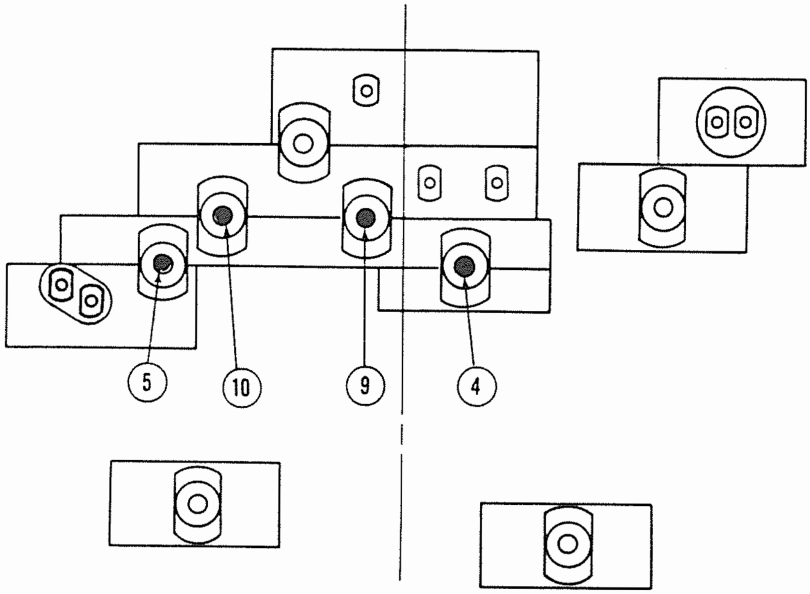
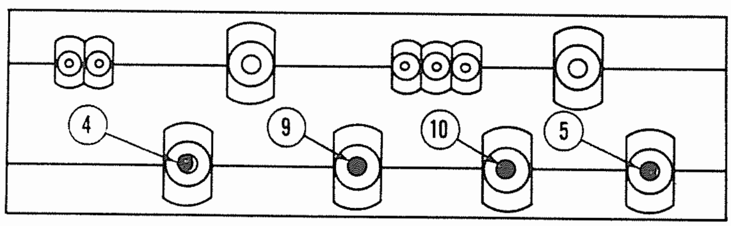
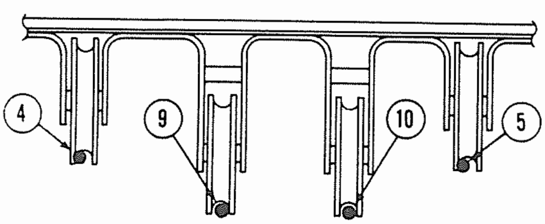
Figure 2-20 (Sheet 3 of 7 Sheets). Elevator and Elevator Spring Control Tab Control System — Pulley Brackets, Station 97; Fairlead, Station 156; Fairleads and Pulley Brackets, Station 222



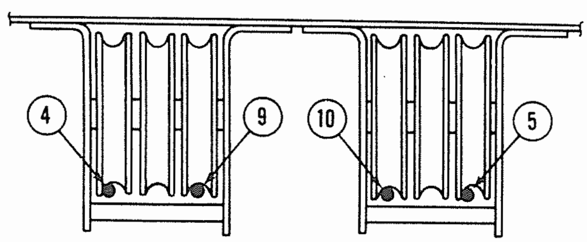
VIEW F — FAIRLEADS AT
STATION 258, CENTER SPAR (LOOKING AFT)



VIEW G — FAIRLEAD AT
STATION 324 (LOOKING FORWARD)



VIEW H — PULLEY BRACKETS
AT STATION 351 (LOOKING FORWARD)



VIEW I — PULLEY BRACKETS
AT STATION 450 (LOOKING FORWARD)

ELEVATORS UP, ELEVATOR SPRING CONTROL TABS DOWN, NOSE UP.

ELEVATORS DOWN, ELEVATOR SPRING CONTROL TABS UP, NOSE DOWN.

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Figure 2-20 (Sheet 4 of 7 Sheets). Elevator and Elevator Spring Control Tab Control System — Fairleads, Stations 258 and 324; Pulley Brackets, Stations 351 and 450

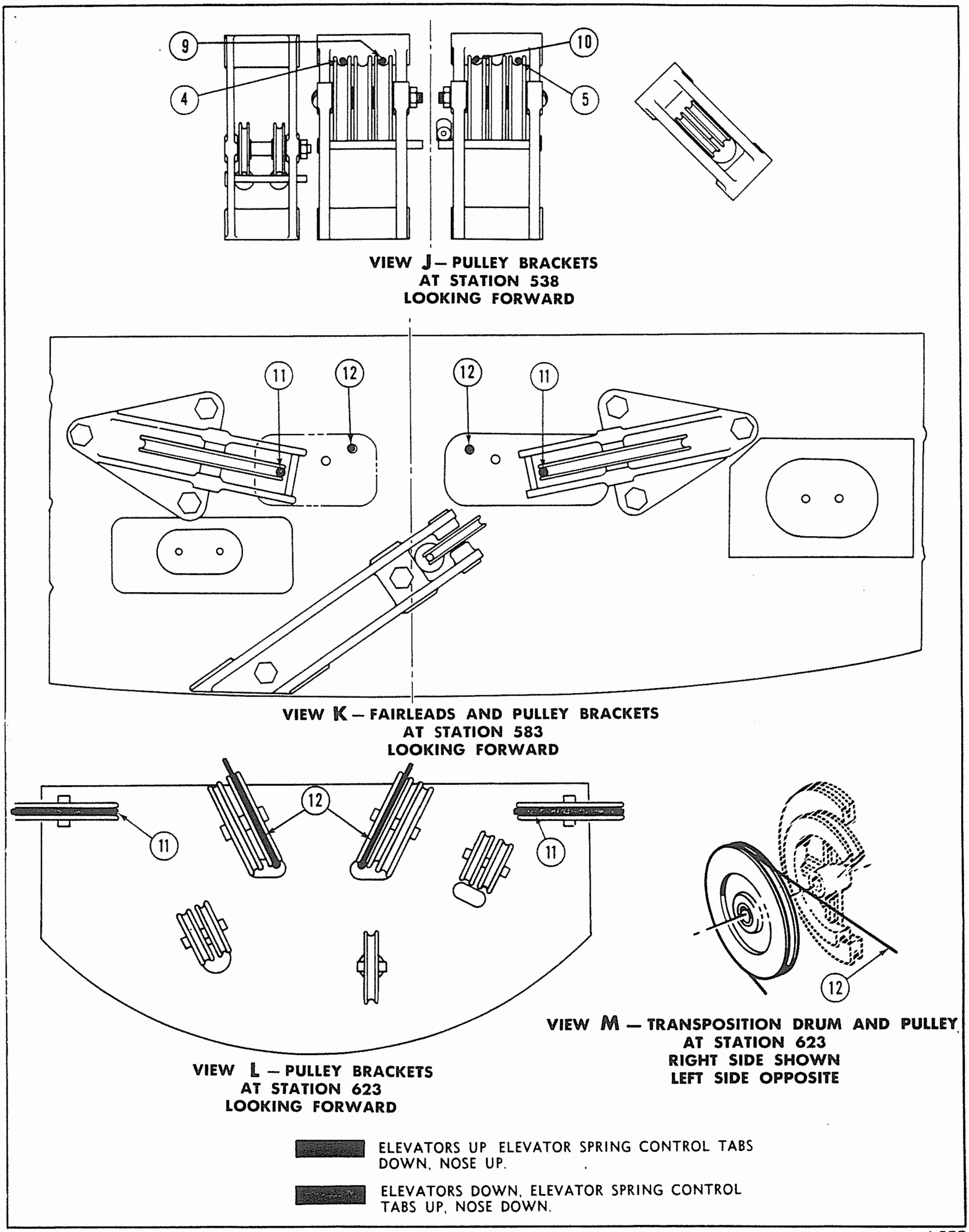

Figure 2-20 (Sheet 5 of 7 Sheets). Elevator and Elevator Spring Control Tab Control System — Pulley Brackets, Stations 538 and 623; Fairleads and Pulley Brackets, Station 583

ELEVATOR AND ELEVATOR SPRING CONTROL TAB CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)
1	3391412-545 Elev. Down	2	A	103 7/8	3/16 dia 7x19 flex	AN668-6	AN669L6 RH		
2	3391412-537 Elev. Up	1	B	29 1/2	3/16 dia 7x19 flex	AN668-6	AN669S6 LH		
3	3391412-543 Elev. Up	1	A	59 7/8	3/16 dia 7x19 flex	AN668-6	AN669L6 RH		
4	3391412-597 Elev. Down	1	C	485 1/2	3/16 dia 7x19 flex	AN669L6 LH	AN669L6 LH		
5	4398907-517 Elev. Down	1	D	(L_1) 486 (L_2) 96 1/2 (L_3) 23 3/4	3/16 dia 7x19 flex	AN669L6 LH	AN663-6	AN663-6	AN669L6 LH
6	4398907-521 Elev. Servo Bus	1	E	12 1/4	1/8 dia 7x19 flex	2391676	AN664-4	AN664-4	
7	4398907-525 Elev. Servo Jumper	1	F	14 1/2	1/8 dia 7x19 flex	AN664-4	AN669S4 LH		
8	4398907-523 Elev. Servo Jumper	1	G	6 3/4	1/8 dia 7x19 flex	AN669S4 RH	AN664-4	2391676	
9	4398907-501 Elev. Up	1	H	562	3/16 dia 7x19 flex	AN669L6 RH	AN669L6 LH		
10	3391412-555 Elev. Up	1	C	531 1/2	3/16 dia 7x19 flex	AN669L6 LH	AN669L6 LH		
11	3391412-599 Elev. Down	2	I	160	3/16 dia 7x19 flex	AN669L6 RH	AN667-6		
12	3391412-559 Elev. Up	2	I	156 3/4	3/16 dia 7x19 flex	AN669L6 RH	AN667-6		

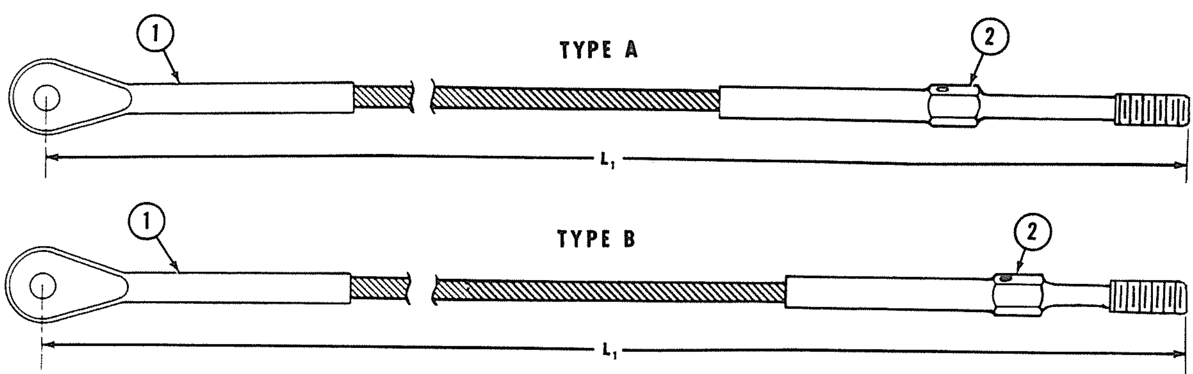

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Figure 2-20 (Sheet 6 of 7 Sheets). Elevator and Elevator Spring Control Tab Control System — Cable Chart and Cable Assemblies

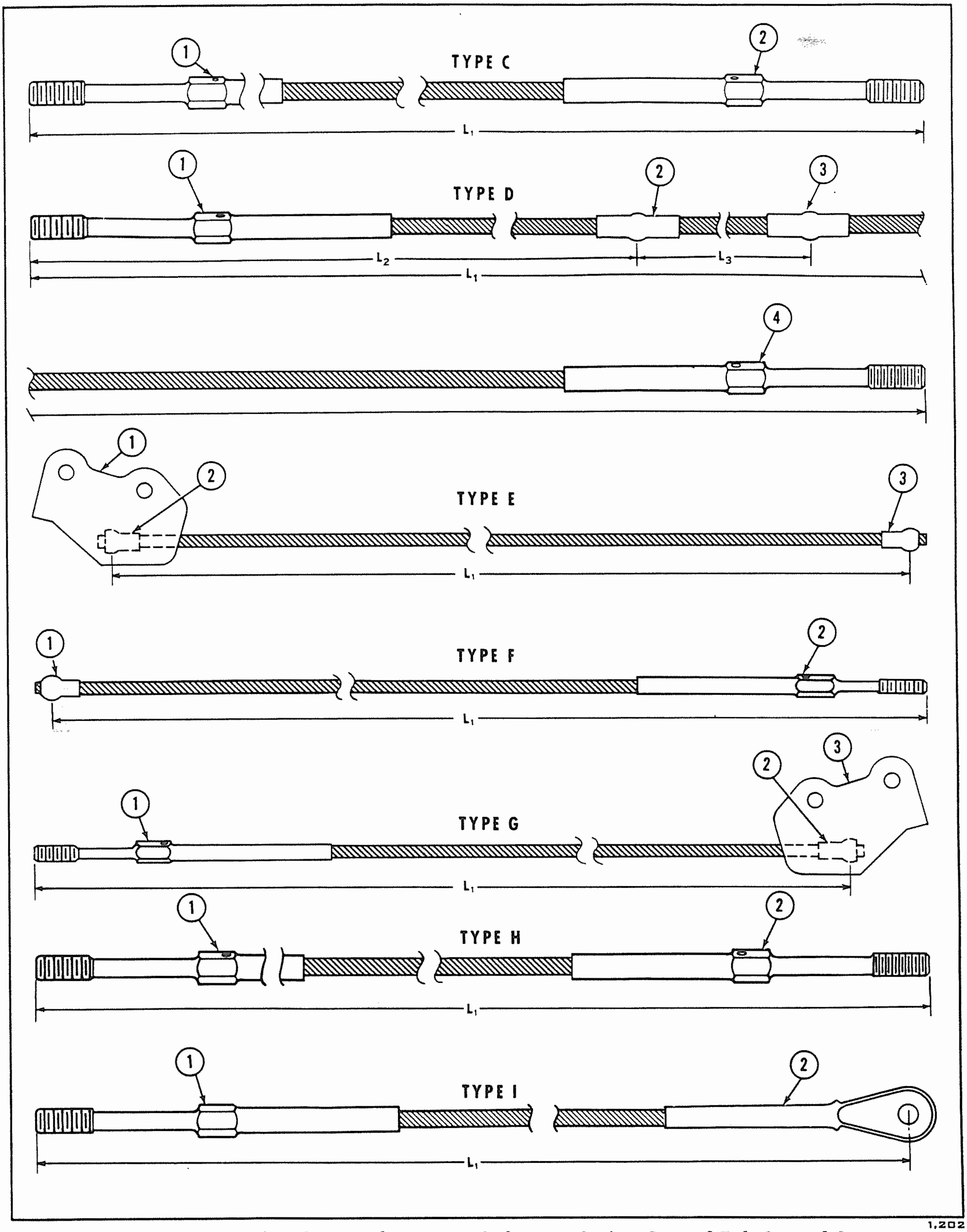


Figure 2-20 (Sheet 7 of 7 Sheets). Elevator and Elevator Spring Control Tab Control System — Cable Assemblies

(Continued from Page 124)

from the under side of the control column torque tube contains the attachment points for the UP and DOWN elevator cables. Column travel is limited by adjustable stops on the upper elbows of the columns and on the fuselage structure aft and in line with the control column horn. A control rod connects a down spring assembly to the control column torque tube crank.

2-125. REMOVAL OF CONTROL COLUMN ASSEMBLY.

a. Place ailerons and elevators in neutral and install surface locks on the surfaces.

b. Remove the pilot's and co-pilot's seats.

c. Remove the flooring in the flight compartment and companionway, aft to station 136.

d. Remove the landing gear mechanical safety latch control attached to the left floor channel *(see figure 2-48)*.

e. Remove the two floor channels in the flight compartment.

f. Remove the access door on the under side of the aircraft and disconnect the aileron control cables at approximately station 64. Thread the forward sections of the aileron cables and secure the aft sections.

g. Remove the eight aileron guide pulleys installed beneath the control column torque tube.

h. After disconnecting the aileron cables, turn the control column wheels to the left or right extreme and tie them to the column. This will secure the chains on the sprockets in the control column heads so that a chain cannot jump on the sprocket and throw one wheel out of synchronization with the other.

i. Release elevator cable tension between stations 47 and 80, and detach the elevator cables from the torque tube horn.

j. Remove the boots clamped to the bottom elbows of the columns to keep dust out of the mechanism.

k. Remove the hinge bolts that attach the control column assembly to the fuselage structure.

l. Lift out the entire control column assembly.

2-126. MINOR REPAIR AND REPLACEMENT OF CONTROL COLUMN. Replace any worn or damaged sprockets, bearings, pulleys, or roller chains.

2-127. INSTALLATION OF CONTROL COLUMN ASSEMBLY.

a. Place the aileron and elevator surfaces in neutral and install surface locks.

b. Place the control column assembly in position and install the hinge bolts that attach the control column to the fuselage structure.

c. Clamp the boots to the bottom elbows of the control columns.

d. Connect the elevator cables to the torque tube horn.

e. Release the control wheels and connect the aileron cables at approximately station 64.

f. Install the aileron guide pulleys beneath the control column torque tube.

g. Replace the two floor channels in the flight compartment.

h. Install the alighting gear mechanical safety latch control on the left floor channel *(see figure 2-48)*.

i. Tension the aileron and elevator control cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*. Tension elevator cables as follows: main elevator cables — rig load for $\frac{5}{32}$-inch diameter cables; automatic pilot servo unit pulley attach cables — rig load for $\frac{1}{8}$-inch diameter cables.

j. Remove all locking devices.

k. Adjust the following systems: aileron control system (see paragraph 2-90); elevator and elevator spring control tab control system (see paragraph 2-140); mechanical safety latch control system *(see figure 2-48)*.

l. Safety all turnbuckles.

2-128. CONTROL COLUMN DOWN-SPRING ASSEMBLY. A control column down-spring assembly, located under the floor on the left side of the flight compartment, assists in providing symmetrical forces for extreme cg conditions. A control rod connects the torque tube, at the aft end of the down-spring assembly, to the control column torque tube crank *(see figure 2-20)*.

2-129. REMOVAL OF CONTROL COLUMN DOWN-SPRING ASSEMBLY.

a. Remove the floor panel on the left side of the flight compartment.

b. Loosen the jam nuts at each end of the down-spring link-rod barrel and turn the barrel so that the rod ends move inward, releasing the tension on the down-spring.

c. Remove the bolt attaching the aft end of the link rod to the torque tube.

d. Remove the three bolts securing the forward end of the down spring to the aircraft structure and remove the down-spring assembly from the aircraft.

2-130. INSTALLATION OF CONTROL COLUMN DOWN-SPRING ASSEMBLY.

a. Place the down-spring assembly in position and install the three bolts securing it to the aircraft structure.

b. Bolt the aft end of the link rod to the torque tube. It may be necessary to adjust the link-rod barrel to align the holes of the torque tube and the link-rod eye.

c. Adjust the tension on the down-spring assembly (see paragraph 2-131).

d. Tighten the jam nuts at each end of the down-spring link-rod barrel and replace the floor panel.

2-131. ADJUSTMENT OF CONTROL COLUMN DOWN-SPRING ASSEMBLY.

a. Remove the floor panel on the left side of the flight compartment and loosen the jam nuts at each end of the down-spring link-rod barrel.

b. Attach a tension scale to the hub of the control wheel.

c. Adjust the down-spring link-rod barrel to obtain a 30-pound tension against the rearward pull of the control column.

d. Tighten the jam nuts and replace the floor panel.

2-132. ELEVATOR AUTOMATIC PILOT PULLEY. The elevator automatic pilot pulley is located just to the right of the center line of the aircraft, below the floor at station 186. For information concerning the pulley, see paragraphs 6-82 through 6-84.

2-133. ELEVATOR TORQUE TUBE ASSEMBLY. *(See figure 2-20.)* The elevator torque tube assembly in the tail section of the aircraft consists of a bus torque tube (elevator) and two actuating torque tubes (spring control tab), to which the two elevator-operating bellcranks are bolted. Each bellcrank is operated by a separate two-way cable system actuated by the control column torque tube horn. The bus torque tube rigidly interconnects the two elevator surfaces. Elevator movement is limited by contact of the elevator torque tube fitting with adjustable stops mounted on the tail section structure, forward of the torque tube assembly. Spring control tab stops on the bus torque tube are contacted by shoulders on the elevator-operating cranks and limit the degree of travel of the elevator spring control tabs.

2-134. REMOVAL OF ELEVATOR TORQUE TUBE ASSEMBLY.

a. Support the elevators and elevator trim tabs in neutral.

b. Remove the tail cone of the aircraft (see paragraph 2-338).

c. Disconnect the elevator cable turnbuckles at stations 571 and 612 in the tail section and disconnect the cables from the elevator-operating cranks. It is not necessary to remove the cables from the aircraft.

d. Remove the bolts attaching the bus torque tube to the elevators. These bolts are accessible from access doors in the elevators.

e. Disconnect the elevator spring control tab torque tubes from the spring tab mechanisms in the elevators.

f. Remove the bolt connecting the spring control tab actuating torque tube fittings from the elevator structure supports.

g. Remove the bolts attaching the bus torque tube and the elevator spring tab torque tubes to the elevator-operating cranks.

h. Remove the cranks and the spring control tab torque tubes.

i. Release the hinge bolts that attach the elevator bus torque tube to the structure in the tail section.

j. Remove the bus torque tube from the aircraft.

2-135. MINOR REPAIR AND REPLACEMENT OF ELEVATOR TORQUE TUBE ASSEMBLY. Replace any worn or damaged bearings.

2-136. INSTALLATION OF ELEVATOR TORQUE TUBE ASSEMBLY.

a. Secure the control columns in neutral and support the elevators in the faired position.

b. Place the elevator bus torque tube in position and install the hinge bolts that attach the elevator bus torque tube to the structure in the tail section.

c. Place the elevator operating cranks and the elevator spring tab torque tubes in position and bolt the bus torque tube and the elevator spring tab torque tubes to the cranks.

d. Connect the spring control tab torque tube fittings to the elevator structure supports.

e. Connect the elevator spring control tab torque tubes to the spring tab mechanisms in the elevators.

f. Bolt the bus torque tube to the elevators.

g. Attach the elevator cables to the elevator operating cranks. Connect the elevator cable turnbuckles at stations 571 and 612 in the tail section.

h. Install the tail cone of the aircraft (see paragraph 2-339).

i. Tension all elevator cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

j. Remove the surface supports.

k. Adjust the elevator control system (see paragraph 2-140).

l. Safety the turnbuckles.

2-137. ELEVATOR CONTROL CABLES. The elevator control cable system consists of two two-way cable systems that originate at the control column horn and

terminate at the two cranks attached to the torque tube assembly in the tail section *(see figure 2-20).*

2-138. REMOVAL OF ELEVATOR CONTROL CABLES.

a. Place the elevators in neutral and install surface locks.

b. Open necessary access doors on the under side of the aircraft and disconnect elevator cable turnbuckles between stations 47 and 80. Thread all cable ends.

c. Detach the elevator cables from the control column torque tube horn.

d. Remove the guard pins from bracket at station 37 and the pulleys from the bracket at station 2. Draw the forward elevator cables forward and out of the aircraft.

e. Loosen the turnbuckle connecting the elevator servo jumper cables at approximately station 193. Release the cables from the elevator automatic pilot pulley. Unfasten the bus cable fittings and remove the elevator servo cables from the aircraft.

f. Disconnect the elevator trailing edge down cables at station 571 and the elevator trailing edge up cables at station 612. Thread the aft cable ends.

g. Remove guard pins and grommets and draw the intermediate elevator cables aft. Remove the cables.

h. Disconnect the aft cables from the elevator-operating cranks at the torque tube assembly, remove guard pins, and draw cables aft and out of the tail section.

2-139. INSTALLATION OF ELEVATOR CONTROL CABLES.

a. Lock the control columns in neutral (10 degrees on aircraft A, B, and 1 through 17, or 11½ degrees on aircraft C, D, 18 and subsequent, forward of vertical).

b. Install surface locks on the elevators.

c. Connect the aft elevator control cables to the elevator-operating cranks at the torque tube assembly, and route them forward through the tail section of the aircraft, replacing necessary guard pins and fairlead grommets.

d. Route the intermediate elevator control cables forward through the fuselage, replacing guard pins and fairlead grommets, and connect the cables with turnbuckles at stations 571 and 612.

e. Install the elevator servo jumper cables at the elevator automatic pilot servo and connect them to the automatic pilot pulley. Connect the bus cable fittings to the intermediate elevator control cable.

f. Attach the forward elevator cables to the control column torque tube horn and route them aft through the aircraft. Install the pulleys in the brackets at station 2. Replace necessary guard pins.

g. Connect the cables with turnbuckles between stations 47 and 80.

h. Tension all cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

i. Remove the surface locks and adjust the elevator control system (see paragraph 2-140).

j. Install the access doors on the under side of the fuselage.

2-140. ADJUSTMENT OF ELEVATOR CONTROL SYSTEM.

Note

Position the control columns so that the spring control tabs are faired with the elevators. Mark points in line on the inboard corners of the spring control tabs and on the outboard corners of the inboard trailing sections of the elevators for measuring elevator spring control tab travel. Measure between these points for NEUTRAL, or extreme UP or DOWN position.

a. Secure the control columns in neutral, 10 degrees (aircraft A, B, and 1 through 17), or 11½ degrees (aircraft C, D, 18 and subsequent), forward of vertical.

b. Set the elevators in neutral (faired with the tail cone). Adjust the upper screws at the elevator-operating cranks, in the tail section, to ⅝ inch as shown in detail on sheet 2, figure 2-20. Adjust the lower screws to lock the crank.

Note

Mark points in line on the elevators and on the tail cone for measuring elevator travel. Measure between these points for extreme UP or DOWN position.

c. Check that the cable tension agrees with that specified in the Cable Rigging Tension Chart *(see figure 2-9)*, as follows: main elevator cables — rig load for 5/32-inch diameter cables; automatic pilot servo unit pulley attach cables — rig load for ⅛-inch diameter cables.

d. Adjust the spring control tab mechanisms until the spring tab trailing edges are UP 5 (±½) degrees or ¾ (±⅛) inch (neutral position).

e. Run in the adjusting screws in the elevator adjustment stops on the tail section structure.

f. Release the control columns and check for unrestricted travel to the full aft position of the columns (stops contacted by the screws on the upper column elbows).

g. Check for unrestricted travel to the full forward position of the columns (when stop, compressed by control column horn, bottoms).

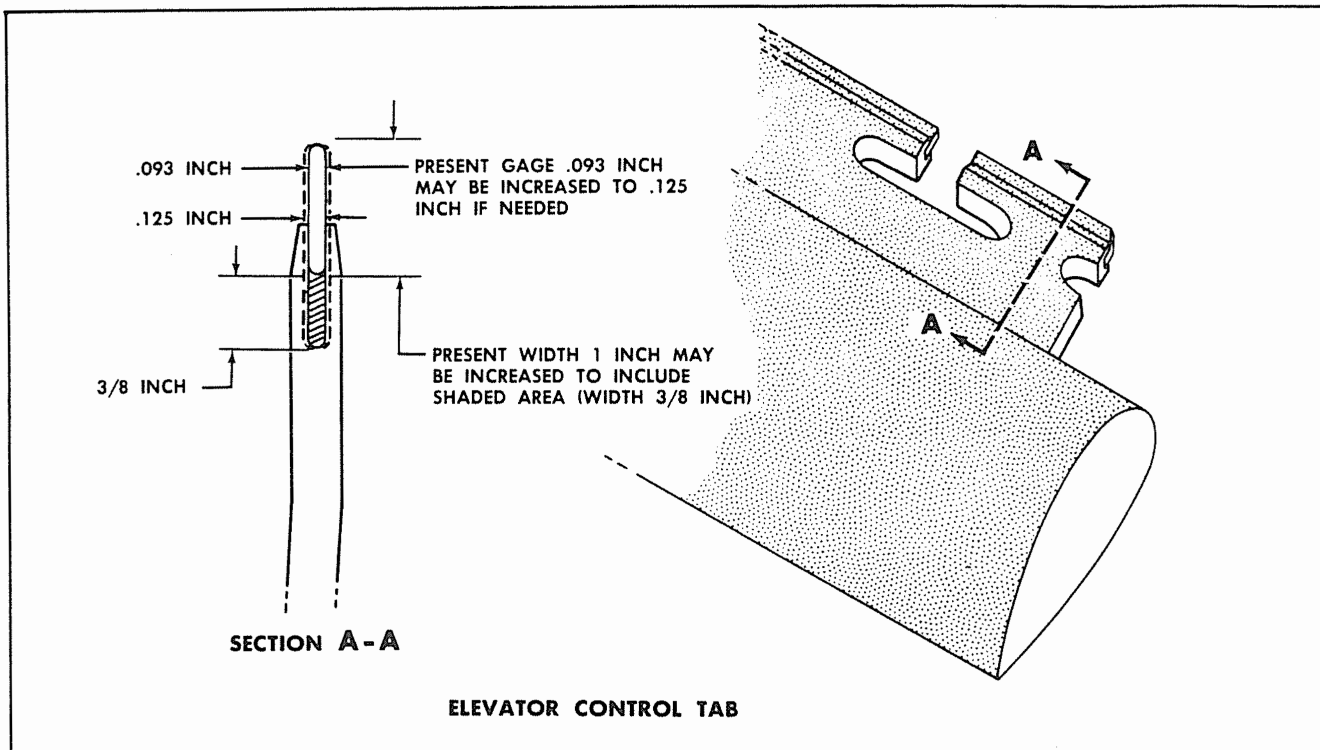


Figure 2-21. Balancing Elevator Spring Control Tab

h. Place the control columns in the full forward position, 24 degrees (aircraft A, B, and 1 through 17), or approximately 25½ degrees (aircraft C, D, 18 and subsequent), forward of vertical (elevators DOWN). Adjust the upper elevator adjustment stop bolt on the tail section structure to allow a DOWN throw, for the elevators, of 20 (±½) degrees or 10¹³⁄₃₂ (±¼) inches from neutral, measured at the inboard corners of the elevators.

i. With the elevators in the full DOWN position, adjust the upper screws at the elevator-operating cranks to allow an UP throw, for the spring control tabs, of 20 (±½) degrees or 3 (±⅛) inches (aircraft A, B, and 1 through 17), or 18 (±½) degrees or 2¾ (±⅛) inches (aircraft C, D, 18 and subsequent), measured at the inboard corners of the spring control tabs.

j. Place the control columns in the full aft position, 5 degrees (aircraft A, B, and 1 through 17), or approximately 3½ degrees (aircraft C, D, 18 and subsequent), aft of vertical (elevators UP). Adjust the lower elevator adjustment stop bolt on the tail section structure to allow an UP throw, for the elevators, of 20 (±½) degrees or 10¹³⁄₃₂ (±¼) inches from neutral, measured at the inboard corners of the elevators.

k. With the elevators in the full UP position, adjust the lower screws at the elevator-operating cranks to allow a DOWN throw, for the spring control tabs, of 10 (±½) degrees or 1½ (±⅛) inches, measured at the inboard corners of the spring tabs.

l. If correct movements or distances are not achieved, it will be necessary to check the rigging, and also the adjustment of the spring control tab mechanism.

m. Recheck cable tension to make sure that no change has resulted from system adjustments.

n. Adjust the down spring to give approximately 30 pounds nose-down load on the control column with the column in neutral position.

o. With the system rigged and unloaded, the control columns should go forward, giving full DOWN elevator. The stop contacted by the control column horn should hold the column in position to give approximately five degrees spring tab trailing edge UP.

p. Safety the turnbuckles.

q. For further information on rigging the elevator control system, see paragraph 2-231.

2-141. ELEVATOR SPRING CONTROL TAB. An elevator spring control tab, constructed of conventional metal ribs and skin, is attached by hinge fittings to the inboard trailing edge of each elevator to provide aerodynamic boost. A cut-out for the tab is provided in each elevator. The tabs are actuated by the control rods from the spring control tab mechanisms in the elevators.

2-142. REMOVAL OF ELEVATOR SPRING CONTROL TAB.

(See figure 2-21.)

a. Disconnect the spring control tab actuating rod from the tab.

b. Remove the stop nuts from the eyebolt studs at the center and outboard hinges and the bolt from the inboard hinge fitting.

c. Remove the spring control tab.

2-143. BALANCING OF ELEVATOR SPRING CONTROL TAB.
(See figure 2-21.)

a. Disconnect the elevator spring control tab actuating rod at the point where it attaches to the tab. Be sure the tab can swing freely.

b. Place a one-pound weight on the tab and move it forward and aft until the surface is horizontal. Measure the distance from the trailing edge of the tab to the center of the weight. This distance must be between 3 21/32 and 8 21/32 inches for the tab to be correctly balanced.

c. If the distance is more than 8 21/32 inches, remove the steel strips on the nose assembly of the tab (Douglas drawing 5371955) and replace them with steel strips 1 3/8 inches wide, 0.093 gage. Repeat step b.

d. If the distance is still more than 8 21/32 inches, replace the steel strips on the nose assembly of the tab with steel strips 1 3/8 inches wide, 0.125 gage. This will necessitate milling the groove to fit the new strips. Repeat step b.

e. If the distance is less than 3 21/32 inches, cut off 1/4 inch from the leading edge of the steel strips on the nose assembly of the tab. Repeat step b.

f. If the distance is still less than 3 21/32 inches, cut off another 1/4 inch from the steel strips on the nose assembly of the tab and repeat step b.

Note

When adding strips to the control tab, care must be exercised in order that the added portions do not interefere with the full required travel of the tab (20 degrees UP and 10 degrees DOWN). Whenever weight is added or removed from the elevator spring control tabs for balancing, the elevators must be rebalanced in accordance with the elevator balancing procedure (see paragraphs 2-119 or 2-120).

2-144. ELEVATOR SPRING CONTROL TAB HINGE WEAR TOLERANCES. Due to the close tolerance fits of the elevator spring control tab hinge bolts and bearings, the maximum wear limit is 0.001 inch in excess of the manufacturing tolerance between the two parts.

2-145. INSTALLATION OF ELEVATOR SPRING CONTROL TAB.
(See figure 2-13.)

a. Place the spring control tab in position at the elevator. Install the bolt at the inboard hinge fitting and the stop nuts on the eyebolt studs at the center and outboard hinges.

b. Connect the spring control tab actuating rod to the tab.

c. Check the adjustment of the elevator spring control tab control system (see paragraph 2-151).

2-146. ELEVATOR SPRING CONTROL TAB CONTROL SYSTEM. *(See figure 2-20.)* In flight, leverage produced by the force of air against the spring control tab in the inboard section of each elevator provides aerodynamic assistance for control of the elevators. Movement of the control columns actuates the spring control tabs (by means of the same cable system and bellcranks that operate the elevators) in an opposite direction to the intended movement of the elevators. With no airload on the surface (aircraft in hangar), movement of the controls in the flight compartment will move the main surface only until that surface meets its stops. Further movement of the controls in the flight compartment will then deflect the tabs in an opposite direction to the main surfaces against the spring load in the tab mechanisms.

2-147. ELEVATOR SPRING CONTROL TAB MECHANISMS. *(See figure 2-20.)* A preloaded spring tab mechanism, located at station 36 in each elevator, augments "feel" of elevator force and aids in returning the spring control tab to the neutral position. At any time during flight, the position of the spring control tab is the result of the force of the spring and the force of the airloading on the elevator surfaces and spring tabs.

2-148. REMOVAL OF ELEVATOR SPRING CONTROL TAB MECHANISM.

a. Place the control columns in neutral and support the spring control tab.

b. Open access doors in the elevator and unbolt the elevator link tab spring support assembly, of the spring tab mechanism, from the spring control tab torque tube attachment and from the support.

c. Disconnect the tab actuating rod from the elevator link tab spring support assembly.

d. Remove the tab fairing and disconnect the bolt securing the actuating rod at the inboard tab actuating hinge. Remove the rod.

e. Back off the adjustment bolts on the spring tab mechanism to reduce compression on the spring.

f. Unbolt the spring tab mechanism from the structure of the elevator.

g. Remove the mechanism.

2-149. MINOR REPAIR AND REPLACEMENT OF ELEVATOR SPRING CONTROL TAB MECHANISM. Replace the spring if weak or broken and replace any damaged or broken bearings.

2-150. INSTALLATION OF ELEVATOR SPRING CONTROL TAB MECHANISM.

a. Place the control columns in neutral and support the spring control tab in the neutral position.

b. Bolt the spring tab mechanism to the structure of the elevator.

c. Connect the actuating rod to the inboard tab actuating hinge and install the tab fairing.

d. Connect the tab actuating rod to the elevator link tab spring support assembly.

e. Connect the elevator link tab spring support assembly to the spring control tab torque tube attachment and to the support.

f. Remove the tab support.

g. Adjust the turnbuckle at the short pushrod until the spring tab mechanism contacts the structural stop below the spring of the mechanism.

h. Regulate adjustment bolts to a contact position and to provide a preload in the spring amounting to ½ to 1 pound force at the spring tab actuating rod inboard attach point.

i. Use fore and aft actuating rod adjustment to position the tab in neutral.

j. Adjust the spring control tab control system. Adjustment of the elevator spring control tab system must be accomplished in conjunction with the adjustment of the elevator control system (see paragraph 2-140).

2-151. ADJUSTMENT OF ELEVATOR SPRING CONTROL TAB CONTROL SYSTEM. Adjustment of the elevator spring control tab control system must be accomplished in conjunction with the adjustment of the elevator control system (see paragraph 2-140).

2-152. ELEVATOR TRIM TABS. An elevator trim tab, constructed of conventional metal ribs and skin, is attached by hinge fittings to a cut-out in the trailing edge of each elevator, outboard of the spring control tab, to provide a means of balancing nose-heavy and tail-heavy conditions of the aircraft. The elevator trim tabs are controlled by a single control wheel, mounted on the left side of the pedestal head.

2-153. REMOVAL OF ELEVATOR TRIM TAB. *(See figure 2-22.)*

a. Disconnect the aft trim tab control rod from the trim tab horn.

b. Remove the bolts from the hinge fittings.

c. Remove the trim tab.

2-154. BALANCING OF ELEVATOR TRIM TAB. Balancing of the elevator trim tab is not required.

2-155. ELEVATOR TRIM TAB HINGE WEAR TOLERANCES. Due to the close tolerance fits of the elevator trim tab hinge bolts and bearings, the maximum wear limit is 0.001 inch in excess of the manufacturing tolerance between the two parts.

2-156. INSTALLATION OF ELEVATOR TRIM TAB. *(See figure 2-22.)*

a. Place the elevator trim tab in position and install the bolts in the hinge fittings.

b. Connect the aft trim tab control rod to the trim tab horn.

c. Check the adjustment of the elevator trim tab control system (see paragraph 2-170).

2-157. ELEVATOR TRIM TAB CONTROL SYSTEM. *(See figure 2-22.)* The elevator trim tabs are controlled by a single control wheel mounted on the left side of the control pedestal head in the flight compartment. Operation of the trim control wheel actuates a cable drum mounted on the same shaft in the pedestal head. The movement is transmitted by the control drum through a two-way cable system extending aft to the trim tab drive mechanism drums in the horizontal stabilizers. Pushrods transmit the movement to the trim tabs. An automatic pilot servo unit is connected to the trim tab control cables at station 225.

2-158. ELEVATOR TRIM TAB CONTROL WHEEL, INDICATOR, AND DRUM. *(See figure 2-22.)* The elevator trim tab control wheel and cable drum are mounted on a common shaft on the left side of the control pedestal head. A trim tab position indicator is geared to the shaft.

2-159. REMOVAL OF ELEVATOR TRIM TAB CONTROL WHEEL, INDICATOR, AND DRUM.

a. Tape cable and drums in pedestal and stabilizers to prevent them from unwrapping.

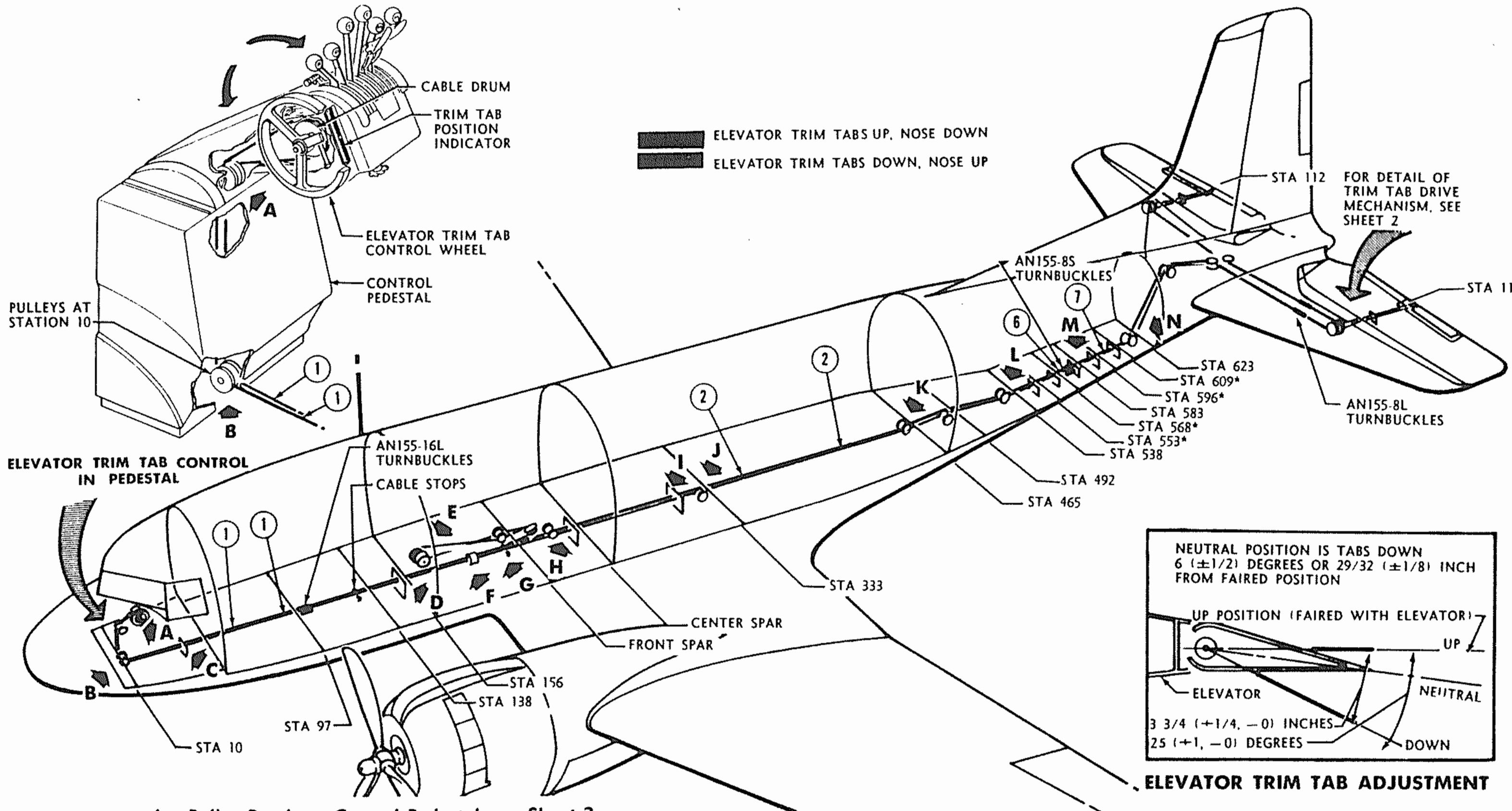
b. Turn the elevator trim tab control wheel until the indicator reads zero degrees. In this position the tabs are DOWN 6 (± ½) degrees (neutral). Lock the surfaces in this position.

c. Open the access doors on the under side of the aircraft and disconnect the elevator trim tab cable turnbuckles at station 102. Thread the forward cables and tape the aft cables to the structure.

d. Remove necessary grommets and guard pins to allow removal of the forward elevator trim tab control cables.

e. Unbolt the pedestal head cover assembly and lift it away from the pedestal, using caution, as the trim tab control drum is attached to the shaft. Draw the forward trim tab control cables out of the pedestal along with the cover assembly, trim control wheel, indicator, and drum.

(Continued on Page 147)



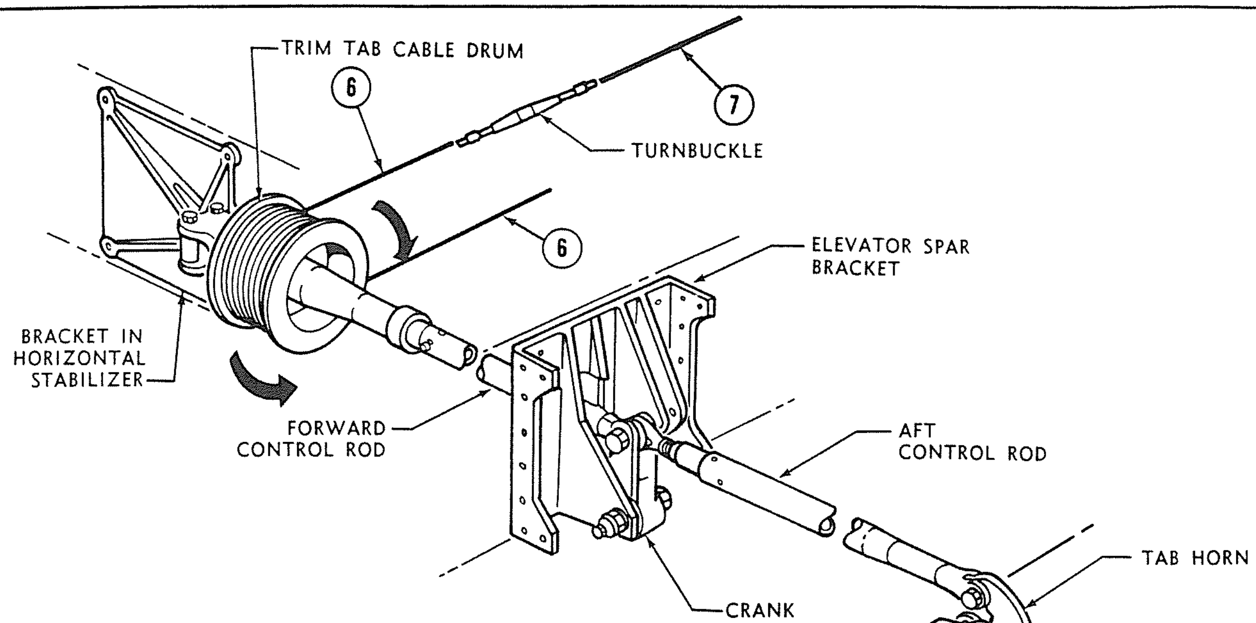
A—Pulley Bracket—Control Pedestal, see Sheet 2
B—Pulley Bracket—Station 10, see Sheet 3
C—Fairlead—Station 24, see Sheet 3
D—Fairleads—Station 156, see Sheet 4
E—Automatic Pilot Rigging, see Sheet 8
F—Pulley Brackets—Station 203, see Sheet 4
G—Fairleads and Pulley Brackets—Station 222, see Sheet 5
H—Fairleads—Station 258, see Sheet 5
I—Fairleads—Station 324, see Sheet 6
J—Pulley Brackets—Station 333, see Sheet 6
K—Pulley Brackets—Station 465, see Sheet 6
L—Pulley Brackets—Station 538, see Sheet 6
M—Fairleads and Pulley Brackets—Station 583, see Sheet 7
N—Pulley Brackets—Station 623, see Sheet 7

Notes:

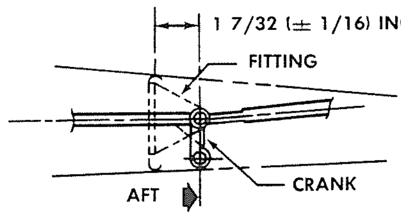
1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 8.
2. For cable assembly details, see Sheets 8 and 9.
*3. Fairleads shown at these stations are installed on aircraft A through D.
4. Tension cables in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values shown in Torque Value Chart, paragraph 2-453.
6. Reference—Douglas drawings 5372516, 5372517, 5372519, 5372520, 5398252, 5392557 and 5400407.

1,203

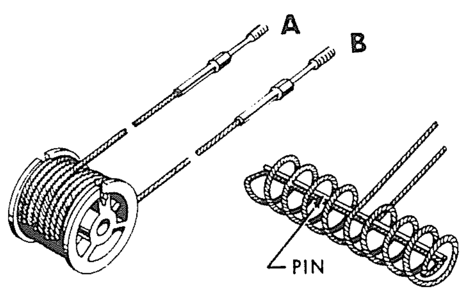
Figure 2-22 (Sheet 1 of 9 Sheets). Elevator Trim Tab Control System — Key Drawing



ELEVATOR TRIM TAB DRIVE MECHANISM (LEFT SHOWN — RIGHT SIMILAR)

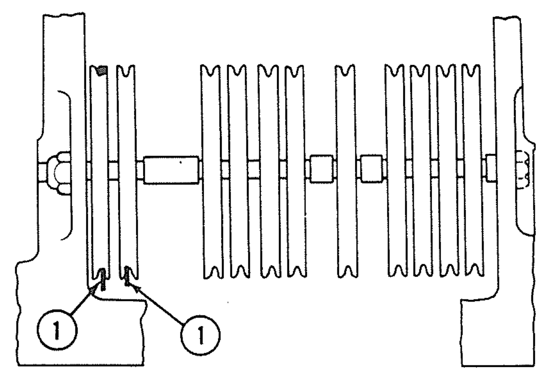


TRIM TAB CONTROL RODS 7 DEGREES NOSE UP POSITION DIAGRAM



ELEVATOR TRIM TAB DRUM WRAPPING PROCEDURE

1. Insert cable and pin. Wrap cable on drum from each end to center with equal number of turns on each side (typical).
2. Wrap drum in control pedestal so that equal lengths project at "A" and "B."
3. Wrap drum in left horizontal stabilizer so that cable "B" is 218 1/4 inches longer than cable "A."
4. Wrap drum in right horizontal stabilizer so that cable "A" is 63 1/4 inches longer than cable "B."



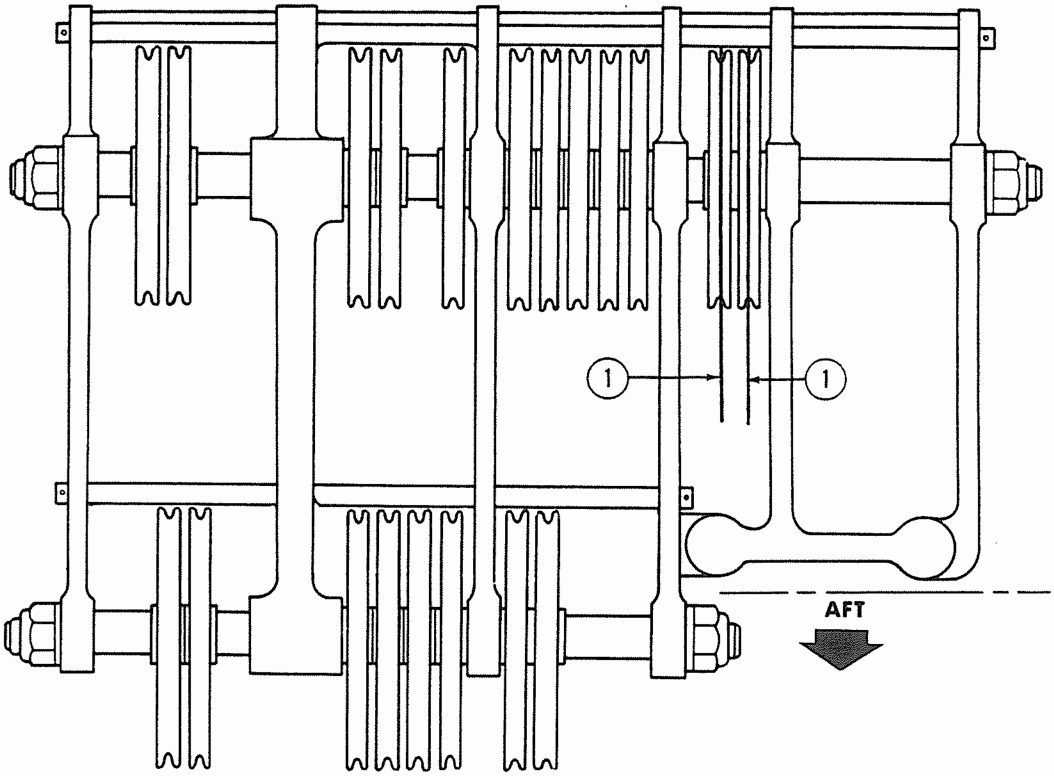
VIEW A — PULLEY BRACKET IN CONTROL PEDESTAL — LOOKING FORWARD

ELEVATOR TRIM TABS UP, NOSE DOWN

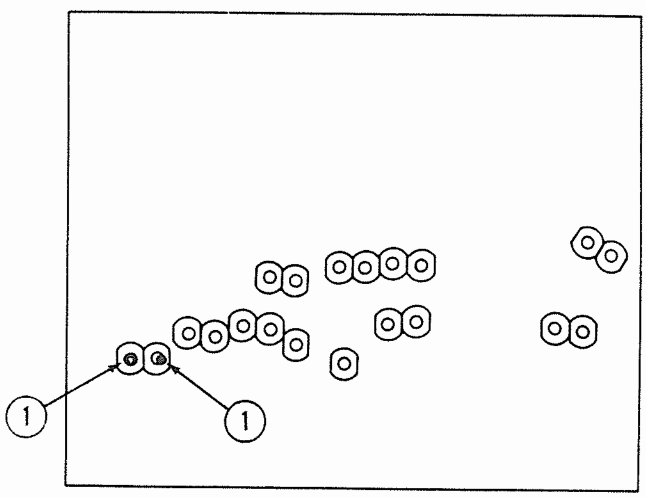
ELEVATOR TRIM TABS DOWN, NOSE UP

10,361

Figure 2-22 (Sheet 2 of 9 Sheets). Elevator Trim Tab Control System — Pulley Bracket, Control Pedestal, Trim Tab Mechanism



VIEW B PULLEY BRACKET AT STATION 10
(UNDER CONTROL PEDESTAL) — BOTTOM VIEW



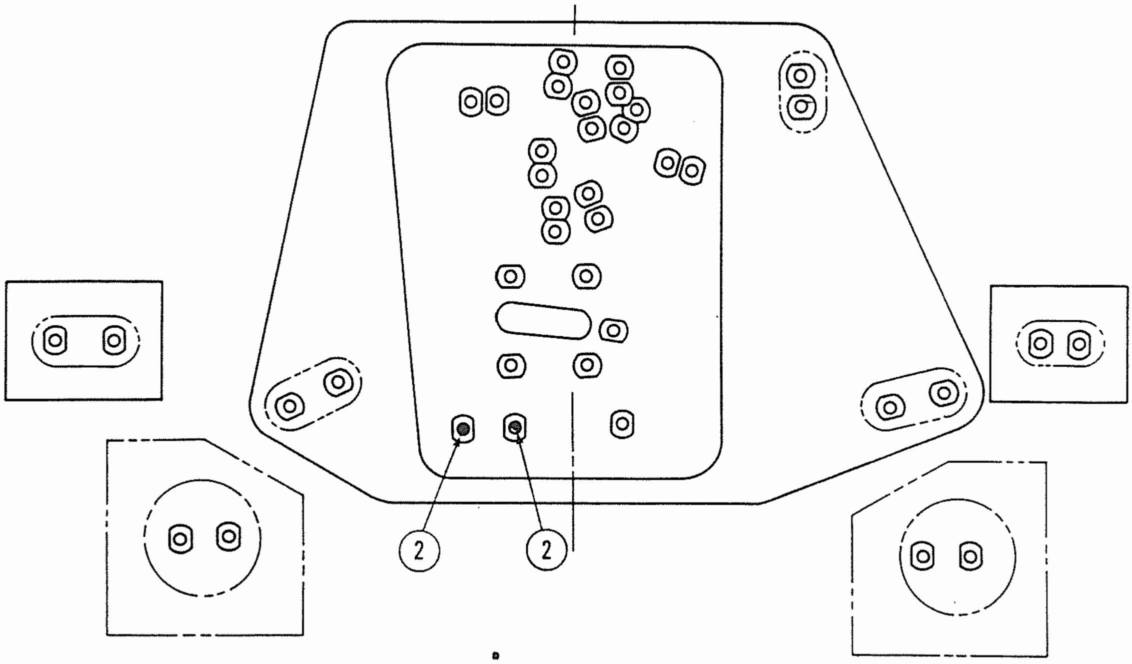
VIEW C — FAIRLEAD AT
STATION 24 — LOOKING FORWARD

ELEVATOR TRIM TABS UP, NOSE DOWN

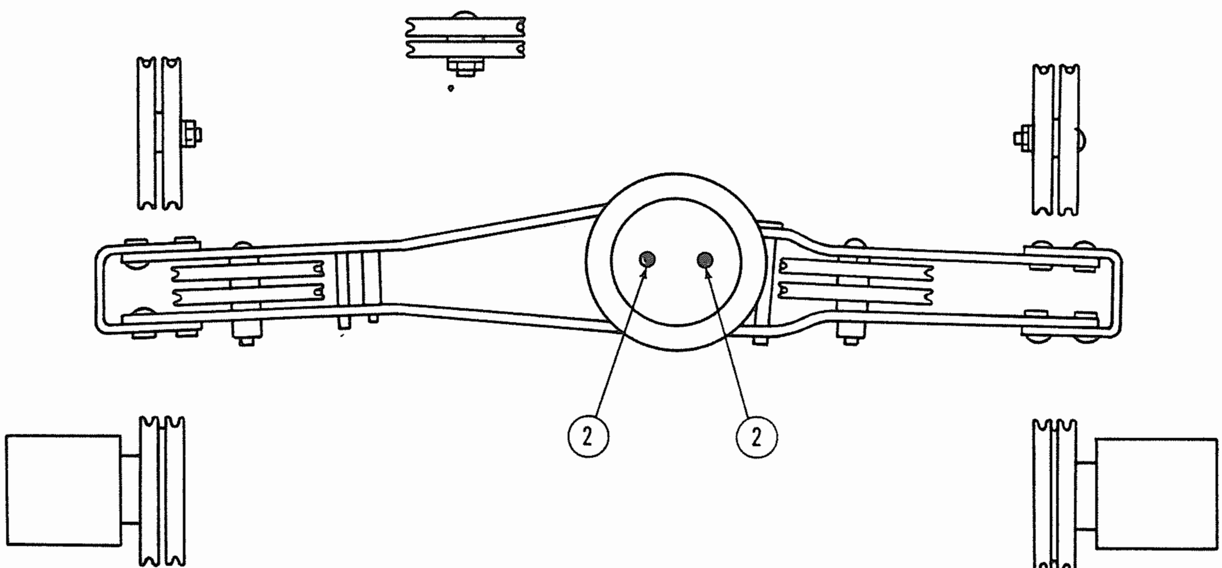
ELEVATOR TRIM TABS DOWN, NOSE UP

1,205

Figure 2-22 (Sheet 3 of 9 Sheets). Elevator Trim Tab Control System — Pulley Bracket, Station 10; Fairlead, Station 24



VIEW D — FAIRLEADS AT
STATION 156 — LOOKING FORWARD



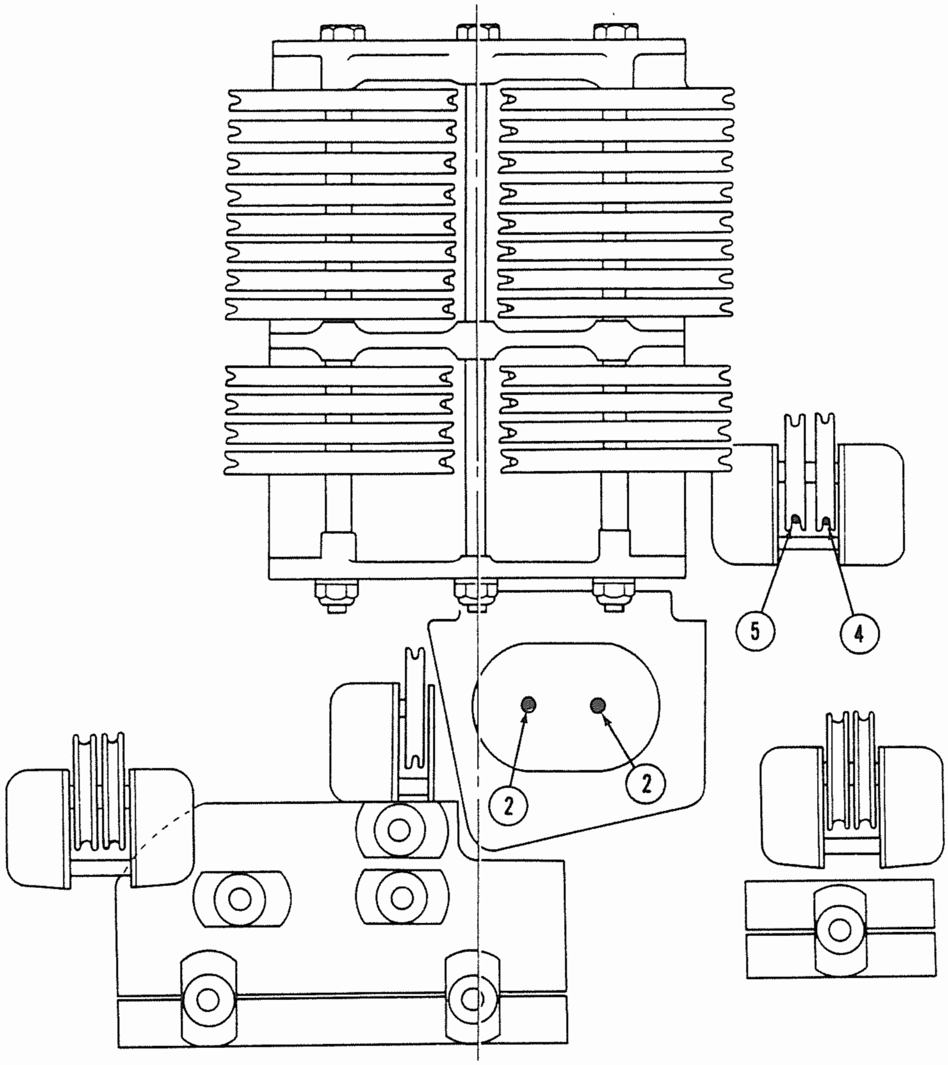
VIEW F — PULLEY BRACKETS AT
STATION 203 — LOOKING AFT

ELEVATOR TRIM TABS UP, NOSE DOWN

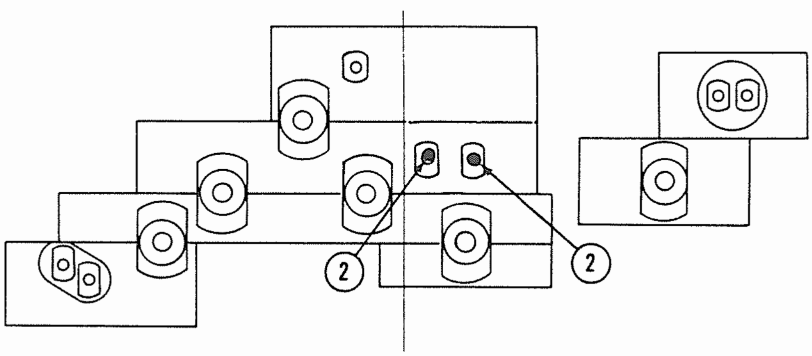
ELEVATOR TRIM TABS DOWN, NOSE UP

1,206

Figure 2-22 (Sheet 4 of 9 Sheets). Elevator Trim Tab Control System — Fairleads, Station 156; Pulley Brackets, Station 203



VIEW G — FAIRLEADS AND PULLEY BRACKETS
AT STATION 222 (FRONT SPAR) — LOOKING AFT



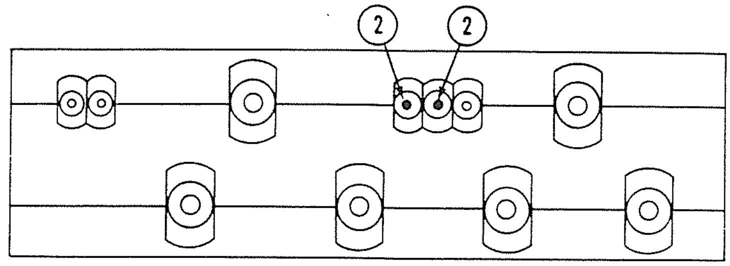
VIEW H — FAIRLEADS AT STATION 258
(CENTER SPAR) — LOOKING AFT

ELEVATOR TRIM TABS UP, NOSE DOWN

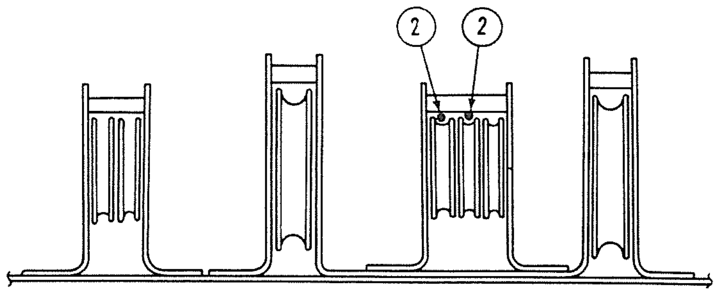
ELEVATOR TRIM TABS DOWN, NOSE UP

1,207

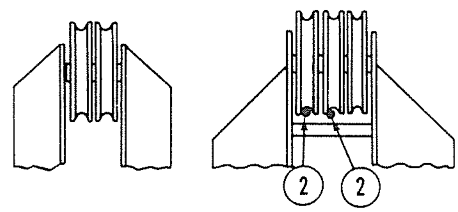
Figure 2-22 (Sheet 5 of 9 Sheets). Elevator Trim Tab Control System — Fairleads and Pulley Brackets, Station 222; Fairleads, Station 258



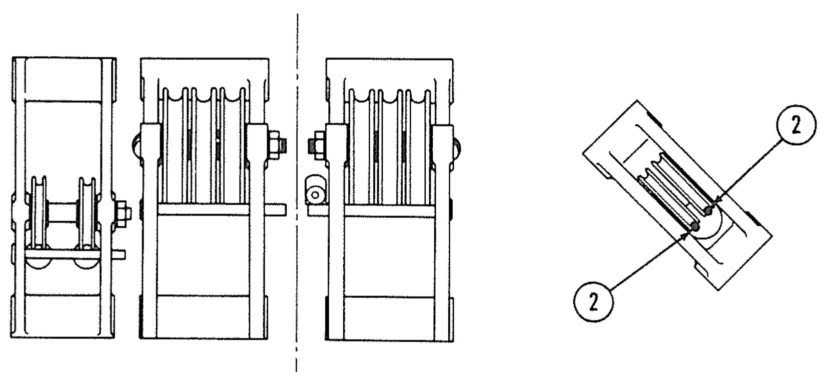
VIEW I — FAIRLEAD AT
STATION 324 — LOOKING FORWARD



VIEW J — PULLEY BRACKETS AT
STATION 333 — LOOKING FORWARD



VIEW K — PULLEY BRACKETS
AT STATION 465
LOOKING FORWARD



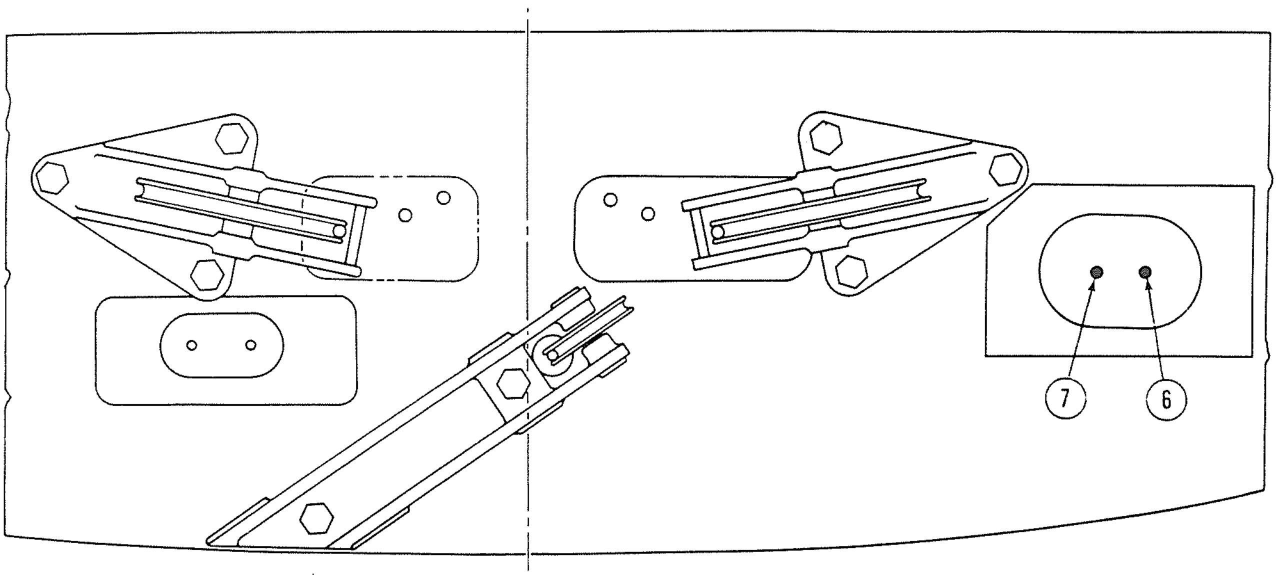
VIEW L — PULLEY BRACKETS AT
STATION 538 — LOOKING FORWARD

ELEVATOR TRIM TABS UP, NOSE DOWN

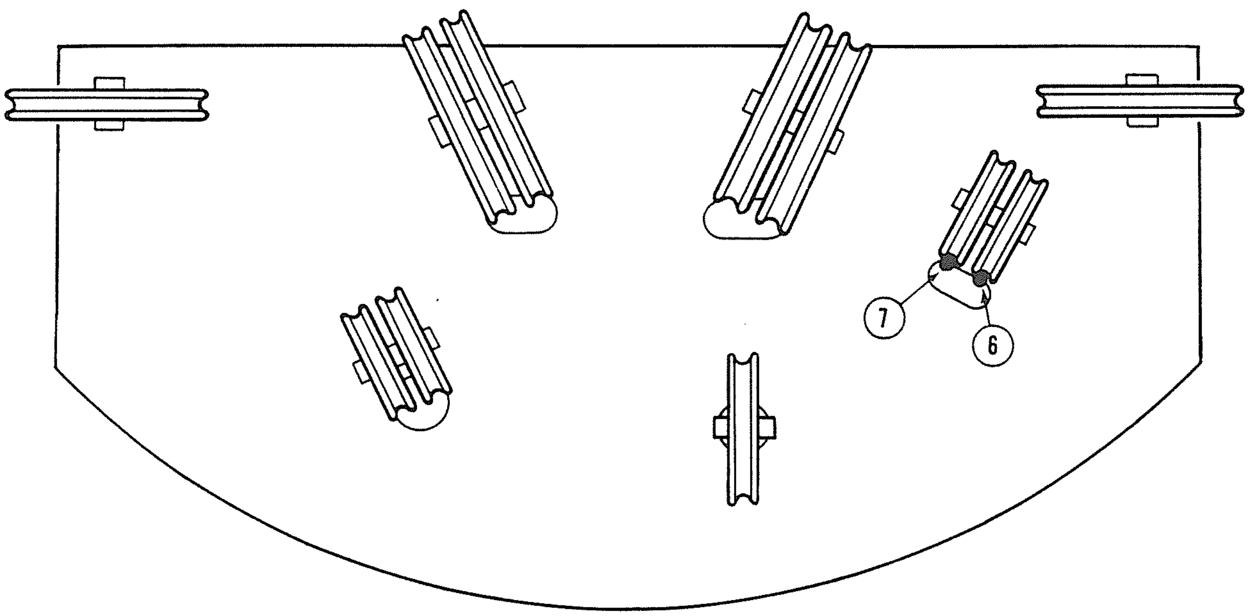
ELEVATOR TRIM TABS DOWN, NOSE UP

1,208

Figure 2-22 (Sheet 6 of 9 Sheets). Elevator Trim Tab Control System — Fairlead, Station 324; Pulley Brackets, Stations 333, 465, and 538



VIEW M — FAIRLEADS AND PULLEY BRACKETS AT STATION 583 — LOOKING FORWARD



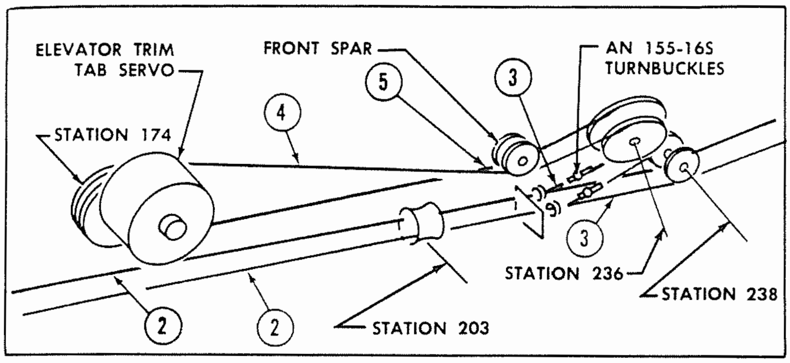
VIEW N — PULLEY BRACKETS AT STATION 623 — LOOKING FORWARD

ELEVATOR TRIM TABS UP, NOSE DOWN

ELEVATOR TRIM TABS DOWN, NOSE UP

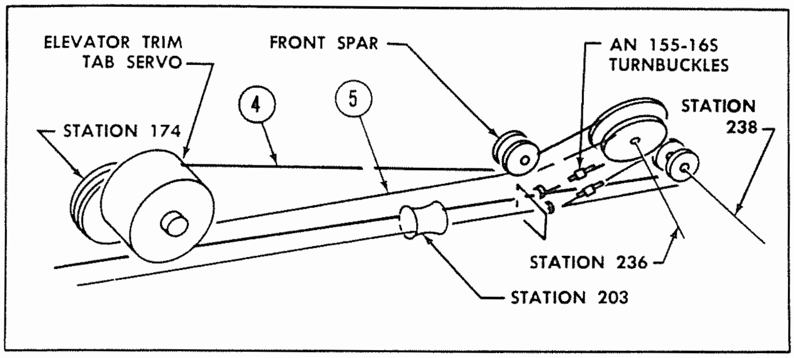
1,209

Figure 2-22 (Sheet 7 of 9 Sheets). Elevator Trim Tab Control System — Fairleads and Pulley Brackets, Station 583; Pulley Brackets, Station 623



VIEW E
AUTOMATIC PILOT SERVO PULLEYS

FOR AIRCRAFT A, B,
1 THROUGH 43
AND 45 THROUGH 59



FOR AIRCRAFT C, D,
40, 60, AND
SUBSEQUENT

NOSE UP

NOSE DOWN

ELEVATOR TRIM TAB CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)
1	3115353 Tabs Up Tabs Down	1	A	132	3/32 dia 7 x 7 flex	S-2049219 -16D-3R	2116126	S-149455 -3-028	S-2049219 -16D-3R
2	2390601-501 Tabs Up Tabs Down	2	B	(L_1) 484 (L_2) 89 3/4 (L_3) 2 5/8	3/32 dia 7 x 7 flex 1/16 dia 7 x 7 flex	AN669L3 LH	AN663-3	2344315	AN669S2 RH
3	4398907-557 Elev. Tab Auto-Pilot	2	C	4 5/8	3/32 dia 7 x 7 flex	AN664-3	AN669S3 LH		
4	4398907-553 Elev. Tab Auto-Pilot	1	D	140 1/2	3/32 dia 7 x 7 flex	AN669S3 RH	2460612- 501	RA2487-3	
5	4398907-555 Elev. Tab Auto-Pilot	1	E	142 1/2	3/32 dia 7 x 7 flex	AN669S3 RH	2460612-1	RA2487-3	
6	3391412-585 Tabs Up Tabs Down	1	F	400	1/16 dia 7 x 7 flex	AN669S2 LH	AN669S2 LH		
7	3391412-587 Tabs Up Tabs Down	1	G	533	1/16 dia 7 x 7 flex	AN669S2 LH	AN669S2 RH		

1,210

Figure 2-22 (Sheet 8 of 9 Sheets). Elevator Trim Tab Control System — Automatic Pilot Rigging and Cable Chart

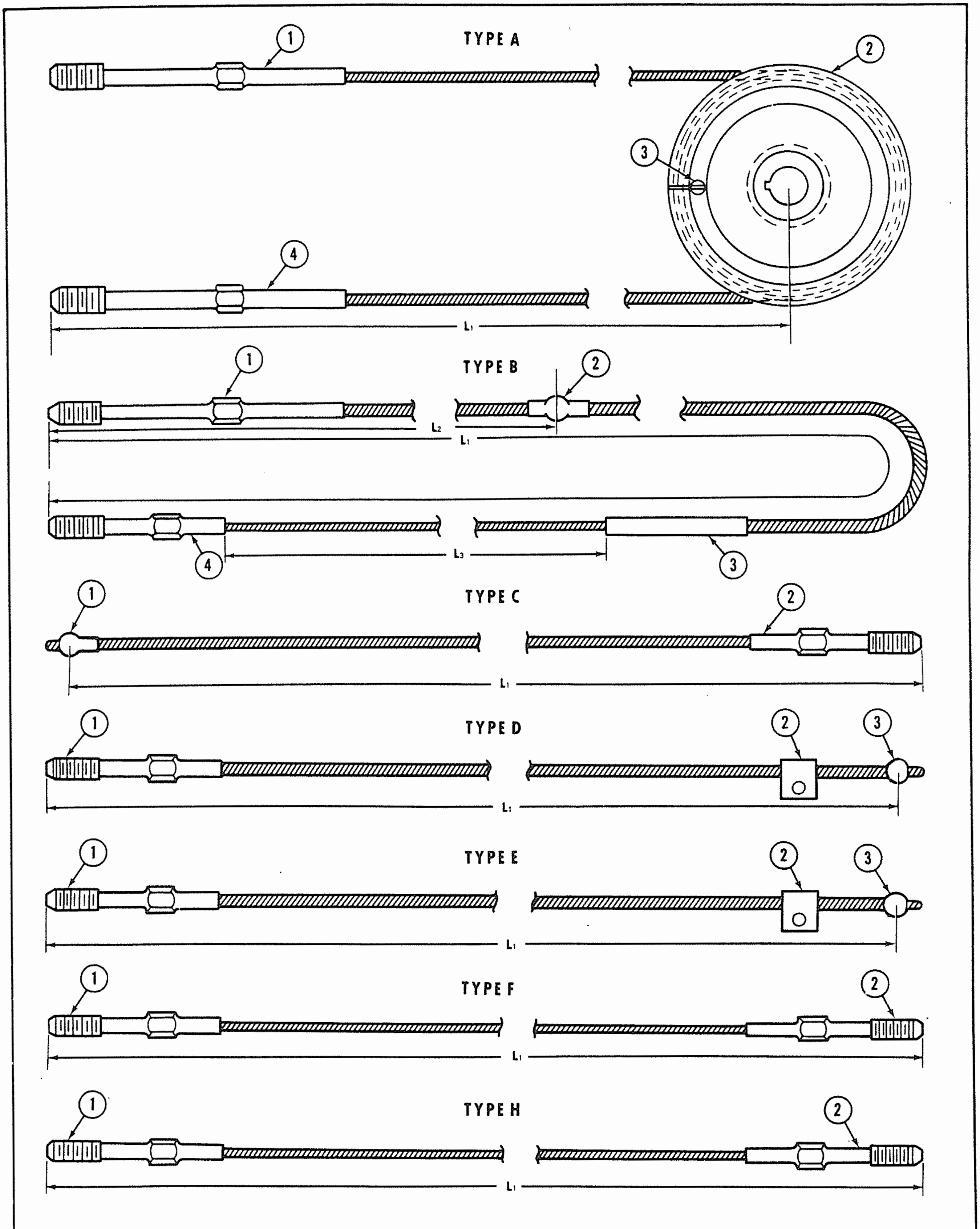


Figure 2-22 (Sheet 9 of 9 Sheets). Elevator Trim Tab Control System — Cable Assemblies

(Continued from Page 137)

2-160. MINOR REPAIR AND REPLACEMENT OF ELEVATOR TRIM TAB CONTROL WHEEL, INDICATOR, AND DRUM. Replace any broken or damaged indicator gears.

2-161. INSTALLATION OF ELEVATOR TRIM TAB CONTROL WHEEL, INDICATOR, AND DRUM.

a. Turn the trim tab control wheel until the indicator reads zero degrees.

b. Place the entire assembly, including the trim control wheel, indicator, drum, and cover assembly in position in the control pedestal head, pulling the cables down through the pulley brackets in the pedestal and aft to station 102. Bolt the cover assembly to the pedestal head.

c. Connect the elevator trim tab control cables with turnbuckles at station 102 and install the necessary grommets and guard pins.

d. Untape the cables and drums in the pedestal and stabilizers.

e. Tension the cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*.

f. Adjust the elevator trim tab control system (see paragraph 2-170).

g. Safety the turnbuckles.

h. Replace the access doors.

2-162. ELEVATOR TRIM TAB AUTOMATIC PILOT PULLEYS. *(See figure 2-22.)* The elevator trim tab automatic pilot pulleys are located just to the left of the center line of the aircraft, below the floor at station 236. For further information of the automatic pilot pulley, see paragraph 6-82.

2-163. ELEVATOR TRIM TAB CONTROL DRUM AND DRIVE MECHANISM. *(See figure 2-22.)* A cable-drum-operated drive mechanism, located in each horizontal stabilizer at station 112, provides rigid control for each elevator trim tab. As the trim tab cables are operated, each cable drum assembly is rotated and the trim tab is positioned through two control rods connected by a crank assembly at the elevator spar.

2-164. REMOVAL OF ELEVATOR TRIM TAB CONTROL DRUM AND DRIVE MECHANISM.

a. Adjust the elevator trim tab control wheel on the pedestal head until the indicator reads zero degrees. In this position the tabs are DOWN 6 ($\pm\frac{1}{2}$) degrees (neutral). Support the surfaces in this position.

b. Tape cables and drums at pedestal and horizontal stabilizer to prevent them from unwrapping.

c. Remove cabin floor panels and disconnect the appropriate trim tab cable turnbuckle at station 583. Thread the aft cable and tape the intermediate cable.

d. Disconnect the trim tab cable turnbuckle in the left horizontal stabilizer at station 72 (accessible through access door in the upper surface of the stabilizer). Thread the cable end.

e. Remove necessary guard pins and draw the cable ends toward the drum.

f. Unbolt the forward trim tab control rod from the crank at the elevator spar and from the trim tab drum in the stabilizer. Remove the control rod.

g. Release the drum from the bracket on the stabilizer structure and remove.

2-165. MINOR REPAIR AND REPLACEMENT OF ELEVATOR TRIM TAB CONTROL DRUM AND DRIVE MECHANISM. Replace any worn or damaged bearings, bushings, and oil seals.

2-166. INSTALLATION OF ELEVATOR TRIM TAB CONTROL DRUM AND DRIVE MECHANISM.

a. Place the elevator trim tab in the neutral position, 6 ($\pm\frac{1}{2}$) degrees DOWN from the faired position, and support the surface in this position.

b. Turn the elevator trim tab control wheel on the pedestal head until the indicator reads zero degrees.

c. Install the trim tab control drum on the bracket on the stabilizer structure, rigging the control cables through the horizontal stabilizers and tail section to station 583.

d. Bolt the forward trim tab control rod to the crank at the elevator spar and to the trim tab drum.

e. Connect the turnbuckles at fuselage station 583 and at left horizontal stabilizer station 72.

f. Remove the tape from the cables and drums in the horizontal stabilizer and at the control pedestal.

g. Tension the cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*.

h. Remove the surface supports.

i. Adjust the elevator trim tab control system (see paragraph 2-170).

j. Safety the turnbuckles.

k. Replace the access doors.

2-167. ELEVATOR TRIM TAB CONTROL CABLES. *(See figure 2-22.)* A two-way cable system, originating at the elevator trim tab control drum in the pedestal head, extends aft to the trim tab control drums and drive mechanisms at station 112 in the horizontal stabilizers. Cable stops, attached to the elevator trim tab control cables, limit the movement of the cables by striking a fixed fairlead at station 156 in the fuselage. The stops are not adjustable.

2-168. REMOVAL OF ELEVATOR TRIM TAB CONTROL CABLES.

a. Tape cables and drums in the pedestal and in the horizontal stabilizer to prevent them from unwrapping.

b. Remove the elevator trim tab cable drum from the pedestal head (see paragraph 2-159).

c. Remove the drum from the shaft and unwrap the cable from the drum.

d. Disconnect the servo cable fittings from the trim tab cables at station 225.

e. Remove cabin floor panels and disconnect the trim tab cable turnbuckles at station 583. Thread the cables for removal.

f. Remove the necessary fairlead grommets and pulley guard pins and draw the intermediate control cables forward. Remove the cables through an access door forward of station 156.

g. Remove the elevator trim tab control drums from the horizontal stabilizers (see paragraph 2-164).

h. Unwrap the cables from the drums.

2-169. INSTALLATION OF ELEVATOR TRIM TAB CONTROL CABLES.

a. Support the elevator trim tabs in neutral, 6 (±½) degrees DOWN.

b. Wrap the forward cable on the trim tab cable drum *(see figure 2-22).*

c. Install the drum on the elevator trim tab control wheel shaft, turn the wheel until the indicator reads zero degrees, route the control cables down through the pedestal and aft through the fuselage, and install the control wheel, drum, and indicator at the pedestal head. Bolt the cover assembly to the pedestal. Route the cables to station 102, replacing necessary grommets and guard pins.

d. Insert the intermediate trim tab cables into the fuselage through an access door forward of station 156 and route the appropriate cable ends aft through the fuselage to station 583. Replace grommets and guard pins.

e. Connect trim tab cables at station 102.

f. Attach the servo cable fittings to the trim tab cables at station 225.

g. Wrap the aft elevator trim tab cables on the trim tab control drums *(see figure 2-22).*

h. Install the tab drums in the horizontal stabilizers at station 112, and connect the cables (see paragraph 2-166).

i. Tension the elevator trim tab and servo cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

j. Remove the surface locks.

k. Adjust the elevator trim tab control system (see paragraph 2-170).

l. Adjust the elevator trim tab servo cables.

m. Safety all turnbuckles.

n. Replace access doors and floor panels.

2-170. ADJUSTMENT OF ELEVATOR TRIM TAB CONTROL SYSTEM.

Note

With the elevator trim tabs in the faired position, mark points in line on the tabs and on the elevators for measuring the elevator trim tab travel. Measure between these points when the tabs are in the extreme UP or DOWN position.

a. Place the control columns in neutral (10 degrees for aircraft A, B, and 1 through 17, or 11½ degrees for aircraft C, D, 18 and subsequent, forward of vertical) and turn the elevator trim tab control wheel until the indicator reads zero degree (neutral). Both elevator trim tabs should be DOWN 6 (±½) degrees or 29/32 (±⅛) inch from the elevator trailing edges.

b. Retain the control columns in neutral and turn the elevator trim tab control wheel until the indicator reads full NOSE DOWN. Both elevator trim tabs should be faired with the elevator trailing edges.

c. Retain the control columns in neutral and turn the elevator trim tab control wheel until the indicator reads full NOSE UP. Both elevator trim tabs should be DOWN 25 (+1, −0) degrees or 3¾ (+¼, −0) inches from the elevator trailing edges.

d. Check the tension of the cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

e. Safety the turnbuckles.

f. If correct throws are not achieved for the trim tabs, it will be necessary to check the rigging of the system, (see paragraph 2-169).

g. For further information on rigging the elevator trim tab control system, see paragraph 2-231.

2-171. RUDDER. The rudder, consisting of an aluminum-alloy framework with fabric covering, is balanced by lead weights installed in the leading edge. It is supported at the vertical stabilizer by two hinge fittings, and at the base by the rudder torque tube. A geared tab, which is also trim controlled, is incorporated in a cut-out in the trailing edge of the surface.

2-172. REMOVAL OF RUDDER.
(See figure 2-23.)

a. Operate the rudder pedals in the flight compartment to position the rudder in neutral (faired with the tail cone).

b. Adjust the rudder trim control wheel on the aft face of the control pedestal until the rudder trim indicator reads zero degrees. Clamp the rudder tab in neutral.

c. Open the access door on the left side of the rudder and disconnect the rudder tab control rods from the crank at the rudder spar.

d. Unfasten the bolts connecting the rudder to the rudder torque tube.

e. Remove the stop nut from the eyebolt stud and the bolt from the hinge fitting. Remove the rudder.

2-173. MINOR REPAIR AND REPLACEMENT OF RUDDER. Replace any worn or damaged bearings.

2-174. BALANCING OF RUDDER.
(See figure 2-24.)

a. Remove the rudder from the aircraft (see paragraph 2-172).

b. Mount the rudder on the rudder balancing stands, with the tab and tab operating mechanism installed. Check the hinge bearings for binding, and adjust the stands so that the surface swings freely.

Note

The location of the hinge line is extremely critical. The attaching eyebolts must be snug so that the hinge line (relative to the surface) is the same as the hinge line when the surface is installed on the aircraft.

c. Place a 1-pound weight on the rudder at station 74 (adjacent to the bottom of the tab). Move the weight along this station line until the rudder is horizontal. Measure the distance from the extreme trailing edge of the rudder to the center of the weight. This distance must be between 18 5/8 and 28 5/8 inches for the rudder to be correctly balanced.

d. If the distance is more than 28 5/8 inches, but less than 33 inches, add 0.43 pound in the bay between stations 115 and 124. Repeat step c.

e. If the distance is more than 33 inches, but less than 39 inches, accomplish step d and add 0.5 pound in the bay between stations 102 and 110. Repeat step c.

f. If the distance is less than 18 5/8 inches, but more than 14 inches, remove 0.43 pound from the bay between stations 115 and 124 and repeat step c.

g. If the distance is less than 14 inches, but more than 8 inches, accomplish step f and remove 0.5 pound from the bay between stations 102 and 110. Repeat step c.

Note

When the rudder has been thrown out of balance because of a weight change, adjust the counterbalance weight which is opposite or nearest to the change. When the rudder has been thrown out of balance due to repairs made in a number of places or when new covering, paint, dope, etc., is applied, balance the rudder in accordance with the steps outlined above.

2-175. RUDDER HINGE WEAR TOLERANCES. Due to the close tolerance fits of the rudder hinge bolts and bearings, the maximum wear limit is 0.001 inch in excess of the manufacturing tolerance between the two parts.

2-176. INSTALLATION OF RUDDER.
(See figure 2-23.)

a. Check that the rudder pedals are in neutral (aligned) and that the rudder trim control indicator on the aft face of the control pedestal reads zero degrees.

b. Support the rudder in position at the vertical stabilizer and install the bolt in the hinge fitting and the stop nut on the eyebolt stud.

c. Connect the rudder torque tube to the rudder.

d. Connect the rudder tab control rods to the crank at the rudder spar.

e. Check the adjustment of the rudder control system (see paragraph 2-189); the rudder geared tab control system (see paragraph 2-200); and the rudder trim control system (see paragraph 2-217).

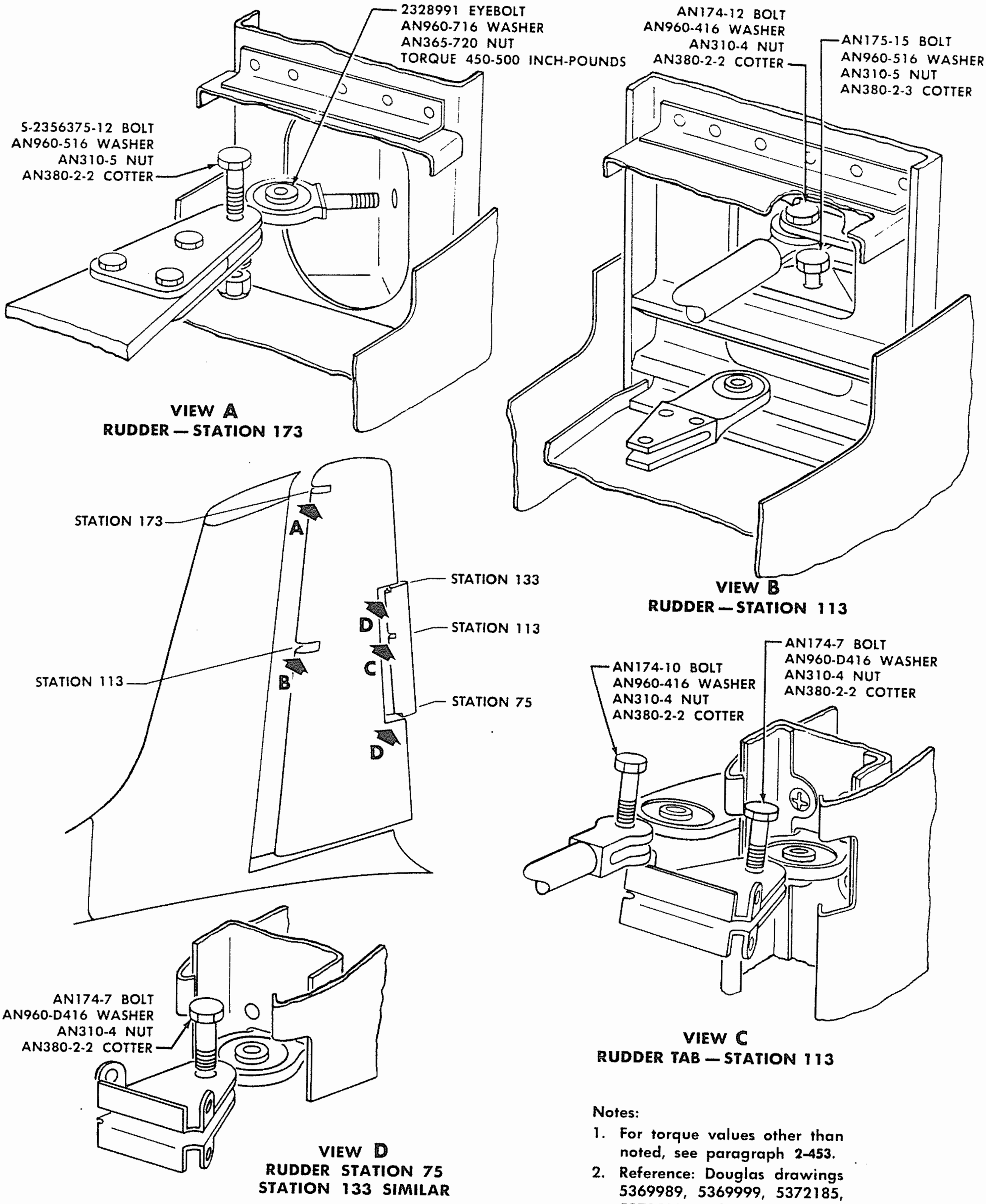
f. Safety all turnbuckles.

g. Remove the clamp from the rudder tab.

h. Install the access door on the left side of the rudder.

2-177. RUDDER CONTROL SYSTEM. *(See figure 2-25.)* The rudder, which controls the yawing of the aircraft about the vertical axis, is controlled by duplicate sets of adjustable pedals in the flight compartment. A two-way cable system extends between the horns of the rudder pedal torque tube assemblies and the large bellcrank at the base of the rudder torque tube to transmit the movement of the pedals to the torque tube and thus to the rudder. Adjustable stops on the rudder pedal torque tube horns limit the movement of the pedals; rudder travel is limited by stops located on the rudder torque tube stop assembly. An automatic pilot servo unit is attached to the nose LEFT (rudder LEFT) cable at station 173.

2-178. RUDDER PEDAL ASSEMBLIES. *(See figure 2-25.)* The rudder is operated by duplicate sets of adjustable pedals in the flight compartment, which



Notes:
1. For torque values other than noted, see paragraph 2-453.
2. Reference: Douglas drawings 5369989, 5369999, 5372185, 5372521, and 5372635

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Figure 2-23. Installation of Rudder and Rudder Tab

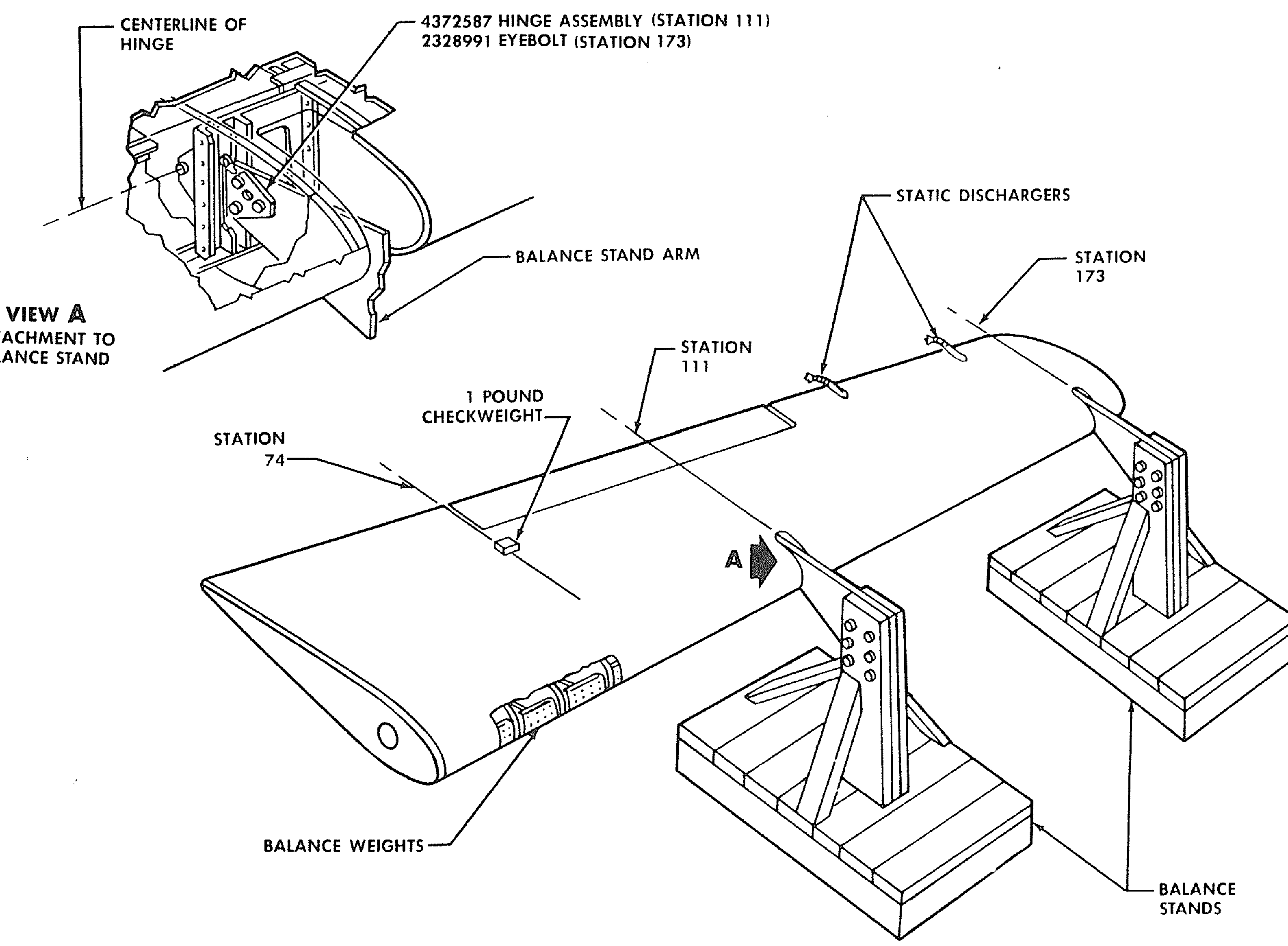


Figure 2-24. Balancing Rudder

function in unison through linked torque tubes beneath the floor. Movement of the pedals is transmitted by the torque tube horns to the control cables, which extend aft to the bellcrank on the rudder torque tube in the tail section. Each rudder pedal is mounted on a perforated spline shaft. A locking plunger, operated by a foot lever at the side of the pedal, adjusts the pedal to the stature of the pilot by engaging the holes in the spline shaft. Rudder pedal action is synchronized by a bus cable that extends forward from the rudder pedal torque tube horns to a bus pulley at station −19.

2-179. REMOVAL OF RUDDER PEDAL ASSEMBLIES.

a. Place rudder pedals in neutral and lock the surface in the faired position.

b. Open the upper hinged fairing on the fuselage nose.

c. Remove the two structural stiffeners located between stations −9½ and 1.

d. Remove the four rods that connect the rudder pedals to the brake torque tubes.

e. Detach the Adel blocks located forward of the rudder pedals. This will allow the pipes to be raised slightly when the pedal mechanism is removed.

Note

It will not be necessary to disconnect any pipes to remove or install the rudder pedal assemblies.

f. Open the access door on the under side of the fuselage and release tension in the rudder control cables at stations 50 and 58.

g. Open the access door below the rudder pedal torque tube and disconnect the rudder control cables and the rudder bus cable from the torque tube horns.

h. Remove the bolts from the two rudder pedal brackets, and screw out the bushings. This will free both the left and right pedal assemblies.

i. Remove the pedals from the arms on the pilot's side.

j. Loosen the aft ends of the two braces which extend from the brake valve at station −9½ to station 1. This will allow clearance for the removal of the pedal assemblies.

k. Remove the right pedal assembly by working it out through the right side of the flight compartment.

l. Remove the left pedal assembly in the same manner.

2-180. MINOR REPAIR AND REPLACEMENT OF RUDDER PEDAL ASSEMBLIES. Replace worn or damaged components of the rudder pedal assemblies.

2-181. INSTALLATION OF RUDDER PEDAL ASSEMBLIES.

a. Lock the rudder in the faired position.

b. Position the pedal assemblies on the arms. Install the bolts on the rudder pedal brackets and screw in the bushings.

c. Tighten the aft ends of the two braces that extend from the brake valve at station −9 to station 1.

d. Connect the rudder control cables and the rudder bus cable to the torque tube horns.

e. Attach the Adel blocks forward of the rudder pedals.

f. Install the four rods connecting the rudder pedals to the brake torque tubes.

g. Install the two structural stiffeners between station −9 and station 1.

h. Close the upper hinged fairing on the fuselage nose.

i. Tension the rudder control cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*.

j. Remove the rudder lock.

k. Adjust the rudder control system (see paragraph 2-189).

l. Safety the turnbuckles.

m. Replace the access doors.

2-182. RUDDER AUTOMATIC PILOT PULLEY. The rudder automatic pilot pulley is located to the left of the center line of the aircraft, below the floor at station 180. For further information of the automatic pilot pulley, see paragraph 6-82.

2-183. RUDDER TORQUE TUBE ASSEMBLY. *(See figure 2-25.)* The rudder torque tube assembly, located in the tail section of the aircraft just below the rudder nose section, transmits the movement of the control cables to the rudder. Adjustable rudder stops are located on the torque tube stop assembly.

2-184. REMOVAL OF RUDDER TORQUE TUBE ASSEMBLY.

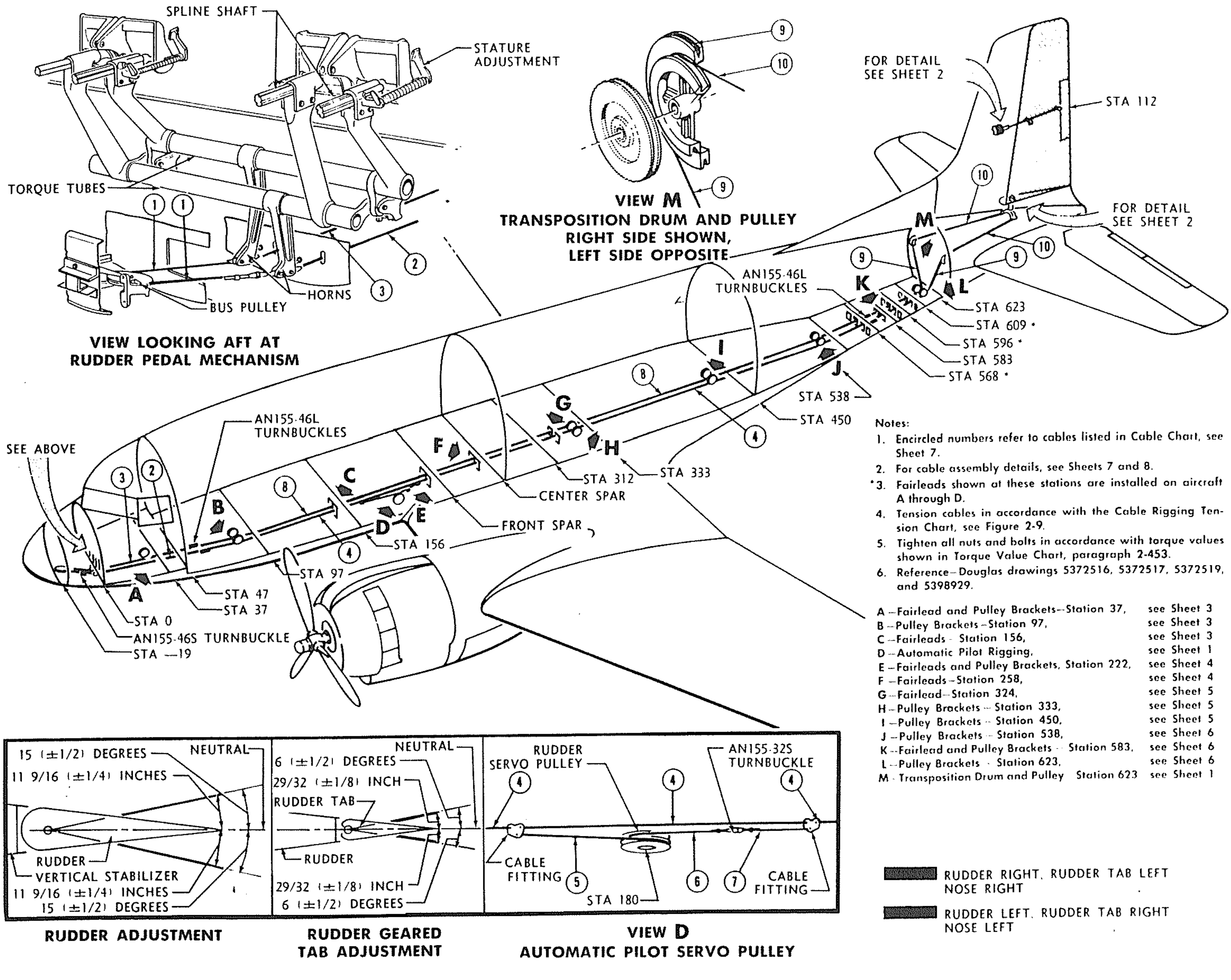
a. Position the rudder pedals in neutral (aligned) and lock the rudder in the faired position.

b. Remove the cabin floor and release tension of rudder cables at station 593.

c. Remove the tail cone from the aircraft (see paragraph 2-338).

d. Disconnect the cables from the rudder torque tube bellcrank. It is not necessary to remove the cables from the aircraft.

(Continued on Page 161)



Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 7.
2. For cable assembly details, see Sheets 7 and 8.
*3. Fairleads shown at these stations are installed on aircraft A through D.
4. Tension cables in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values shown in Torque Value Chart, paragraph 2-453.
6. Reference—Douglas drawings 5372516, 5372517, 5372519, and 5398929.

A—Fairlead and Pulley Brackets—Station 37, see Sheet 3
B—Pulley Brackets—Station 97, see Sheet 3
C—Fairleads—Station 156, see Sheet 3
D—Automatic Pilot Rigging, see Sheet 1
E—Fairleads and Pulley Brackets, Station 222, see Sheet 4
F—Fairleads—Station 258, see Sheet 4
G—Fairlead—Station 324, see Sheet 5
H—Pulley Brackets—Station 333, see Sheet 5
I—Pulley Brackets—Station 450, see Sheet 5
J—Pulley Brackets—Station 538, see Sheet 6
K—Fairlead and Pulley Brackets—Station 583, see Sheet 6
L—Pulley Brackets—Station 623, see Sheet 6
M—Transposition Drum and Pulley Station 623 see Sheet 1

Figure 2-25 (Sheet 1 of 8 Sheets). Rudder and Rudder Geared Tab Control System — Key Drawing

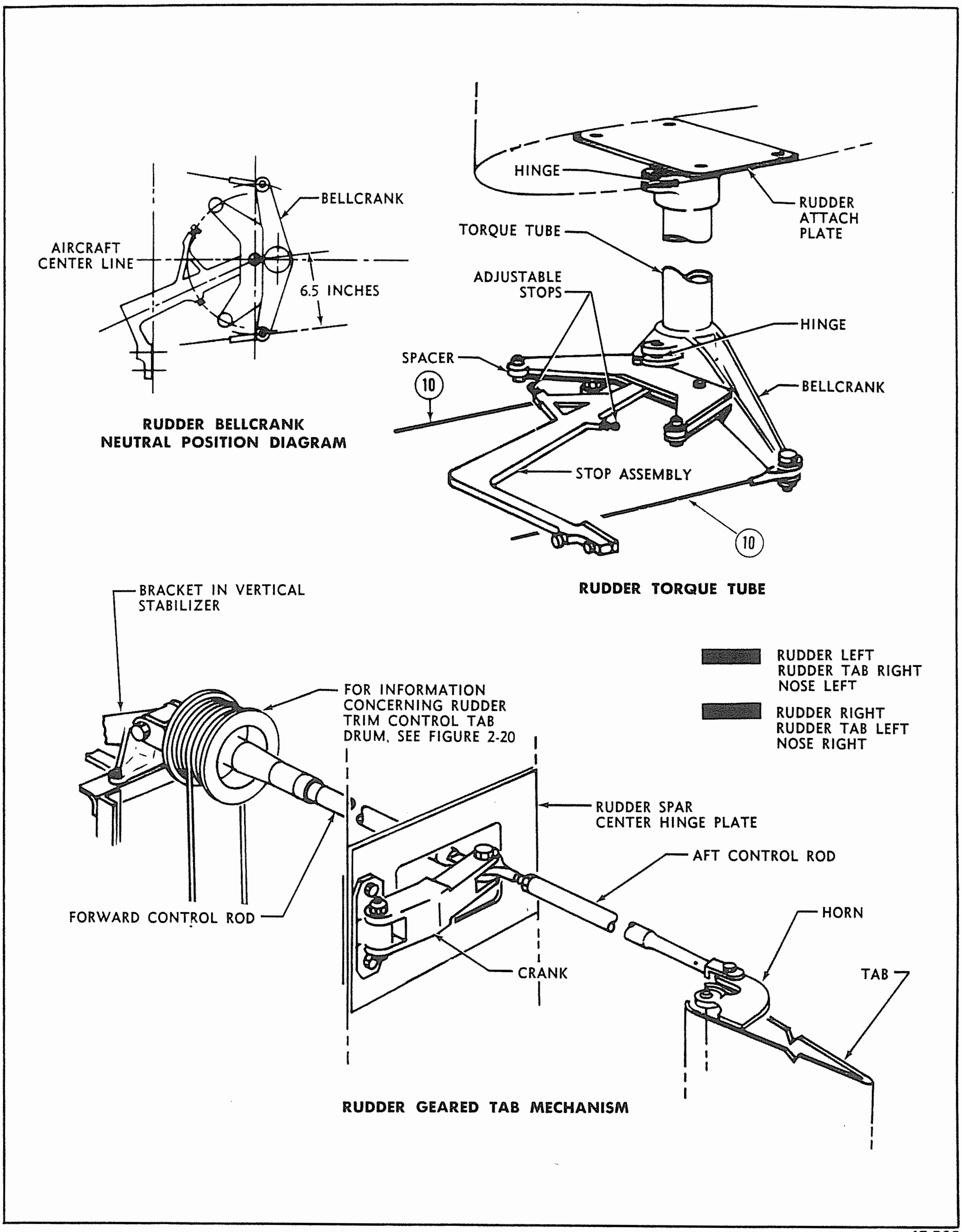
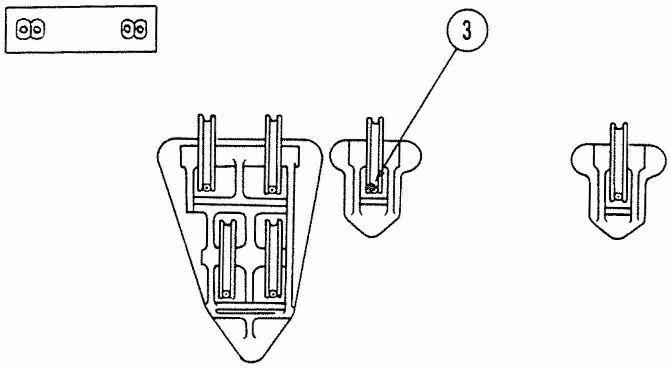
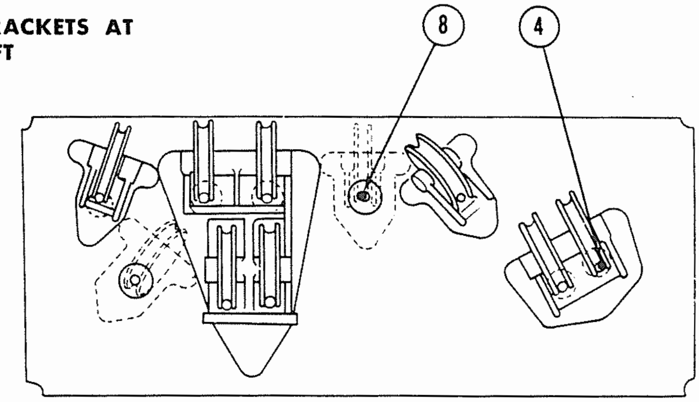


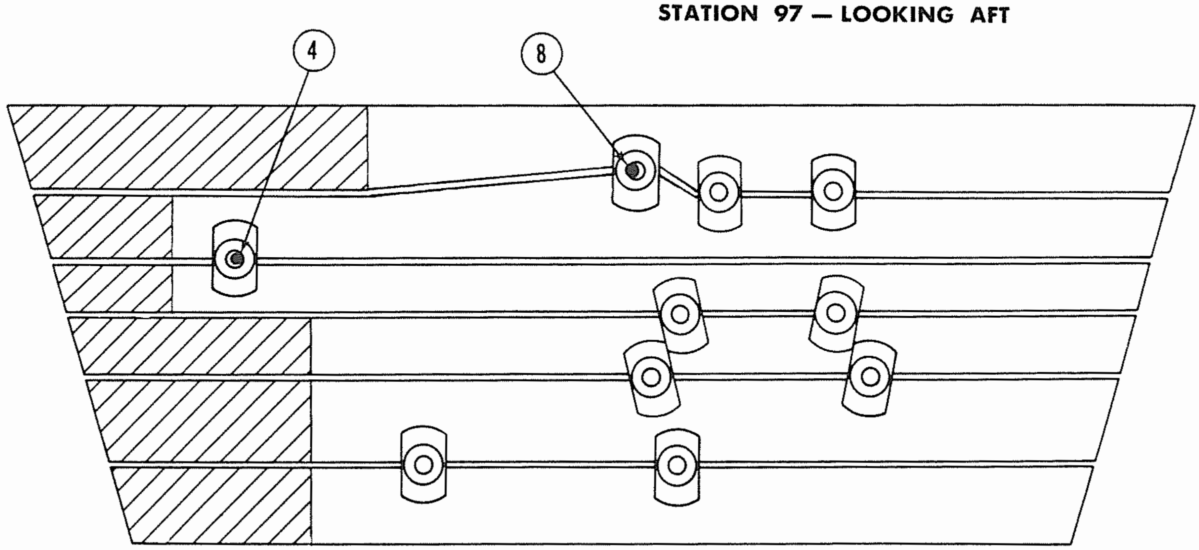
Figure 2-25 (Sheet 2 of 8 Sheets). Rudder and Rudder Geared Tab Control System — Torque Tube, Geared Tab Mechanism



VIEW A — FAIRLEAD AND PULLEY BRACKETS AT STATION 37 — LOOKING AFT



VIEW B — PULLEY BRACKETS AT STATION 97 — LOOKING AFT



VIEW C — FAIRLEADS AT STATION 156 — LOOKING FORWARD

RUDDER RIGHT, RUDDER TAB LEFT, NOSE RIGHT

RUDDER LEFT, RUDDER TAB RIGHT, NOSE LEFT

SHADED PORTION OF FAIRLEAD EFFECTIVE ON AIRCRAFT A, B, AND 1 THROUGH 28.

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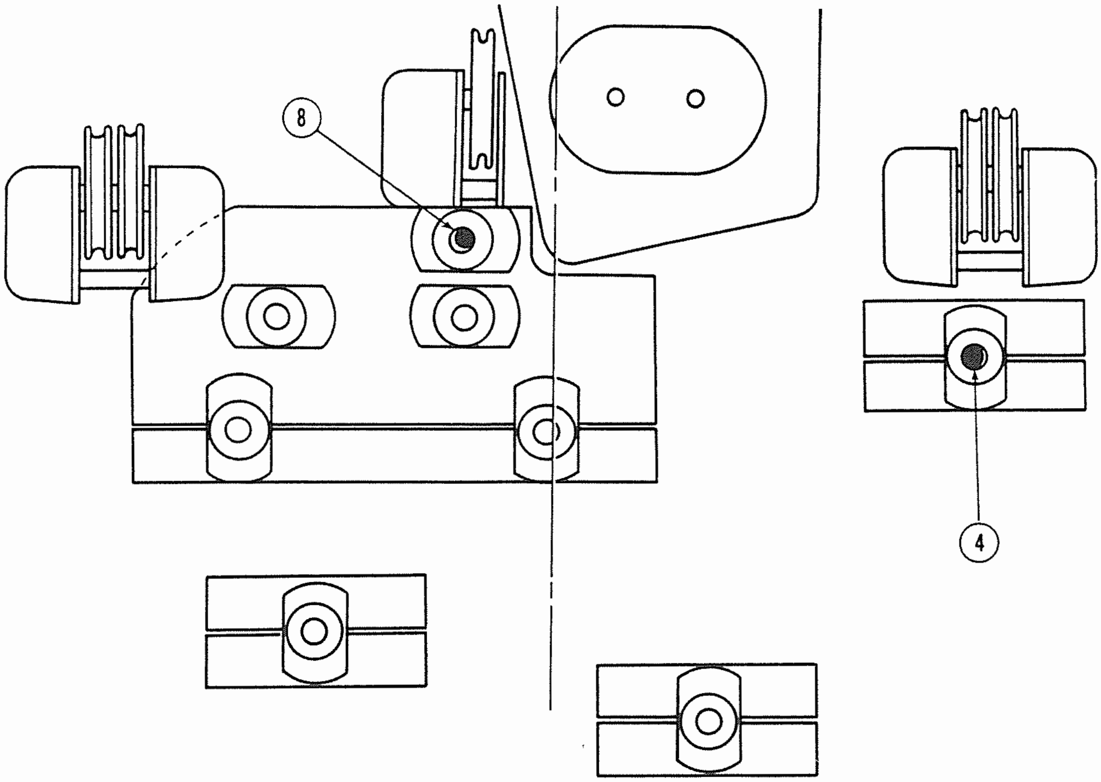
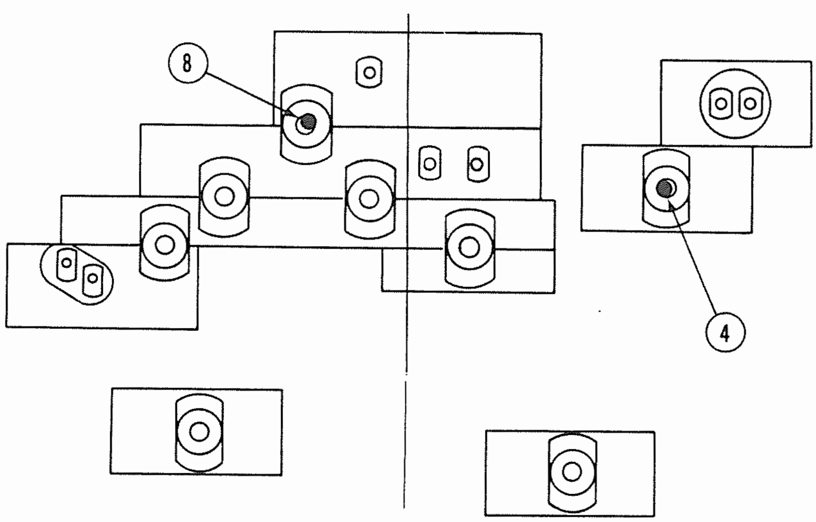
Figure 2-25 (Sheet 3 of 8 Sheets). Rudder and Rudder Geared Tab Control System — Fairlead and Pulley Brackets, Station 37; Pulley Brackets, Station 97; Fairleads, Station 156



VIEW E — FAIRLEADS AND PULLEY BRACKETS AT STATION 222, FRONT SPAR — LOOKING AFT



VIEW F — FAIRLEADS AT STATION 258, CENTER SPAR — LOOKING AFT

RUDDER RIGHT, RUDDER TAB LEFT, NOSE RIGHT

RUDDER LEFT, RUDDER TAB RIGHT, NOSE LEFT

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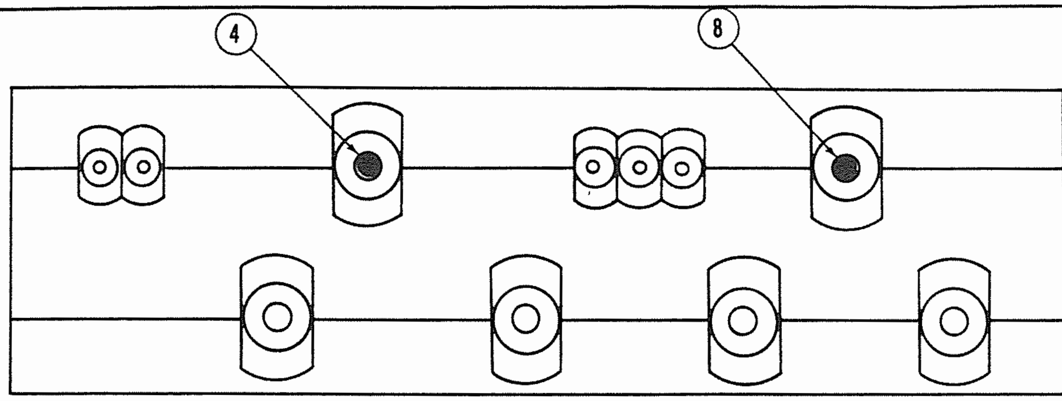
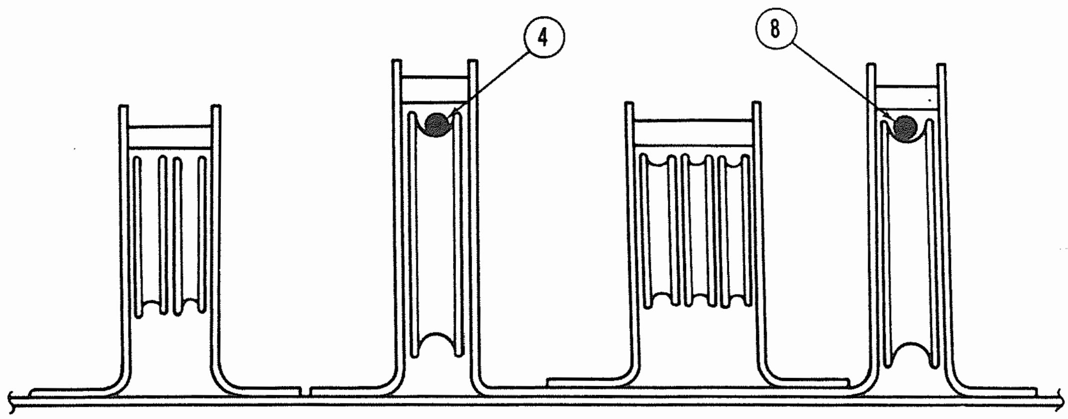
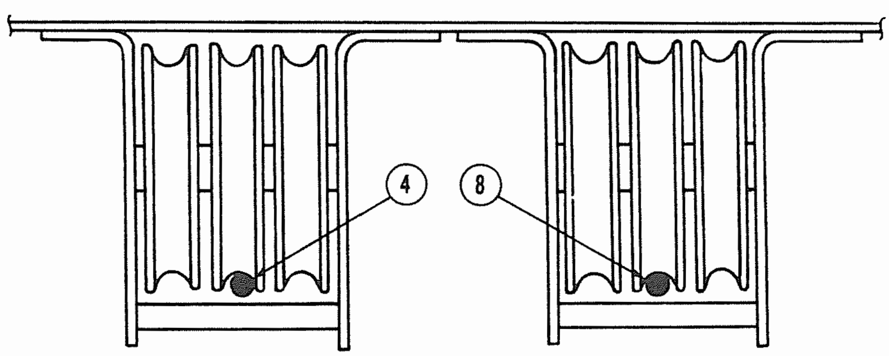
Figure 2-25 (Sheet 4 of 8 Sheets). Rudder and Rudder Geared Tab Control System — Fairleads and Pulley Brackets, Station 222; Fairleads, Station 258



VIEW G FAIRLEAD
STATION 324 — LOOKING FORWARD



VIEW H PULLEY BRACKETS
STATION 333 — LOOKING FORWARD



VIEW I PULLEY BRACKETS
STATION 450 — LOOKING FORWARD

RUDDER RIGHT — RUDDER TAB LEFT — NOSE RIGHT

RUDDER LEFT — RUDDER TAB RIGHT — NOSE LEFT

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Figure 2-25 (Sheet 5 of 8 Sheets). Rudder and Rudder Geared Tab Control System — Fairlead, Station 324; Pulley Brackets, Stations 333 and 450

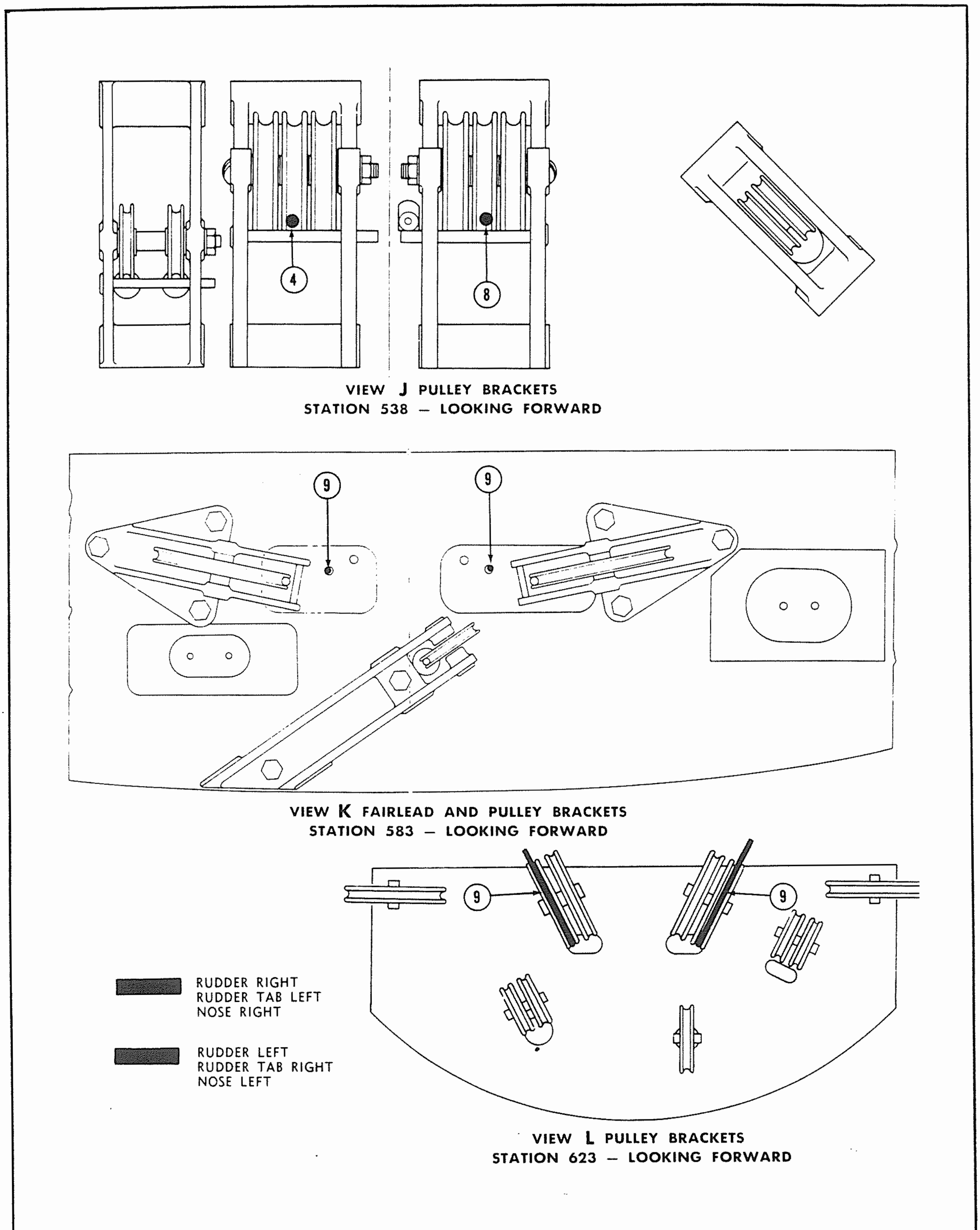
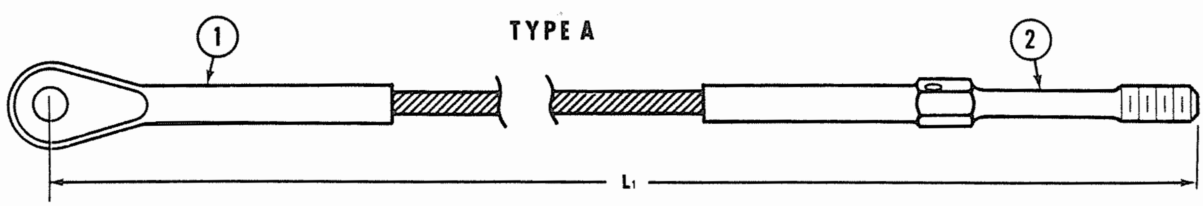


Figure 2-25 (Sheet 6 of 8 Sheets). Rudder and Rudder Geared Tab Control System — Pulley Brackets, Stations 538 and 623; Fairleads and Pulley Brackets, Station 583

RUDDER AND RUDDER GEARED TAB CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L1)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)
1	3391412-533 Rudder Left Rudder Right	1	A	34 3/8	3/16 dia 7x19 flex	AN668-6	AN669S6 LH		
2	3391412-541 Rudder Left	1	B	57 3/8	3/16 dia 7x19 flex	AN668-6	AN669L6 RH		
3	3391412-535 Rudder Right	1	A	50 3/4	3/16 dia 7x19 flex	AN668-6	AN669S6 LH		
4	4398907-515 Rudder Left	1	C	(L_1) 536 1/2 (L_2) 115 1/4 (L_3) 23 3/4	3/16 dia 7x19 flex	AN669L6 LH	AN663-6	AN663-6	AN669L6 LH
5	4398907-521 Rudder Servo Bus	1	D	12 1/4	1/8 dia 7x19 flex	2391676	AN664-4	AN664-4	
6	4398907-525 Rudder Servo Jumper	1	E	14 1/2	1/8 dia 7x19 flex	AN664-4	AN669S4 LH		
7	4398907-523 Rudder Servo Jumper	1	F	6 3/4	1/8 dia 7x19 flex	AN669S4 RH	AN664-4	2391676	
8	4398907-1 Rudder Right	1	G	543	3/16 dia 7x19 flex	AN669L6 RH	AN669L6 LH		
9	3391412-573 Rudder Left Rudder Right	2	H	80 1/4	3/16 dia 7x19 flex	AN669L6 RH	AN664-6		
10	3391412-571 Rudder Left Rudder Right	2	I	92	3/16 dia 7x19 flex	AN664-6	AN668-6		



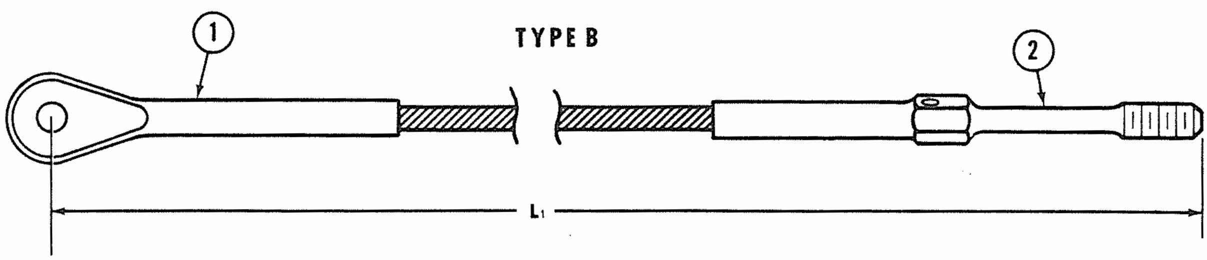


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Figure 2-25 (Sheet 7 of 8 Sheets). Rudder and Rudder Geared Tab Control System — Cable Chart and Cable Assemblies

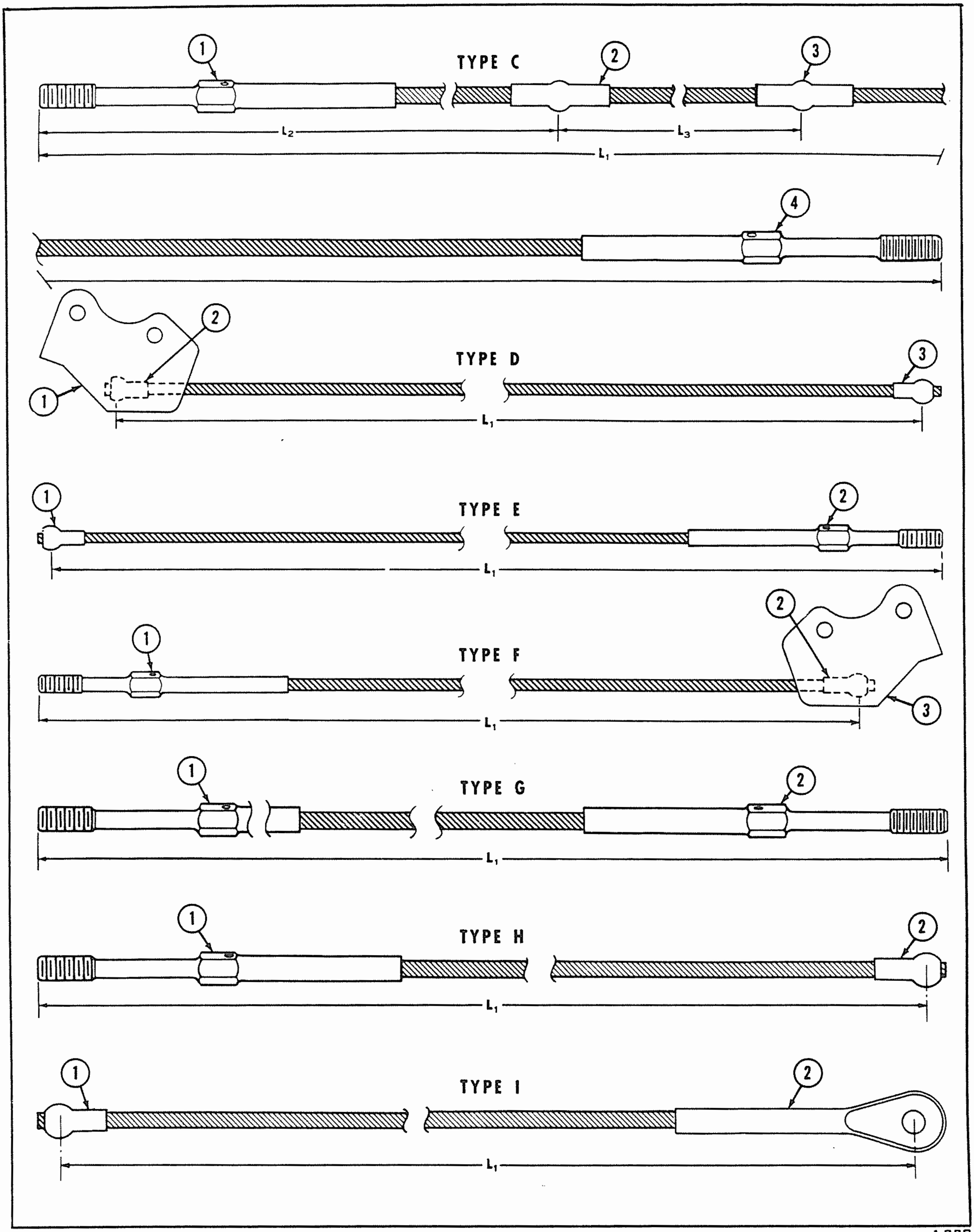


Figure 2-25 (Sheet 8 of 8 Sheets). Rudder and Rudder Geared Tab Control System — Cable Assemblies

(Continued from Page 152)

e. Remove the bolt connecting the torque tube bellcrank, stop assembly, and lower rudder torque tube hinge bracket.

f. Detach the three bolts securing the stop assembly to the tail section structure.

g. Remove the stop assembly from the aircraft.

h. Remove the six bolts that secure the rudder torque tube to the base of the rudder.

i. Support the torque tube and release the rudder torque tube hinge from the upper hinge bracket.

j. Remove the torque tube assembly from the aircraft.

2-185. INSTALLATION OF RUDDER TORQUE TUBE ASSEMBLY.

a. Place the rudder pedals in neutral and lock the rudder in the faired position.

b. Support the rudder torque tube in position and connect the torque tube hinge to the upper hinge bracket.

c. Attach the rudder torque tube to the base of the rudder with the six bolts.

d. Install the stop assembly on the tail section structure.

e. Install the bolt connecting the torque tube bellcrank, stop assembly, and lower rudder torque tube hinge bracket.

f. Connect the cables to the rudder torque tube bellcrank.

g. Install the tail cone on the aircraft (see paragraph 2-339).

h. Tension the rudder control cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*.

i. Release the rudder lock.

j. Adjust the rudder control system (see paragraph 2-189).

k. Safety the turnbuckles.

l. Replace the cabin floor at station 593.

2-186. RUDDER CONTROL CABLES. *(See figure 2-25.)* A two-way rudder control cable system begins at the rudder pedal torque tube horns below the flight compartment floor and extends aft through the fuselage to the rudder torque tube bellcrank in the tail section of the aircraft.

2-187. REMOVAL OF RUDDER CONTROL CABLES.

a. Place the rudder pedals in neutral.

b. Open the access door on the under side of the fuselage and disconnect the rudder control cables at stations 50 and 58. Thread the cable ends.

c. Open the access door below the rudder pedal torque tubes and disconnect the rudder control cables and the rudder bus cable from the torque tube horns.

d. Release the rudder bus cable pulley from the bracket at station —19 and remove the bus cable from the aircraft.

e. Remove the guard pins from the bracket at station 37 and draw the forward rudder cables forward and out of the aircraft.

f. Loosen the turnbuckle connecting the rudder servo jumper cables at approximately station 191. Release the cables from the rudder automatic pilot pulley. Unfasten the bus cable fittings and remove the rudder servo cables from the aircraft.

g. Remove the cabin floor and disconnect the rudder cables at station 593.

h. Remove guard pins and grommets and draw the forward intermediate rudder cables aft. Remove the cables.

i. Disconnect the cables from the rudder torque tube bellcrank in the tail section.

j. Release the four cables from the transposition drums, remove the necessary guard pins, thread the aft intermediate cables, and draw them out of the aircraft.

k. Remove the aft cables.

2-188. INSTALLATION OF RUDDER CONTROL CABLES.

a. Place the rudder pedals in neutral (aligned), and lock the rudder in the faired position.

b. Connect the aft rudder control cables and the aft intermediate cables to the transposition drums, and route the aft intermediate cables forward through the aircraft, installing necessary guard pins and fairlead grommets.

c. Attach the aft cables to the rudder torque tube bellcrank and connect the cables with turnbuckles at station 593.

d. Install the rudder servo cables at the rudder automatic pilot pulley. Attach the bus cable fittings to the rudder left cable.

e. Route the forward rudder control cables aft through the aircraft, installing necessary guard pins and fairlead grommets.

f. Install the forward rudder bus cable.

g. Attach the bus cable and the forward rudder control cables to the rudder torque tube horns.

h. Connect the cables with turnbuckles at stations 50 and 58.

i. Tension the cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*.

j. Remove the surface lock.

k. Adjust the rudder control system (see paragraph 2-189).

l. Safety the turnbuckles.

m. Replace the access doors.

2-189. ADJUSTMENT OF RUDDER CONTROL SYSTEM.

a. Adjust the stops on the rudder pedal torque tube horns to give the rudder pedals a travel of 4⅜ (±⅛) inches forward and aft from the neutral position. The pedals are in neutral position when aligned.

b. Check the stature adjustment of the splined shafts for equal extension.

c. Place the rudder pedals in neutral (aligned) and turn the rudder trim control wheel, on the aft face of the control pedestal, until the rudder trim control indicator reads zero degrees. The rudder should be faired with the tail cone and the rudder tab should be faired with the rudder (neutral positions).

d. Clamp the rudder tab in the faired position and disconnect and remove the forward tab control rod.

Note

With the rudder in the faired position and the tab clamped in the faired position, mark points in line on the bottom corner of the rudder and on the tail section for measuring rudder travel. Measure between these points when the rudder is in the extreme LEFT or RIGHT position.

e. Push the left rudder pedals full forward, 4⅜ (±⅛) inches from neutral. Adjust the left rudder stop on the rudder torque tube stop assembly so that the rudder throw is LEFT 15 (±½) degrees or 11 9/16 (±¼) inches from neutral.

f. Push the right rudder pedals full forward 4⅜ (±⅛) inches from neutral. Adjust the right rudder stop on the rudder torque tube stop assembly so that the rudder throw is RIGHT 15 (±½) degrees or 11 9/16 (±¼) inches from neutral.

g. If correct movements or distances are not achieved, it will be necessary to check the rigging *(see figure 2-25).*

h. Check the rudder transposition drum at station 625 in the tail section to assure approximately 45 degrees of cable wrap remaining on the drum and sector at extreme limits of rudder throw.

i. Install the forward tab control rod.

j. Tension the cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

k. Safety the turnbuckles.

l. Release the surface lock on the rudder tab.

m. Check the geared tab throws (see paragraph 2-200).

n. For further information on rigging the rudder control system, see paragraph 2-231.

2-190. RUDDER GEARED TAB. A rudder geared tab, constructed of conventional metal ribs and skin, is attached by three hinge fittings at a cut-out in the rudder trailing edge to provide aerodynamic assistance for the movement of the rudder. The tab is actuated by the control rods of the geared tab mechanism; also by a trim control mechanism, attached to the forward end of the geared tab mechanism forward control rod, for trimming purposes.

2-191. REMOVAL OF RUDDER CONTROL. *(See figure 2-23.)*

a. Disconnect the aft rudder tab control rod from the tab horn.

b. Remove the bolts from the three hinge fittings.

c. Remove the tab.

2-192. BALANCING OF RUDDER GEARED TAB. Balancing of the rudder geared tab is not required.

2-193. RUDDER GEARED TAB HINGE WEAR TOLERANCES. Due to the close tolerance fits of the rudder geared tab hinge bolts and bearings, the maximum wear limit is 0.001 inch in excess of the manufacturing tolerance between the two parts.

2-194. MINOR REPAIR AND REPLACEMENT OF RUDDER GEARED TAB. Replace any worn or damaged bearings.

2-195. INSTALLATION OF RUDDER GEARED TAB.
(See figures 2-23 and 2-25.)

a. Place the rudder in neutral (faired with the tail cone) and turn the rudder trim control wheel, on the aft face of the pedestal, until the trim control indicator reads zero degrees.

b. Support the rudder geared tab in position and install the bolts in the three hinge fittings.

c. Connect the aft rudder tab control rod to the tab horn.

d. Check the adjustment of the rudder geared tab control system (see paragraph 2-200), and the rudder trim control system (see paragraph 2-217).

2-196. RUDDER GEARED TAB CONTROL SYSTEM. *(See figure 2-25.)* When functioning as a geared tab, the rudder tab surface will be moved in a direction opposite to that of the rudder. Operation of the geared tab is controlled by the movement of the rudder, which is transmitted by the geared tab mechanism to the tab.

2-197. RUDDER GEARED TAB MECHANISM. *(See figure 2-25.)* The rudder geared tab mechanism consists of two control rods connected by a crank at the rudder spar. The forward control rod terminates at the rudder trim control drum, installed at the rear spar of the vertical stabilizer at station 112. The aft control rod connects with the horn of the geared tab. Movement of the rudder is transmitted through the geared tab mechanism to the rudder tab to provide aerodynamic assistance for the positioning of the rudder.

2-198. REMOVAL OF RUDDER GEARED TAB MECHANISM.

a. Place the rudder in neutral and position the rudder trim control wheel, on the aft face of the pedestal, so that the trim indicator reads zero degrees.

b. Clamp the rudder tab in the faired position.

c. Disconnect the forward control rod from the mechanism crank on the rudder spar and from the rudder trim control drum. Remove the forward control rod.

d. Disconnect the aft control rod from the geared tab horn and remove it from the aircraft.

e. Remove the bolt securing the mechanism crank to the tab idler fitting on the rudder spar and remove the crank.

2-199. INSTALLATION OF RUDDER GEARED TAB MECHANISM.

a. Place the rudder in neutral (faired with the tail cone).

b. Install the rudder geared tab mechanism crank on the tab idler fitting located on the rudder spar.

c. Connect the aft control rod to the geared tab horn.

d. Secure the forward control rod to the rudder trim control drum.

e. Install the bolt connecting the aft control rod and the forward control rod to the mechanism crank on the rudder spar.

f. Adjust the rudder geared tab control system (see paragraph 2-200).

2-200. ADJUSTMENT OF RUDDER GEARED TAB CONTROL SYSTEM.

a. Check the adjustment of the rudder control system (see paragraph 2-189).

b. Place the rudder pedals in neutral (aligned) and turn the rudder trim control wheel on the aft face of the pedestal so that the trim indicator reads zero degrees. The rudder should be faired with the tail cone and the rudder tab should be faired with the rudder.

Note

With the rudder and rudder tab in the faired positions, mark points in line on the bottom corner of the tab and on the top corner of the lower trailing section of the rudder for measuring rudder tab travel. Measure between these points when the rudder tab is in the extreme LEFT or RIGHT position.

c. Set the forward geared tab control rod so that one-half of the thread of the rod engages with the rudder trim control drum. With the forward control rod in this position, adjust the aft tab control rod to position the tab in neutral.

d. Push the left rudder pedals full forward, $4\frac{3}{8}$ ($\pm\frac{1}{8}$) inches from neutral. The rudder should be LEFT 15 ($\pm\frac{1}{2}$) degrees or 11 $\frac{9}{16}$ ($\pm\frac{1}{4}$) inches from neutral. Check to make certain that the rudder geared tab throw is RIGHT 6 ($\pm\frac{1}{2}$) degrees or $\frac{29}{32}$ ($\pm\frac{1}{8}$) inch from neutral (faired), measured at the lower corner of the rudder tab.

e. Push the right rudder pedals full forward, $4\frac{3}{8}$ ($\pm\frac{1}{8}$) inches from neutral. The rudder should be RIGHT 15 ($\pm\frac{1}{2}$) degrees or 11 $\frac{9}{16}$ ($\pm\frac{1}{4}$) inches from neutral. Check to make certain that the rudder geared tab throw is LEFT 6 ($\pm\frac{1}{2}$) degrees or $\frac{29}{32}$ ($\pm\frac{1}{8}$) inch from neutral (faired), measured at the lower corner of the rudder tab.

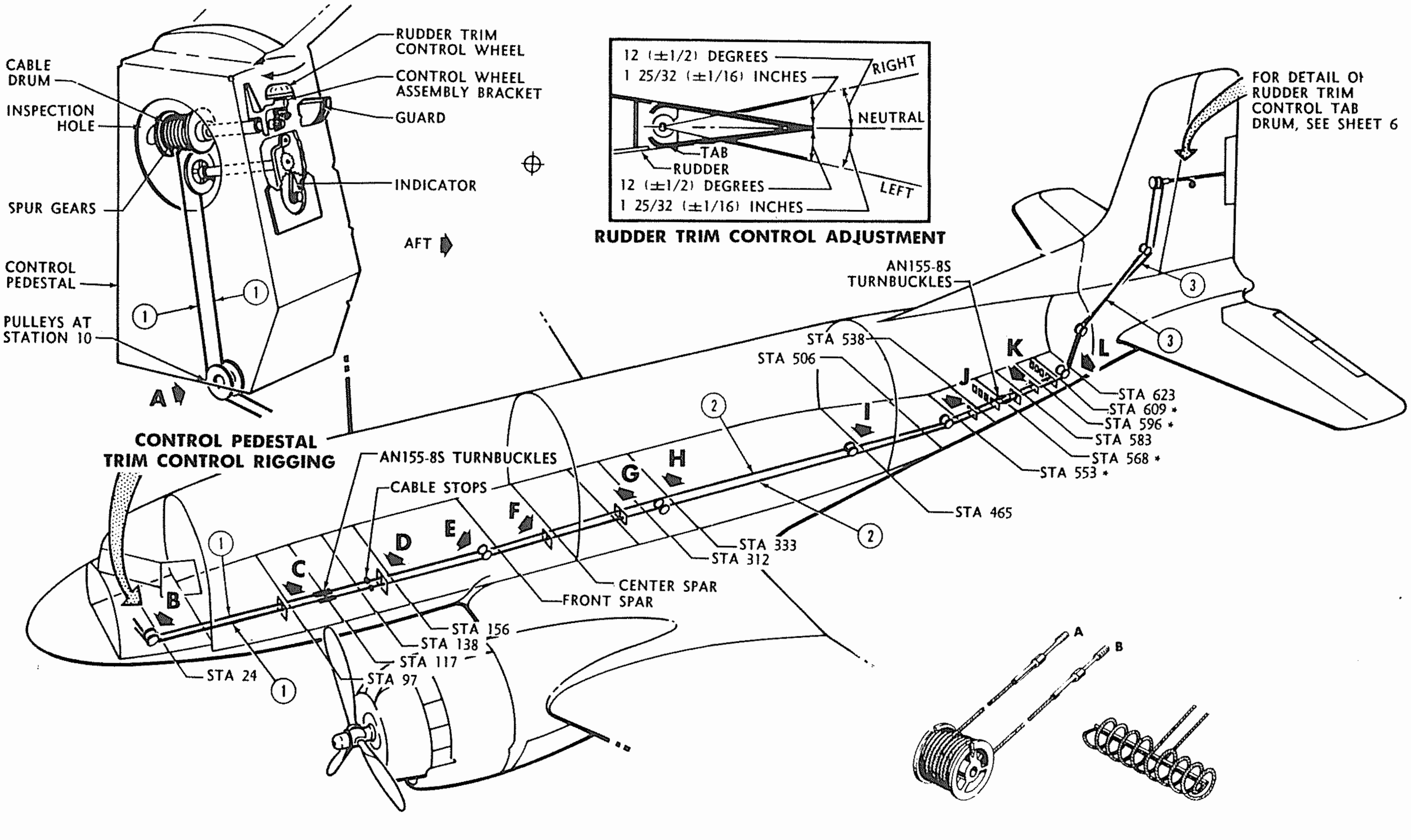
f. If correct throws are not achieved, it will be necessary to adjust the forward rudder geared tab control rod for correct throw, adjust the tab to the faired position with the aft control rod, and check for correct throws of the tab according to steps c and d, preceding.

g. For further information on rigging the rudder geared tab control system, see paragraph 2-231.

2-201. RUDDER TRIM CONTROL SYSTEM. *(See figure 2-26.)* The rudder tab functions both as a geared tab and as a trim control tab. When the tab is used for directional trim, it is controlled by the rudder trim control wheel, on the aft face of the control pedestal, through a cable drum and two-way cable system that terminates at the trim control tab drum, located on a bracket in the vertical stabilizer forward of the rear spar at station 112. Movement is transmitted by the control drum in the stabilizer, through the geared tab mechanism, to the rudder tab.

2-202. RUDDER TRIM CONTROL WHEEL ASSEMBLY AND DRUM. *(See figure 2-26.)* A rudder trim control wheel protrudes from the left aft face of the control pedestal and is accessible to both pilots. It is geared to a shaft, extending through the pedestal, on which the trim control cable drum is installed. The drum is actuated by the rudder trim control wheel and transmits movement, through the two-way cable system, to the rudder trim control tab drum in the

(Continued on Page 170)



RUDDER TRIM CONTROL DRUM WRAPPING PROCEDURE (TYPICAL FOR BOTH DRUMS)

1. Insert cable and pin. Wrap cable on drum from each end to center with equal number of turns on each side.
2. Wrap cable drums so that equal lengths project at "A" and "B."

RUDDER TAB LEFT, NOSE RIGHT

RUDDER TAB RIGHT, NOSE LEFT

A—Pulley Brackets—Station 10, see Sheet 2.
B—Pulley Brackets—Station 26, see Sheet 2.
C—Fairleads—Station 97, see Sheet 2.
D—Fairleads—Station 156, see Sheet 2.
E—Fairleads and Pulley Brackets—Station 222, see Sheet 3.
F—Fairleads—Station 258, see Sheet 3.
G—Fairlead—Station 324, see Sheet 4.
H—Pulley Brackets—Station 333, see Sheet 4.
I—Pulley Brackets—Station 465, see Sheet 4.
J—Pulley Brackets—Station 538, see Sheet 5.
K—Fairleads and Pulley Brackets—Station 583, see Sheet 5.
L—Pulley Brackets—Station 623, see Sheet 5.

Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 6.
2. For cable assembly details, see Sheet 6.
*3. Fairleads shown at these stations are installed on aircraft A through D.
4. Tension cables in accordance with the Cable Rigging Tension Chart, figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values shown in Torque Value Chart, paragraph 2-453.
6. Reference—Douglas drawings 5372516, 5392517, 5372519, 4372521, 4398252 and 5398557.

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Figure 2-26 (Sheet 1 of 6 Sheets). Rudder Trim Control System — Key Drawing

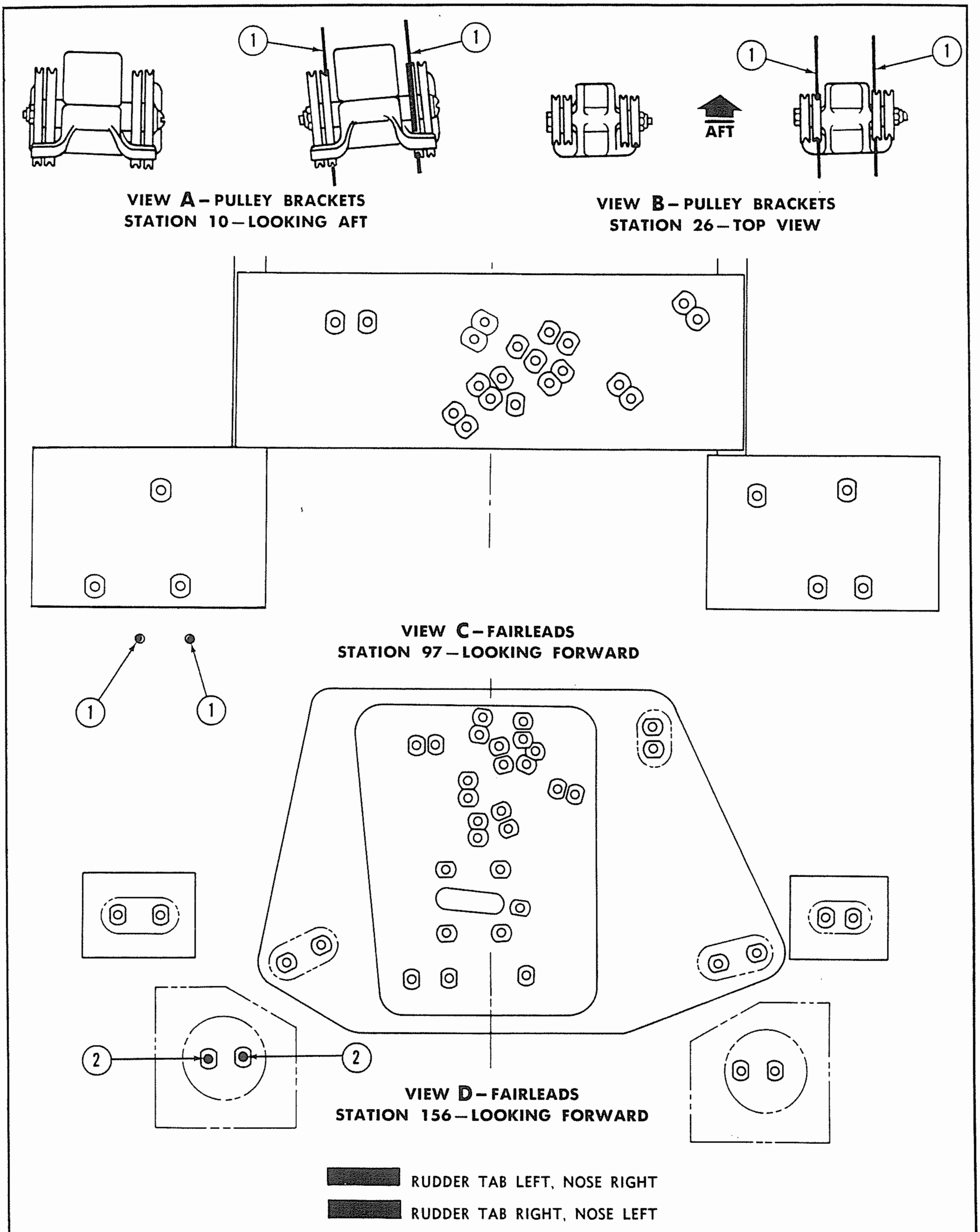
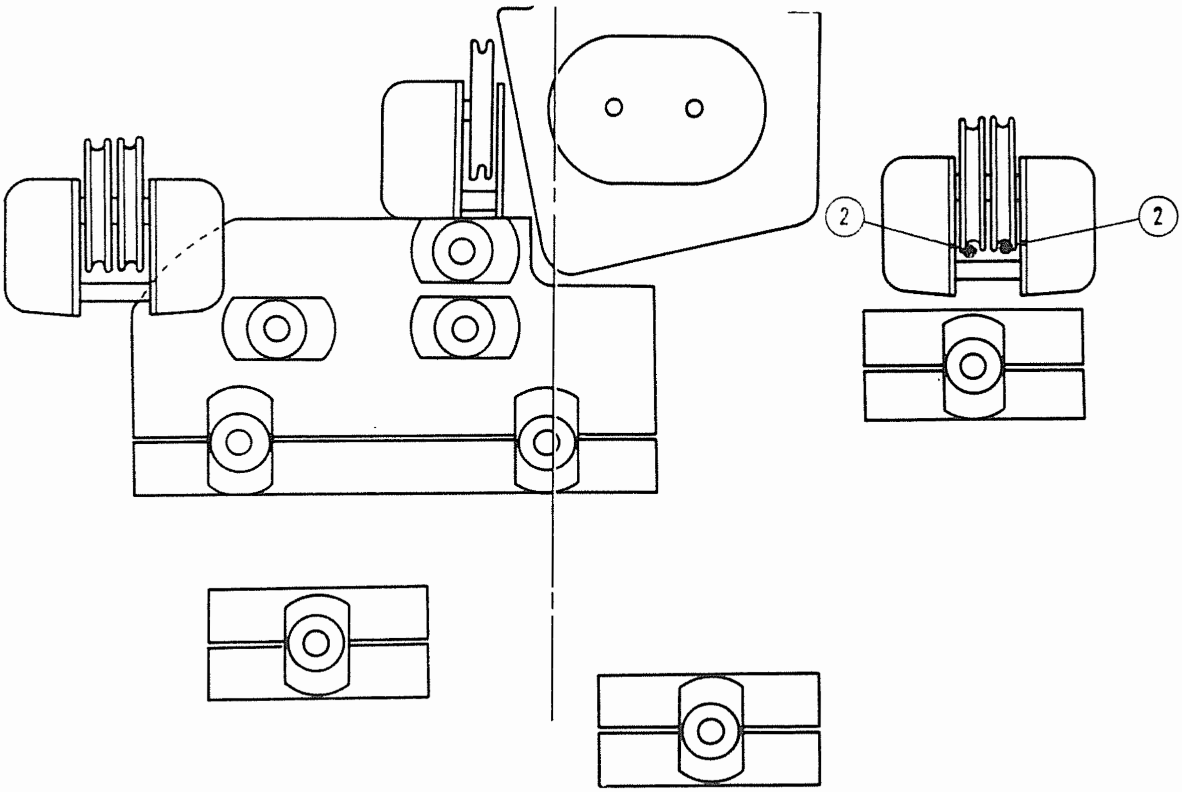
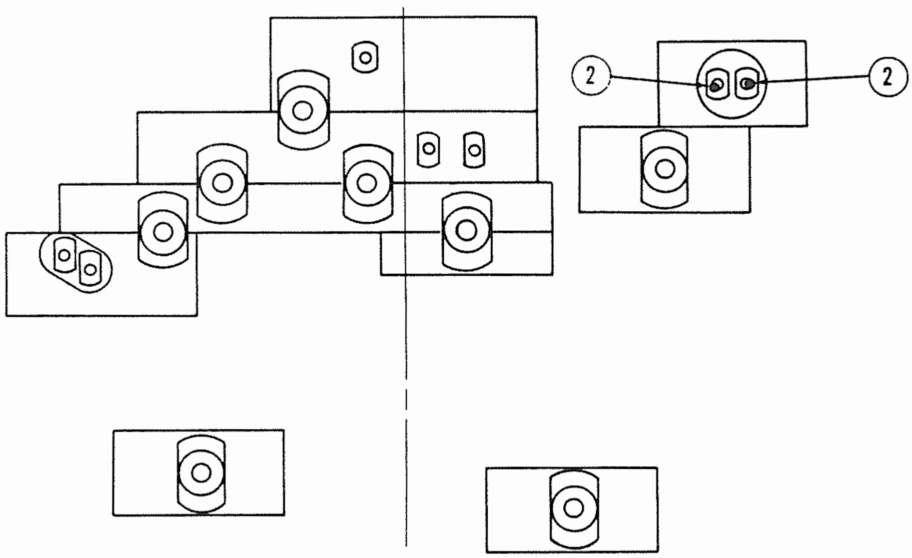


Figure 2-26 (Sheet 2 of 6 Sheets). Rudder Trim Control System – Pulley Brackets, Stations 10 and 26; Fairleads, Stations 97 and 156



VIEW E—FAIRLEADS AND PULLEY BRACKETS
STATION 222, FRONT SPAR—LOOKING AFT



VIEW F—FAIRLEADS
STATION 258, CENTER SPAR—LOOKING AFT

RUDDER TAB LEFT, NOSE RIGHT

RUDDER TAB RIGHT, NOSE LEFT

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Figure 2-26 (Sheet 3 of 6 Sheets). Rudder Trim Control System—Fairleads and Pulley Brackets, Station 222; Fairleads, Station 258

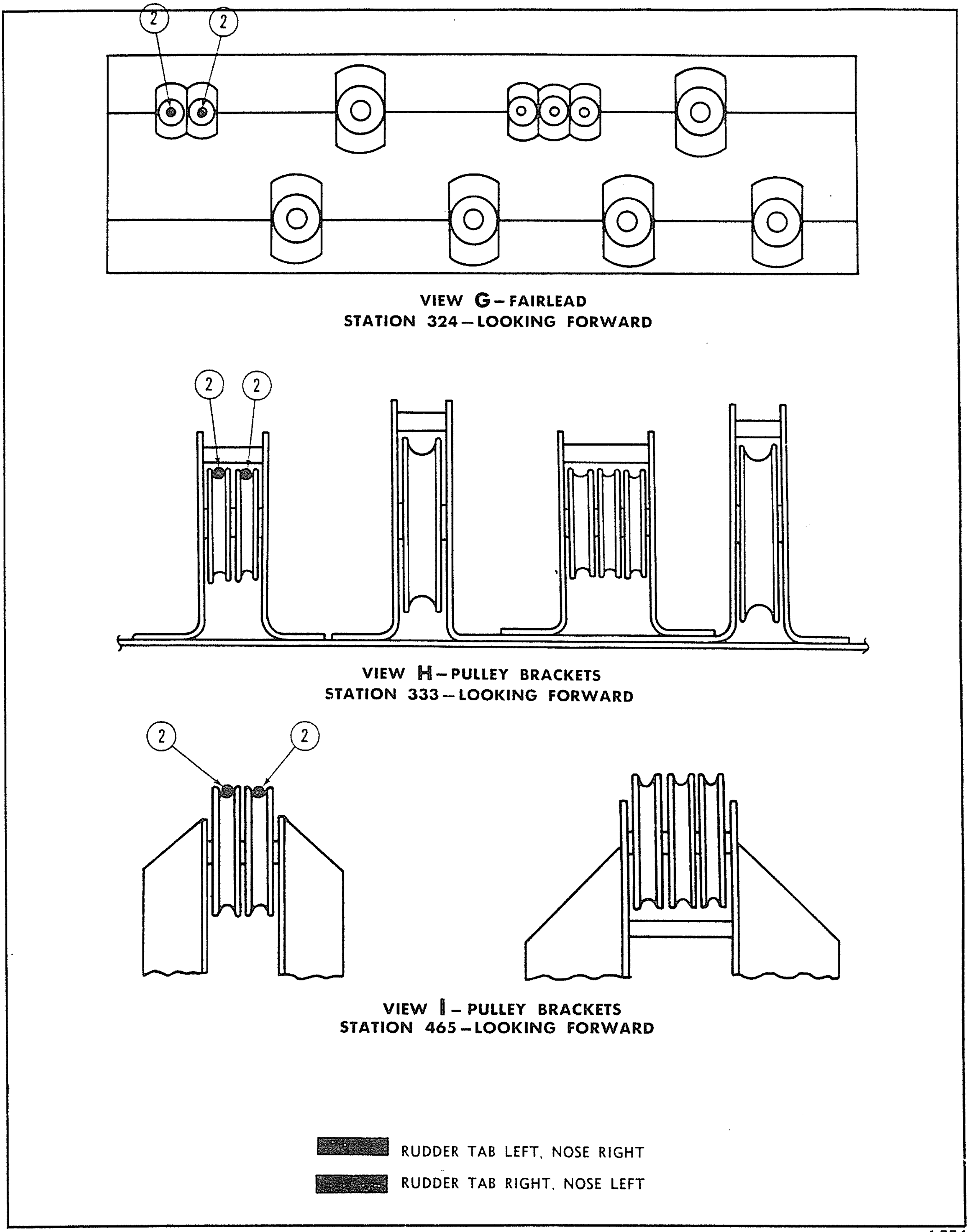


Figure 2-26 (Sheet 4 of 6 Sheets). Rudder Trim Control System—Fairlead, Station 324; Pulley Brackets, Stations 333 and 465

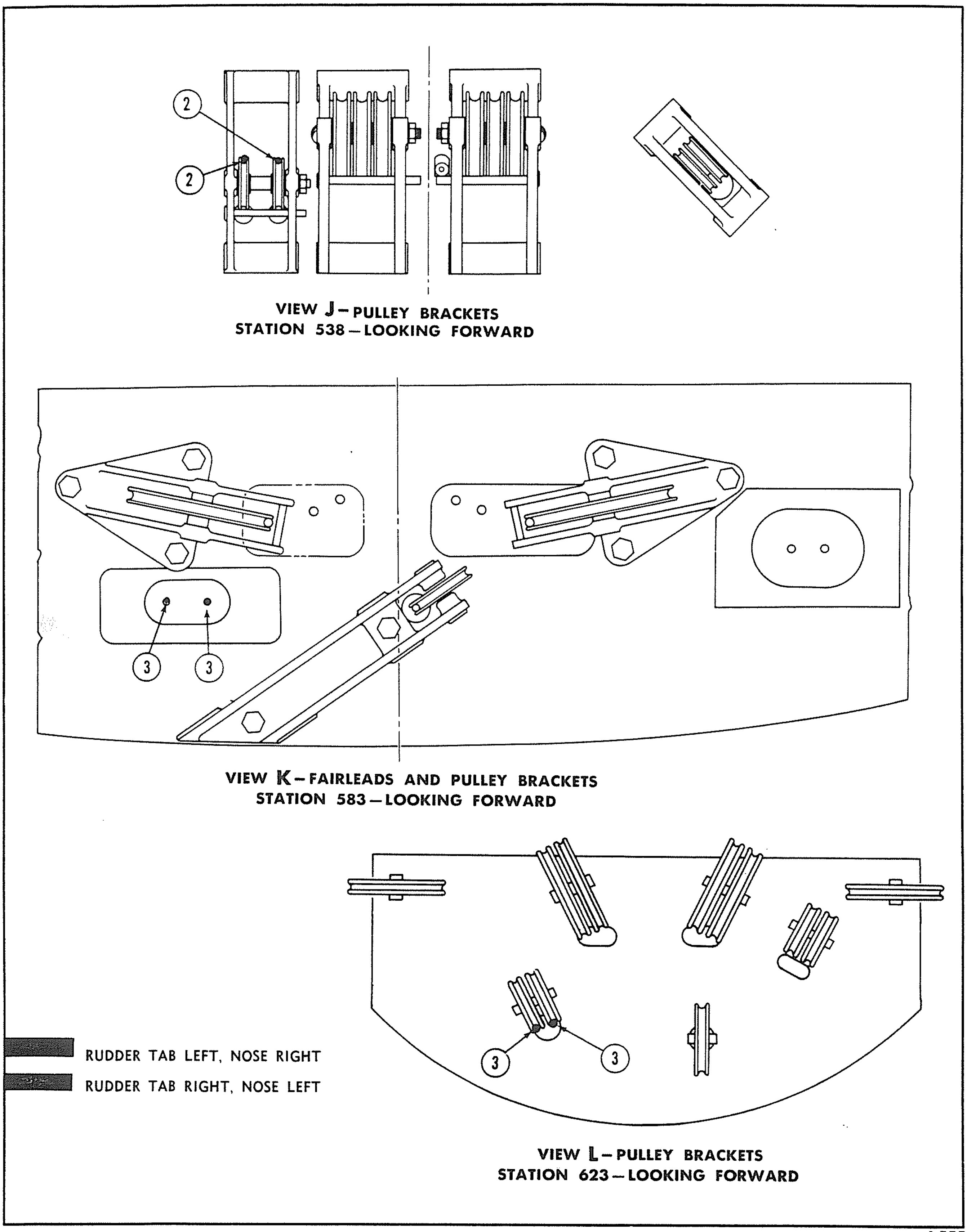
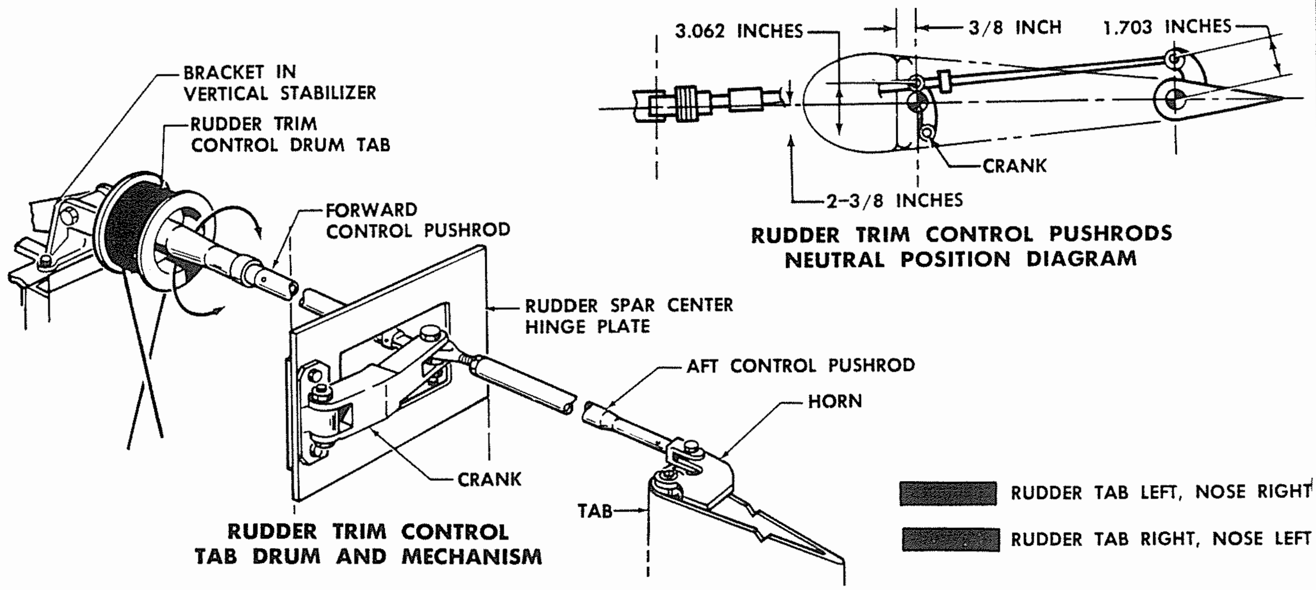


Figure 2-26 (Sheet 5 of 6 Sheets). Rudder Trim Control System — Pulley Brackets, Stations 538 and 623; Fairleads and Pulley Brackets, Station 583



RUDDER TRIM CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)
1	2115350-8 Tab Left Tab Right	1	A	125 3/4	1/16 dia 7x7 flex	S-2049220 -8DS-2R	S-2049220 -8DS-2R		
2	3391412-1 Tab Left Tab Right	2	B	462	1/16 dia 7x7 flex	AN669S2 LH	AN669S2 RH		
3	3391412-589 Tab Left Tab Right	1	C	580	1/16 dia 7x7 flex	AN669S2 LH	AN669S2 LH		

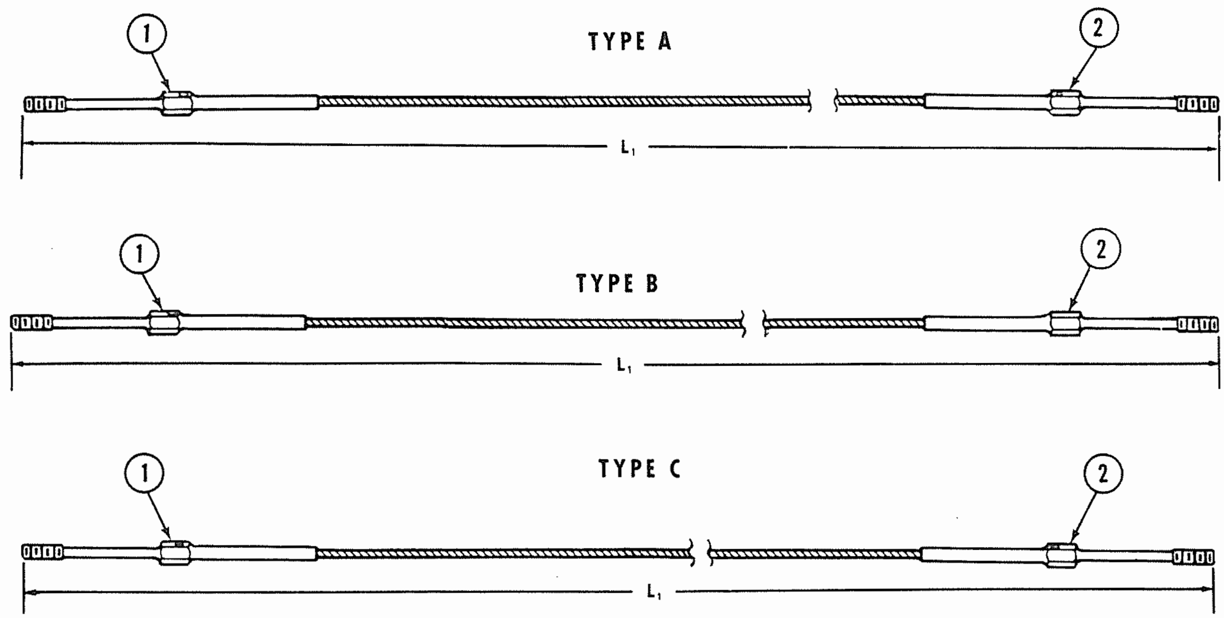


Figure 2-26 (Sheet 6 of 6 Sheets). Rudder Trim Control System — Trim Tab Mechanism, Cable Chart, and Cable Assemblies

(Continued from Page 163)

vertical stabilizer. A rudder trim control indicator, located below the control wheel on the pedestal, records the movement of the cable drum by means of a gearing arrangement.

2-203. REMOVAL OF RUDDER TRIM CONTROL WHEEL ASSEMBLY AND DRUM.

a. Turn the rudder trim control wheel so that the indicator reads zero degrees.

b. Tape cables and drums in pedestal and vertical fin to prevent them from unwrapping.

c. Remove the access doors on the under side of the aircraft and disconnect the rudder trim control cables at station 117.

d. Remove the necessary pulleys from the pulley brackets to release the rudder trim control cables between station 117 and the pedestal.

e. Remove the round inspection plate from the left side of the control pedestal.

f. Release and remove the rudder trim control shaft collar on the forward face of the pedestal.

g. Disconnect the control wheel assembly bracket from the aft face of the control pedestal, and remove the rudder trim control wheel assembly.

h. Remove the shaft collar from the aft end of the shaft, loosen the cable drum on the shaft, hold the cable drum and draw the shaft out of the pedestal.

i. Remove the cable drum and cable from the pedestal.

2-204. MINOR REPAIR AND REPLACEMENT OF RUDDER TRIM CONTROL WHEEL ASSEMBLY AND DRUM. Replace gears if teeth are worn or broken.

2-205. INSTALLATION OF RUDDER TRIM CONTROL WHEEL ASSEMBLY AND DRUM.

a. Place the rudder pedals in neutral and lock the rudder tab in the faired position.

b. Route the forward trim control cables down through the pedestal and aft, replacing necessary pulleys. Attach cables at station 117, but do not tighten the turnbuckles.

c. Set the rudder trim control indicator in neutral.

d. Place the trim control drum so that the teeth of the spur gear on the drum engage with the teeth on the top of the rudder indicator shaft gear (indicator at neutral) and slide the trim control shaft through the cable drum and into position in the forward face of the pedestal.

e. Attach the collars to the ends of the shaft.

f. Secure the cable drum to the shaft.

g. Attach the trim control wheel assembly to the aft face of the pedestal with two screws, locking the driving shaft with the control shaft.

h. Tension the control cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

i. Release the rudder tab and adjust the rudder trim control system (see paragraph 2-217).

j. Safety the turnbuckles.

k. Replace the access doors on the under side of the aircraft and the round inspection plate on the left side of the pedestal.

2-206. RUDDER TRIM CONTROL INDICATOR ASSEMBLY. *(See figure 2-26.)* A rudder trim control indicator assembly, provided to record the movement applied to the rudder tab by the trim control wheel, is located on the left aft face of the control pedestal just below the control wheel.

2-207. REMOVAL OF RUDDER TRIM CONTROL INDICATOR ASSEMBLY.

a. Remove the rudder trim control wheel assembly and drum (see paragraph 2-203).

b. Release the taper pin from the indicator shaft.

c. Release the screws from the indicator bracket and remove the indicator.

d. Work the indicator shaft and gear out of the pedestal.

2-208. MINOR REPAIR AND REPLACEMENT OF RUDDER TRIM CONTROL INDICATOR ASSEMBLY. Replace the gears if the teeth are worn or broken.

2-209. INSTALLATION OF RUDDER TRIM CONTROL INDICATOR ASSEMBLY.

a. Position the rudder trim control indicator shaft and gear in the control pedestal and install the indicator on the indicator bracket.

b. Insert the taper pin in the indicator shaft.

c. Install the rudder trim control wheel assembly and drum (see paragraph 2-205).

2-210. RUDDER TRIM CONTROL TAB DRUM. *(See figure 2-26.)* The rudder trim control tab drum is installed on a bracket in the vertical stabilizer forward of the rear spar at station 112. The drum is actuated by the two-way cable system and transmits movement through the geared tab mechanism to the rudder tab.

2-211. REMOVAL OF RUDDER TRIM CONTROL TAB DRUM.

a. Place the rudder in neutral. Position the rudder trim control wheel so that the trim indicator reads zero degrees.

b. Clamp the rudder tab in the faired position.

c. Tape the cable and drums in the pedestal and vertical stabilizer to prevent them from unwrapping.

d. Remove the cabin floor and disconnect the rudder trim control cables at station 572. Thread the cable ends, remove the pulley guard pins, and pull the cable ends toward the drum.

e. Disconnect the forward control rod from the geared tab mechanism crank at the rudder spar and from the rudder trim control tab drum. Remove the control rod.

f. Remove the drum from the bracket in the vertical stabilizer.

2-212. MINOR REPAIR AND REPLACEMENT OF RUDDER TRIM CONTROL TAB DRUM. If worn or damaged, replace the bearing, bushing, or oil seal.

2-213. INSTALLATION OF RUDDER TRIM CONTROL TAB DRUM.

a. Place the rudder pedals in neutral and position the rudder trim control wheel so that the trim indicator reads zero degrees.

b. Clamp the rudder tab in the faired position.

c. Install the rudder trim control tab drum on the bracket in the vertical stabilizer and connect the forward control rod to the drum and to the geared tab mechanism crank at the rudder spar.

d. Route the rudder trim control cables down through the vertical stabilizer into the fuselage and connect them to the intermediate cables at station 572. Install the necessary guard pins and fairlead grommets.

e. Untape the cables and drums in the pedestal and vertical stabilizer.

f. Tension the control cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

g. Unclamp the rudder tab and adjust the rudder trim control system (see paragraph 2-217).

h. Safety the turnbuckles.

2-214. RUDDER TRIM CONTROL CABLES. *(See figure 2-26.)* A two-way cable system originating at the rudder trim control drum in the pedestal extends aft through the fuselage to the tail section and upward through the vertical stabilizer to the rudder trim control tab drum. Cable stops, attached to the rudder trim control cables, limit the movement of the cables by striking a fixed fairlead at station 156 in the fuselage. The stops are not adjustable.

2-215. REMOVAL OF RUDDER TRIM CONTROL CABLES.

a. Place the rudder pedals in neutral and position the rudder trim control wheel so that the position indicator reads zero degrees.

b. Clamp the rudder tab in the faired position.

c. Tape the cables and drums in pedestal and vertical fin to prevent them from unwrapping.

d. Remove the rudder trim control drum from the pedestal (see paragraph 2-203). Unwrap the forward cable from the drum.

e. Remove the cabin floor and disconnect the rudder trim control cables at station 572. Thread the cables.

f. Remove the necessary fairlead grommets and pulley guard pins and draw the intermediate control cables forward. Remove the cables through the access door on the under side of the fuselage forward of station 156.

g. Remove necessary pulley guard pins and pull the aft cable ends toward the rudder trim control tab drum at station 112 in the vertical stabilizer.

h. Disconnect the forward control rod from the geared tab mechanism crank at the rudder spar and from the trim control tab drum. Remove the control rod.

i. Release the drum from the bracket in the vertical stabilizer. Unwrap the aft cable from the drum.

2-216. INSTALLATION OF RUDDER TRIM CONTROL CABLES.

a. Place the rudder pedals in neutral and clamp the rudder tab in the faired position.

b. Wrap the aft rudder trim control cable on the drum *(see figure 2-26),* and install the drum on the bracket in the vertical stabilizer.

c. Connect the forward control rod to the trim control tab drum and to the geared tab mechanism crank at the rudder rear spar.

d. Route the aft rudder trim control cables down through the vertical stabilizer into the fuselage, installing necessary guard pins and fairlead grommets.

e. Route the intermediate rudder trim control cables aft through the fuselage, installing necessary guard pins and fairlead grommets, and connect the cables with turnbuckles at station 572.

f. Wrap the forward rudder trim control cable on the drum *(see figure 2-26),* and install the rudder trim control drum in the pedestal (see paragraph 2-205).

g. Route the forward cables aft through the fuselage and connect the cables with turnbuckles at station 117.

h. Tension the control cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9).*

i. Unclamp the rudder tab and adjust the rudder trim control system (see paragraph 2-217).

j. Safety the turnbuckles.

k. Replace access doors and floor panels.

2-217. ADJUSTMENT OF RUDDER TRIM CONTROL SYSTEM.

a. Place the rudder pedals in the neutral position (aligned). Turn the rudder trim control wheel on the aft face of the pedestal so that the rudder trim indicator reads zero degrees. The rudder tab should be faired with the rudder.

Note

With the rudder tab in the faired position, mark points in line on the bottom corner of the tab and on the top corner of the lower trailing section of the rudder for measuring the rudder tab travel. Measure between these points when the tab is in the extreme LEFT or RIGHT position.

b. Retain the rudder pedals in neutral position (aligned) and turn the rudder trim control wheel in a counterclockwise direction until the trim control indicator reads full NOSE LEFT. The rudder tab should be RIGHT 12 ($\pm\frac{1}{2}$) degrees or 1 $\frac{25}{32}$ ($\pm\frac{1}{16}$) inches from neutral (faired position), measured at the lower corner of the rudder tab.

c. Retain the rudder pedals in neutral position (aligned) and turn the rudder trim control wheel in a clockwise direction until the trim control indicator reads full NOSE RIGHT. The rudder tab should be LEFT 12 ($\pm\frac{1}{2}$) degrees or 1$\frac{25}{32}$ ($\pm\frac{1}{16}$) inches from neutral (faired position), measured at the lower corner of the rudder tab.

d. Tension the cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

e. Safety the turnbuckles.

f. If correct throws are not achieved for the rudder tab, it will be necessary to check the rigging of the rudder trim control system *(see figure 2-26),* and also the adjustment of the rudder geared tab control system (see paragraph 2-200).

g. For further information on rigging the rudder trim control system (see paragraph 2-231).

2-218. WING FLAPS. Four metal wing flaps of the split trailing-edge type are provided. The flaps extend between the inboard ends of the ailerons and function as a single unit. They are hinged to the under side of the aircraft and wings and are hydraulically operated.

2-219. REMOVAL OF WING FLAP.

a. With the flaps lowered, disconnect the turnbuckles that connect the flap to the operating rods.

b. Pull the pins out of the wing flap hinge.

c. Remove the wing flap by pulling it downward.

2-220. MINOR REPAIR AND REPLACEMENT OF WING FLAP. Replace worn hinge pins, or hinges.

2-221. INSTALLATION OF WING FLAP. *(See figure 2-27.)*

a. Support the wing flap in position and install the pins in the wing flap hinge.

b. Connect the operating rods to the wing flap with turnbuckles.

c. Check the adjustment of the wing flap control system (see paragraph 2-230).

2-222. WING FLAP CONTROL SYSTEM. *(See figure 2-28.)* Wing flap movement is controlled by a wing flap hydraulic valve located on the hydraulic control panel in the flight compartment. The flaps are lowered by moving the handle from the NEUTRAL position to the DOWN position and are raised by placing the handle in the UP position. Tubing extends from the hydraulic control valve to the actuating cylinder. Both the cylinder and the piston move when hydraulic pressure is applied, actuating toggle rods, which are connected to them in the form of a variable parallelogram. The toggle rods are connected to push-pull rods that extend into the wings and connect to turnbuckles attached to the wing flaps. The extension of the cylinder and piston causes the push-pull rods to move inward and lower the flaps. Contraction of the cylinder and piston causes the wing flaps to rise. The position of the wing flaps is shown on an indicator located on the left side of the flight compartment. A wire, installed in a conduit, extends from the indicator to the forward end of the hydraulic cylinder. Any movement of the cylinder is shown by an equivalent movement of the arrow on the indicator.

2-223. WING FLAP HYDRAULIC SYSTEM. *(See figure 2-29.)* The wing flap hydraulic system is composed of the control valve, the relief valve, the thermal relief valve, and the actuating cylinder. Hydraulic fluid at system pressure is directed to the wing flap components from the pressure regulator to extend or retract the flaps. Application of system pressure to the forward of the piston causes the flaps to extend and pressure applied to the aft side of the piston causes the flaps to retract. A relief valve is installed in the system to prevent damage to the retracting mechanism or to its attaching structure due to gust loads encountered in flight. A thermal relief valve is installed in the system to prevent possible damage to the mechanism due to excessively high temperatures, which might be encountered either in flight or on the ground. The thermal relief valve is effective only when the flaps are in the retracted position.

2-224. WING FLAP ACTUATING CYLINDER. The wing flap actuating cylinder is mounted on rollers and located in the control tunnel in the center wing. The supports that carry the rollers are clamped to each end of the actuating cylinder. The rollers operate on rails mounted in the tunnel in the center wing. Four rods are connected to the piston and actuating cylinder, forming a variable parallelogram. The rods also

(Continued on Page 177)

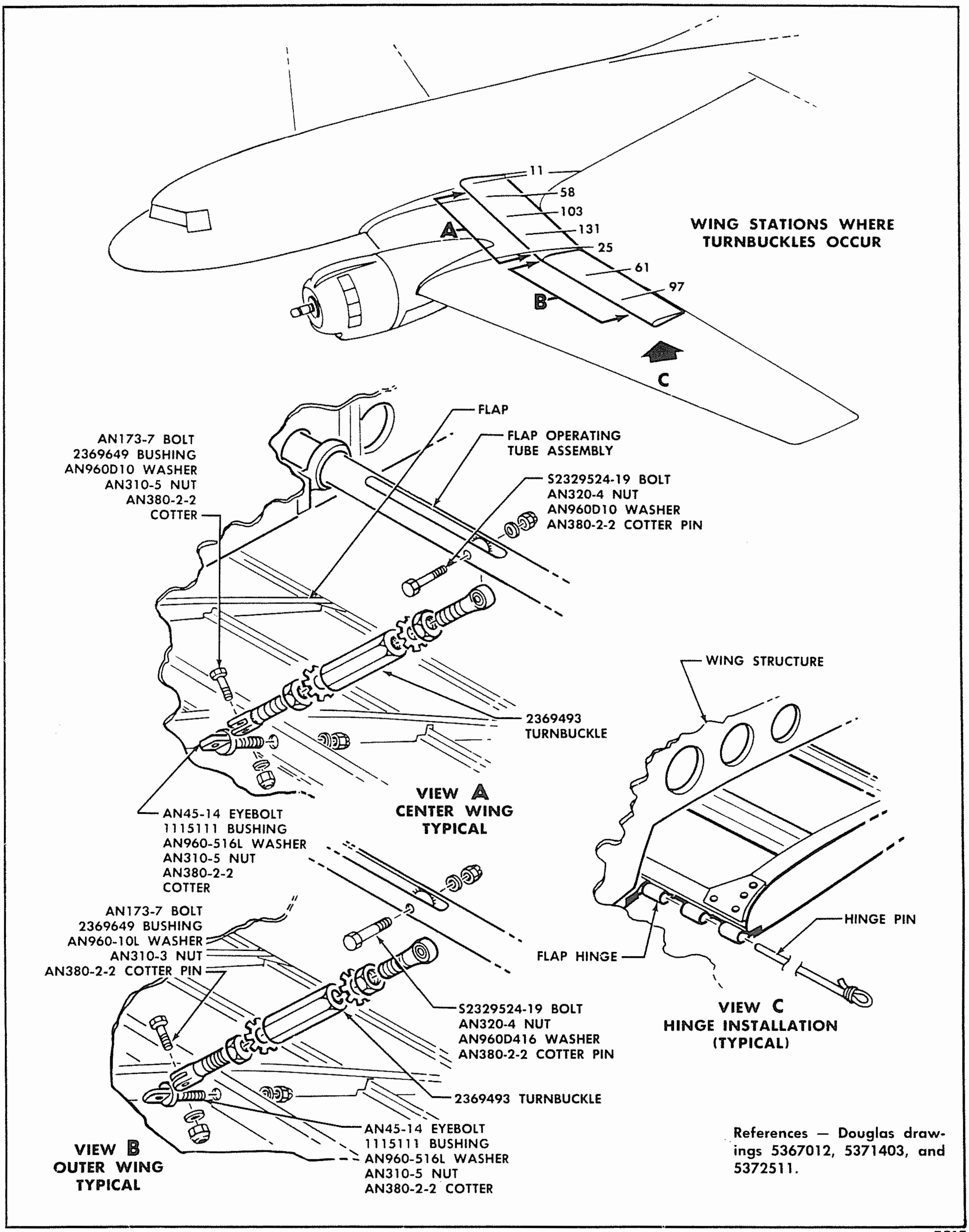


Figure 2-27. Installation of Wing Flaps

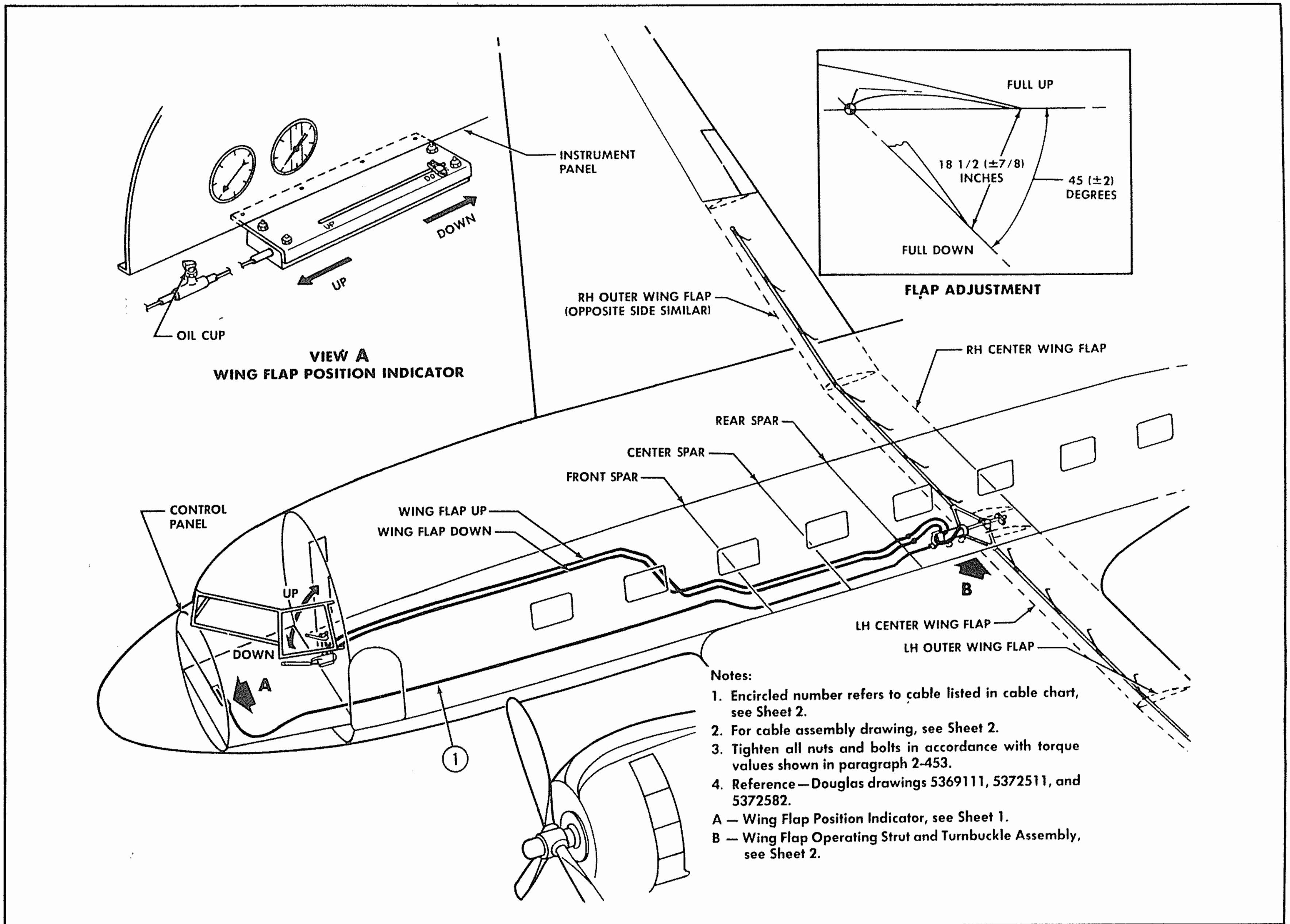
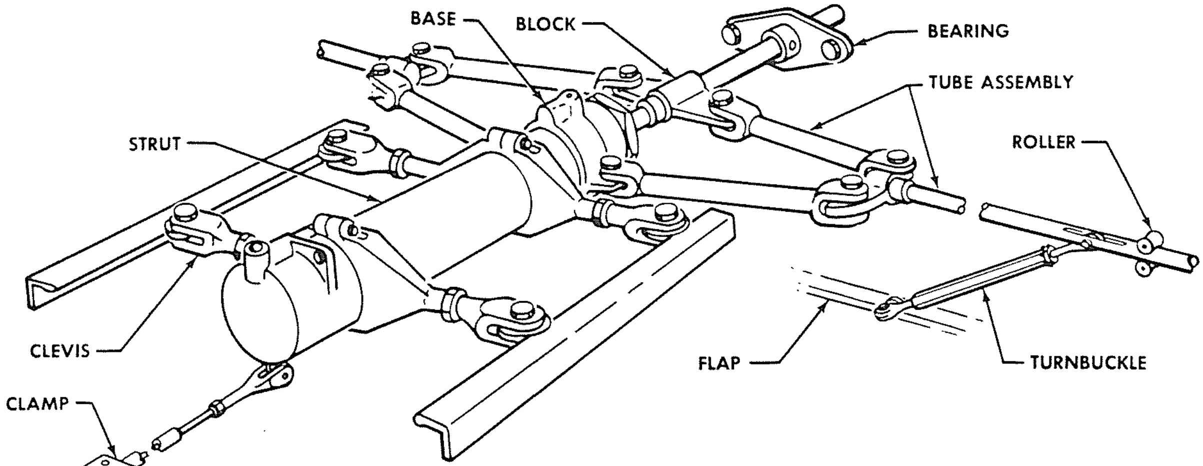


Figure 2-28 (Sheet 1 of 2 Sheets). Wing Flap Control System — Key Drawing



VIEW B
WING FLAP OPERATING STRUT ASSEMBLY

WING FLAP INDICATOR CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L1)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)
1	3390041	1		339	.063 music wire	AN468-3	1119754	1119755	1119756
						(5) 3390041-9	(6) 3390041-7	(7) 3390041-13	(8) 3390041-7
						(9) 3390041-15	(10) 3390041-5	(11) AN295-40	(12) 3390041-17
						(13) 3390041-11	(14) 1119753	(15) 1119757	(16) 1119758
						(17) 2460440	(18) AN507-440R3	(19) 2460439	

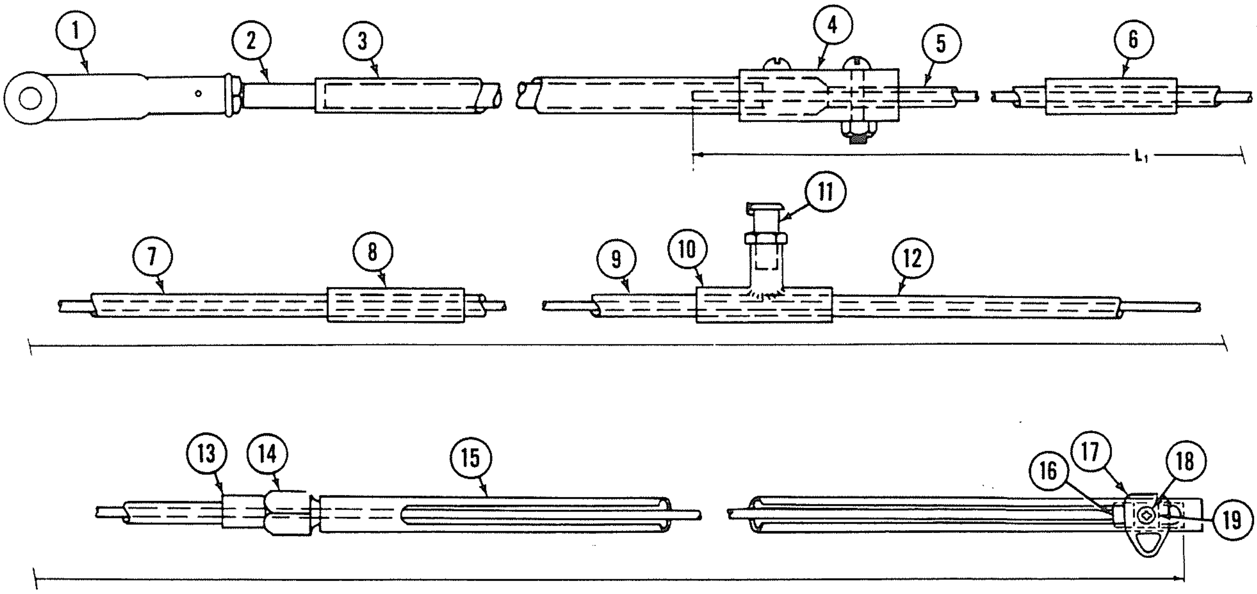


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Figure 2-28 (Sheet 2 of 2 Sheets). Wing Flap Control System — Strut Assembly, Cable Chart, and Cable Assembly

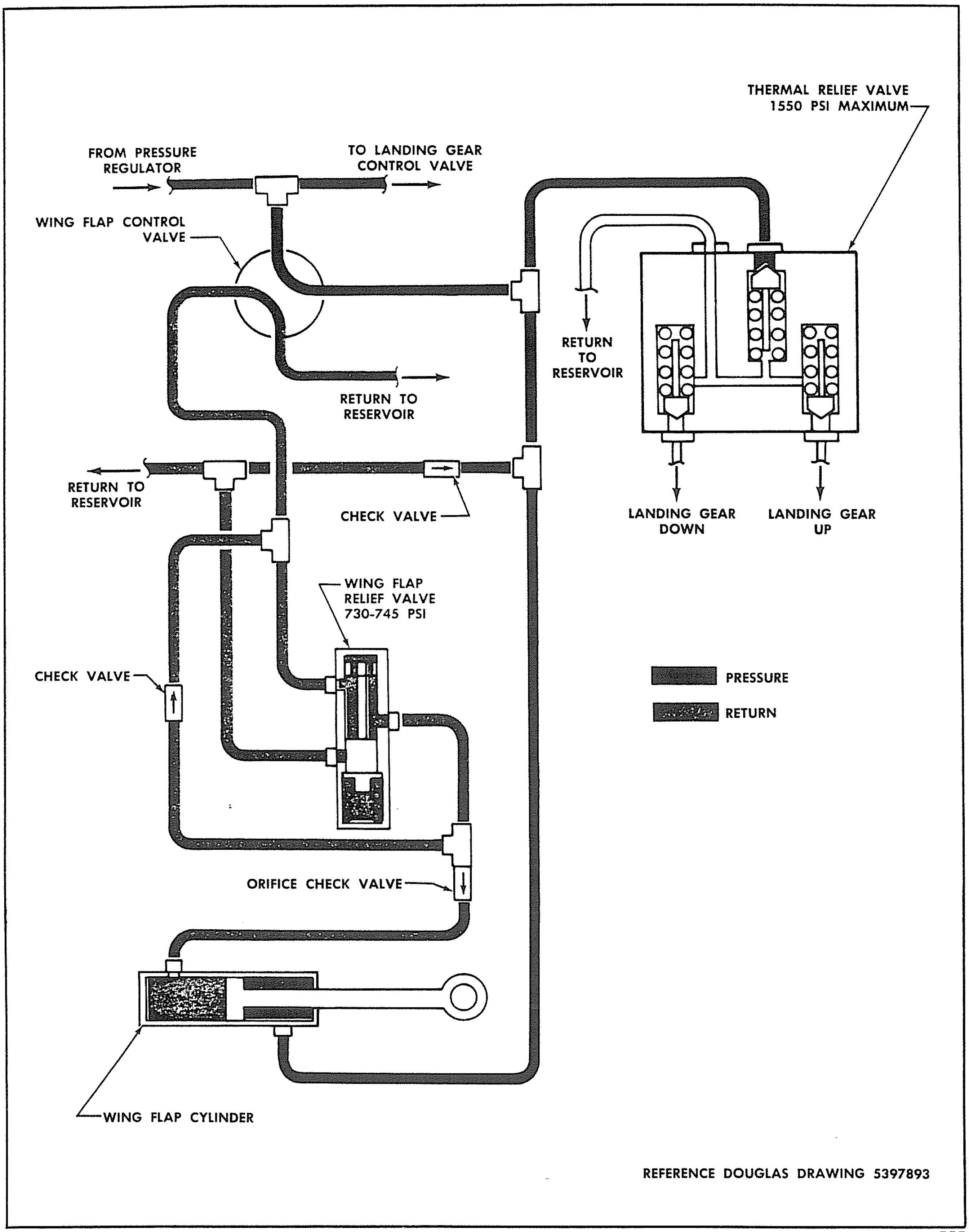

Figure 2-29. Wing Flap Hydraulic System

(Continued from Page 172)

are attached to the push-pull rods in the wings. The push-pull rods are connected to the wing flaps by rods and turnbuckles. When the actuating cylinder extends, the cylinder moves forward on the rollers to produce an equalized force through the operating rods, pulling the push-pull rods inwardly and lowering the wing flaps. When the piston and cylinder retract, the push-pull rods are forced outward and the wing flaps are raised.

2-225. REMOVAL OF WING FLAP ACTUATING CYLINDER.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and place the wing flap control valve handle in neutral.

b. Remove the cover plate from the lower surface of the wing center section to gain access to the actuating cylinder.

c. Disconnect and cap the hydraulic pipes leading to the wing flap actuating cylinder.

d. Disconnect the wing flap indicator wire from the wing flap actuating cylinder.

e. Disconnect the rods that attach to the piston rod and actuating cylinder.

f. Remove the roller bolts from one side of the actuating cylinder and remove the rollers.

g. Unbolt the auxiliary bearing on the piston rod.

h. Move the cylinder forward and up until it can be removed by backing it down and out through the cutout in the rib.

CAUTION

Do not remove the roller supports from the actuating cylinder, since these are adjusted at the factory to insure alignment with the piston rod bearings.

2-226. INSTALLATION OF WING FLAP ACTUATING CYLINDER. Reverse the removal procedure.

2-227. WING FLAP INDICATOR CABLE. *(See figure 2-28.)* The wing flap indicator cable extends from the indicator in the flight compartment aft to the flap actuating cylinder at the trailing edge of the wing center section on the center line of the aircraft. The control wire is encased within a flexible housing and is attached to the aircraft structure with clips.

2-228. REMOVAL OF WING FLAP INDICATOR CABLE.

a. Remove the screws, plug, and pointer from the end of the wire, at the indicator.

b. Remove the bolts attaching the indicator to the bracket on the instrument panel.

c. Remove the bolt and nut attaching the wire assembly to the actuating cylinder.

d. Remove the clips attaching the cable assembly to the aircraft structure.

2-229. INSTALLATION OF WING FLAP INDICATOR CABLE.

a. Attach the wing flap indicator cable assembly to the aircraft structure with clips.

b. Connect the aft end of the indicator wire to the actuating cylinder at the trailing edge of the wing center section on the center line of the aircraft.

c. Connect the indicator assembly to the bracket on the instrument panel.

d. Install the pointer, plug, and screws to the indicator end of the wire.

e. Adjust the indicator assembly so that it reads full DOWN when the flaps are lowered.

2-230. ADJUSTMENT OF WING FLAP CONTROL SYSTEM.

a. Check all movable joints, rollers, bearings, and guides for freedom of movement.

b. Place the wing flap control handle in the DOWN position and, with the hand pump, slowly pump the flaps into the DOWN position. Adjust the turnbuckles until the trailing edges of the wing flaps are DOWN 45 ($\pm$ 2) degrees or 18½ ($\pm$ ⅞) inches from the trailing edges of the wings. The flap position indicator should read WING FLAPS DOWN.

c. Adjust the flap operating rods until the angle of the turnbuckles is approximately 75 degrees with relation to the rods.

d. Place the wing flap control handle in the UP position and, with the hand pump, slowly pump the flaps into the UP position, noting any differences in travel and roller clearances. The flap position indicator should read WING FLAPS UP.

e. When the flap rods have been adjusted so that all surfaces have equal travel, and trailing edges are in line, tighten the turnbuckles until the flaps press lightly on the felt pads when in the full UP position.

CAUTION

The wing flaps must not be preloaded with the actuating cylinder. When the flaps are pressing lightly against the felt pads, the actuating cylinder must be completely bottomed.

f. For further information on rigging the wing flap control system, see paragraph 2-231.

2-231. SURFACE CONTROLS RIGGING CHART.

Surface Control System	*Flight Compartment Control Position*	*Surface Position*	*Stop Position*
a. *Aileron and aileron geared tab.	Pilots' control wheels are neutral. Aileron trim indicator reads zero degrees.	Ailerons are faired with wing trailing edge. Aileron tabs are faired with aileron trailing edge.	
	Pilots' control wheels are turned approximately 180 degrees clockwise from neutral.	Right aileron is up 20 ($\pm \frac{1}{2}$) degrees or $9\frac{1}{16}$ ($\pm \frac{1}{4}$) inches from neutral. Left aileron is down 14 ($\pm \frac{1}{2}$) degrees or $6\frac{3}{8}$ ($\pm \frac{1}{4}$) inches from neutral. Right aileron geared tab is down 8 ($\pm \frac{1}{2}$) degrees or $1\frac{5}{32}$ ($\pm \frac{1}{8}$) inches from neutral. Left aileron geared tab is up 5.6 ($\pm \frac{1}{2}$) degrees or $\frac{25}{32}$ ($\pm \frac{1}{8}$) inch from neutral.	Surface stops and aileron control wheel stops in contact.
	Pilots' control wheels are turned approximately 180 degrees counterclockwise from neutral.	Left aileron is up 20 ($\pm \frac{1}{2}$) degrees or $9\frac{1}{16}$ ($\pm \frac{1}{4}$) inches from neutral. Right aileron is down 14 ($\pm \frac{1}{2}$) degrees or $6\frac{3}{8}$ ($\pm \frac{1}{4}$) inches from neutral. Left aileron geared tab is down 8 ($\pm \frac{1}{2}$) degrees or $1\frac{5}{32}$ ($\pm \frac{1}{8}$) inches from neutral. Right aileron geared tab is up 5.6 ($\pm \frac{1}{2}$) degrees or $\frac{25}{32}$ ($\pm \frac{1}{8}$) inch from neutral.	Surface stops and aileron control wheel stops in contact.
b. *Aileron trim.	Pilots' control wheels are neutral. Aileron trim indicator reads zero degrees (neutral).	Ailerons are faired with wing trailing edge. Aileron tabs are faired with aileron trailing edge.	
	Pilots' control wheels are neutral. Aileron trim indicator reads full RIGHT WING DOWN.	Right aileron tab is down 12 ($\pm \frac{1}{2}$) degrees or $1\frac{15}{16}$ ($\pm \frac{3}{32}$) inches from neutral.	Cable stops contact.
	Pilots' control wheels are neutral. Aileron trim indicator reads full LEFT WING DOWN.	Right aileron tab is up 12 ($\pm \frac{1}{2}$) degrees or $1\frac{15}{16}$ ($\pm \frac{3}{32}$) inches from neutral.	Cable stops contact.
c. Elevator spring control and elevator tab (inboard tabs).	Pilots' control columns are neutral (10 degrees on aircraft A, B, and 1 through 17, or $11\frac{1}{2}$ degrees on aircraft C, D, 18 and subsequent, forward of vertical).	Elevators are neutral with trailing edge faired with tail cone. Spring control tabs are up 5 ($\pm \frac{1}{2}$) degrees or $\frac{3}{4}$ ($\pm \frac{1}{8}$) inch from elevator trailing edge.	
	Pilots' control columns are full forward (24 degrees on aircraft A, B, and 1 through 17, or approximately $25\frac{1}{2}$ degrees on aircraft C, D, 18 and subsequent, forward of vertical).	Elevators are full down 20 ($\pm \frac{1}{2}$) degrees or $10\frac{13}{32}$ ($\pm \frac{1}{4}$) inches from neutral. Elevator spring control tabs are up 20 ($\pm \frac{1}{2}$) degrees or $2\frac{3}{4}$ ($\pm \frac{1}{8}$) inches (aircraft A, B, and 1 through 17), or 18 ($\pm \frac{1}{2}$) degrees or $2\frac{3}{4}$ ($\pm \frac{1}{8}$) inches (aircraft C, D, 18 and subsequent), from elevator trailing edge.	Surface stops and column stops in contact.
	Pilots' control columns are full aft (5 degrees on aircraft A, B, and 1 through 17, or approximately $3\frac{1}{2}$ degrees on aircraft C, D, 18 and subsequent, aft of vertical).	Elevators are full up 20 ($\pm \frac{1}{2}$) degrees or $10\frac{13}{32}$ ($\pm \frac{1}{4}$) inches from neutral. Elevator spring control tabs are down 10 ($\pm \frac{1}{2}$) degrees or $1\frac{1}{2}$ ($\pm \frac{1}{8}$) inches from elevator trailing edge.	Surface stops and column stops in contact.
d. Elevator trim tab (outboard tabs).	Pilots' control columns are neutral. Elevator trim tab indicator reads zero degrees (neutral).	Left and right elevator trim tabs are down 6 ($\pm \frac{1}{2}$) degrees or $\frac{29}{32}$ ($\pm \frac{1}{8}$) inch from elevator trailing edge.	
	Pilots' control columns are neutral. Elevator trim tab indicator reads full NOSE DOWN.	Left and right elevator trim tabs are faired with elevator trailing edge.	Cable stops contact.
	Pilots' control columns are neutral. Elevator trim tab indicator reads full NOSE UP.	Left and right elevator trim tabs are down 25 (+ 1, − 0) degrees or $3\frac{3}{4}$ (+ $\frac{1}{4}$, − 0) inches from elevator trailing edge.	Cable stops contact.

*The rudder and right aileron geared tabs also function as trim control tabs. The functions should be checked separately as indicated.

SURFACE CONTROLS RIGGING CHART *(Continued)*

Surface Control System	*Flight Compartment Control Position*	*Surface Position*	*Stop Position*
e. *Rudder and rudder geared tab.	Pilots' rudder pedals are neutral. Rudder trim indicator reads zero degrees.	Rudder and rudder geared tab are faired with trailing edge.	
	Pilots' left rudder pedals are full forward 4⅜ inches from neutral.	Rudder is left 15 (±½) degrees or 11 $\frac{9}{16}$ (±¼) inches from neutral. Rudder geared tab is right 6 (±½) degrees or $\frac{29}{32}$ (±⅛) inch from rudder trailing edge.	Surface stops and rudder pedal stops in contact.
	Pilots' right rudder pedals are full forward 4⅜ inches from neutral.	Rudder is right 15 (±½) degrees or 11 $\frac{9}{16}$ (±¼) inches from neutral. Rudder geared tab is left 6 (±½) degrees or $\frac{29}{32}$ (±⅛) inch from rudder trailing edge.	Surface stops and rudder pedal stops in contact.
f. *Rudder trim.	Pilots' rudder pedals are neutral. Rudder trim indicator reads zero degrees (neutral).	Rudder is faired with tail cone. Rudder tab is faired with rudder trailing edge.	
	Pilots' rudder pedals are neutral. Rudder trim indicator reads full NOSE LEFT.	Rudder tab is right 12 (±½) degrees or 1 $\frac{25}{32}$ (±$\frac{1}{16}$) inches from rudder trailing edge.	Cable stops contact.
	Pilots' rudder pedals are neutral. Rudder trim indicator reads full NOSE RIGHT.	Rudder tab is left 12 (±½) degrees or 1 $\frac{25}{32}$ (±$\frac{1}{16}$) inches from rudder trailing edge.	Cable stops contact.
g. Wing flap.	Wing flap control handle in up position. Flap position indicator reads WING FLAPS UP.	Wing flaps are full up.	
	Wing flap control handle in down position. Flap position indicator reads WING FLAPS DOWN.	Wing flaps are down 45 (± 2) degrees or 18½ (±⅞) inches.	

*The rudder and right aileron geared tabs also function as trim control tabs. The functions should be checked separately as indicated.

All instructions are given on the basis the aircraft is being viewed looking forward.

All measurements are to be taken at the root of the surfaces.

Rigging tensions given in the Cable Rigging Tension Chart, figure 2-9, should be followed for all systems except the following:

1. Single cables of the aileron control system, just forward of the aileron cross in the fuselage, should be adjusted to a $\frac{3}{16}$-inch cable load.
2. Single cables of the elevator trim tab control system, forward of the splice in the fuselage, should be adjusted to ½ the normal cable tension.
3. Cables of the elevator control system should be adjusted to the $\frac{5}{32}$-inch standard rig load.

Surface control systems should be rigged with all cable terminal attaching points rigidly held in neutral position.

All control surface tabs are to have a maximum looseness of $\frac{1}{16}$ inch at the trailing edge of the tab.

All of the control element travels include full surface throw plus system stretch.

Control column down-spring setting should be 30 pounds force in the forward direction (nose down direction).

The output torque of the aileron, elevator, and rudder servos shall be 120 inch-pounds minimum, and 135 inch-pounds maximum, with the servos installed in the aircraft. This corresponds to 60 pounds minimum, and 67½ pounds maximum, tension in the cables wrapped on the servo drums.

2-232. SURFACE CONTROLS BOLT TORQUE VALUES. For information on surface controls bolt torque values, see paragraph 2-453.

2-233. FUSELAGE.

2-234. DESCRIPTION. The aluminum-alloy semi-monocoque fuselage is constructed of lateral floor beams, lateral and longitudinal stiffeners, transverse frames, ribs and bulkheads, and stressed skin. The crew compartment forward of the bulkhead at station 97 is 8 feet 1 inch long, and the cabin or cargo space extends aft 43 feet 9 inches. The maximum cross-section inside height of the fuselage is 6 feet 9 inches, and the maximum inside width is 8 feet 1 inch. The wing center section is bolted to the fuselage between stations 195 and 363. Ice protection strips reinforce the skin in the plane of propeller rotation. The cockpit is located forward of station 47 and is entered through a door from the main cabin. A service door located in the left side of the fuselage, between stations 47 and 66, has no handle on the outside; it may be opened from within and used as an auxiliary exit if the aircraft is on the ground and the engines are not running. An emergency exit is located overhead, midway between the pilot's and co-pilot's seats, and is fitted with two red-painted release handles at its forward end. In the main cabin, there are nine windows on the right side and eight windows on the left side. The emergency exits are located as follows: two exits on the left side between stations 156 and 177 and stations 351 and 372, and two exits on the right side between stations 312 and 333 and stations 351 and 372. The frames of the emergency exits are painted orange-yellow. The release handles, located under the windows, are marked AUXILIARY EXIT.

2-235. On aircraft 1 through 96, 30 collapsible high-density personnel seats are installed. Canvas troop seats, which are removed and carried as loose equipment when the high-density personnel seats are installed, are also provided. The canvas troop seats can be folded for storage and need not be removed for cargo loading. The floor of the main cabin is covered with wood veneer. It is level with the reference plane of the aircraft from the forward bulkhead at station 47 aft to the main cargo door. From the main cargo door, the floor rises at an angle of 7.5 degrees to the aft bulkhead at station 450. The slope becomes horizontal and serves as a loading platform when the aircraft is on the ground. The main loading door is 7 feet ½ inch wide and 5 feet 10½ inches high. It is located on the left side of the fuselage between stations 450 and 535, and has a removable panel for emergency escape or for the use of paratroopers. The lavatory compartment is located aft of the bulkhead at station 538. A tail compartment aft of the lavatory is accessible through a door in the bulkhead at station 583.

2-236. In aircraft A through D, a vestibule area is aft of the bulkhead at station 413, with a lavatory on the right side from stations 413 to 465 and a galley on the left from stations 413 to 450. On the right side, between stations 486 and 538, is a coatroom. Aft of the bulkhead at station 538 are the crew's sleeping quarters. An access door on the bulkhead at station 623 leads to the tail compartment. The main cabin door, which incorporates built-in steps, is located between stations 450 and 481.

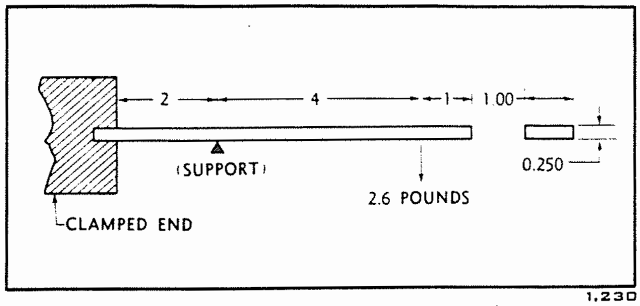

Figure 2-30. Plexiglas Stress-Crazing Test

2-237. CLEANING FUSELAGE EXTERIOR. For information on cleaning the fuselage exterior, see paragraph 2-3.

2-238. CLEANING PLEXIGLAS WINDOWS. Numerous products are available for cleaning Plexiglas and Lucite; but, before a product is used, it should be tested for stress crazing to make certain that it will not dissolve the acrylate, as follows:

a. Set up a strip of the plastic as a cantilever beam, with dimensions as shown in figure 2-30.

b. Apply a load of 2.6 pounds for a period of 10 minutes.

c. Apply the cleaner, in accordance with manufacturer's instructions, to the top side of the plastic specimen that is under tension, at a point two inches from the clamped end and directly above the support. Allow the solution to remain for a period of five minutes.

d. Examine the plastic for small surface cracks or crazing while the specimen is under test.

CAUTION

Plastic solvents, such as esters, ethers, ketones, alcohols, and aromatic or chlorinated hydrocarbons, must never be used as window cleaners.

2-239. To clean Plexiglas windows, flush the exterior surface with plenty of water, using the bare hand to locate and dislodge any dirt, salt, or mud. Then proceed as follows:

a. Wash the windows with soap and water, being sure that the water is free from dirt and possible abrasives. A soft sponge or chamois may be used in washing as a means of carrying soapy and/or plain water to the window but not for scrubbing or rubbing purposes. Go over the surface with the bare hand to detect any dirt, and remove it before it scratches the plastic surface.

b. Dry the windows, preferably with a clean, damp chamois. If soft clean cloth or soft tissues are used, do not rub the acrylic plastic after it is dry.

c. Remove oil and grease by rubbing lightly with a cloth wet with solvent (Federal Specification P-S-661), "Plexicleaner," or "Plesti-Glyst."

CAUTION

Do not use the following materials on acrylic plastics: acetone, benzine, carbon tetrachloride, anti-icer fluid, fire extinguisher fluids, kerosene, lacquer thinners, or window cleaning sprays. These materials will soften the surface and cause crazing.

d. Do not rub the acrylic plastic with a dry cloth, since this may scatch the surface and also may build up a charge of static electricity which attracts dust particles to the surface. If the surface does become charged, patting or gently blotting with a clean damp chamois will remove the charge as well as the dust.

2-240. REMOVING SCRATCHES FROM PLEXIGLAS WINDOWS. If, after dirt and grease are removed, small scratches are visible, they may be filled and their effects eliminated by application of a good grade of commercial wax, preferably of the same refractive index as the Plexiglas. This should be applied in a thin, even coat and brought to a high polish by rubbing lightly with a soft dry cloth.

2-241. REMOVING SCRATCHES FROM PLEXIGLAS WINDOWS BY SANDING. Deep scratches may be removed from Plexiglas windows by means of a portable buffing wheel with a carbon pad or lamb's wool buffer loaded with du Pont Rubbing Compound, conforming to Specification VZ-2006, followed by polishing with wax. An alternate method is hand rubbing with No. 320 or No. 400 wet or dry abrasive paper, followed by a coat of wax.

CAUTION

Do not attempt hand polishing or buffing until the surface is clean. If dirt, grit, or sand are present during these operations they may cause more damage than the original scratches.

2-242. REMOVING SCRATCHES FROM PLEXIGLAS WINDOWS BY POLISHING AND BUFFING. Scratches too deep to be filled with wax, and not deep enough to require an extensive sanding operation, may be buffed out with a cloth wheel or rubbed away by brisk application of a polishing cloth. Observe the precautions outlined in paragraph 2-239, and never remove more than approximately 20 per cent of the thickness of the Plexiglas by buffing or sanding. For coarse buffing, use a buffing compound conforming to Specification VZ-2006. For finish buffing, use a soft, open-type cotton or flannel disc with buffing compounds conforming to Specifications ET-205 or D-14. Do not exceed speeds of approximately 630 rpm for a 12-inch wheel, 850 rpm for a 9-inch wheel, and 1260 rpm for a 6-inch wheel. After buffing, finish with Wilco Scratch Remover, Specification 35, followed by an application of wax.

2-243. CLEANING FUSELAGE INTERIOR. In most respects, the instructions for aircraft maintenance outlined in this manual apply to both the cargo transport aircraft, aircraft 1 through 96, and the staff transport aircraft, aircraft A through D. However, the interior cleaning and maintenance differs between the cargo transport and the staff transport due to their different interior arrangements, as noted in the following paragraphs.

2-244. GENERAL CLEANING OF FUSELAGE INTERIOR (AIRCRAFT 1 THROUGH 96). For general light cleaning of the interior of the cargo transport aircraft, aircraft 1 through 96, use warm soapy water. For heavier cleaning, use Turco L567, applied full strength, and rub the surface vigorously with a bristle brush. Allow the surface to stand until the dirt is loosened, and then remove it with a brush or cloth.

2-245. CLEANING INTERIOR FABRICS. Vinyl fabrics, in aircraft A through D, should not be exposed to strong solvents. Use Stoddard solvent to clean the fabrics as follows:

a. Apply the solvent with a sponge or soft clean cloth. Use a blotting action to remove small spots, because the dirt is dissolved by the action of the solvent.

b. Use the solvent sparingly to avoid wetting the fabric and forming rings.

c. If rings form, remove them by lightly cleaning a larger area around the ring.

2-246. CLEANING PAINT, ENAMEL, AND LEATHER.

a. To remove finger marks, smudges, grime, and lipstick, dilute one part of Turco Vitol in 5 to 10 parts of water. Apply this solution with a sponge or soft cloth. There is no need to scrub or apply pressure, as the dirt is dissolved by the action of the solution. Then wipe the surface dry.

b. To remove tar, asphalt, and chewing gum, remove as much as possible with a knife or scraper, being careful not to cut or scratch the surface of the material. Then use undiluted Turco Vitol to remove any remaining tar, asphalt, or chewing gum. Wipe the surface dry after application of the solution. This has a buffing effect that prevents the solution from leaving a stain.

2-247. CLEANING RUGS AND UPHOLSTERY. To clean the rugs and upholstery, on aircraft A through D, remove the dirt and grease as follows:

a. Vacuum-clean the rugs and upholstery.

b. Place a tablespoonful of rug shampoo in a bucket, and direct a jet of water into it to produce abundant foam.

c. Apply the foam uniformly to the surface to be cleaned. Remove the suds by wiping with a brush.

2-248. DEODORIZING. It is recommended that odors that may emanate from material carried in cargo planes be removed by using a deodorant as follows:

a. Spray the aircraft interior with full-strength "Turco-Form" spray deodorant about two hours before flight.

b. Ventilate the aircraft thoroughly just before putting it into service.

2-249. CEMENTING FABRICS AND MATERIALS. If vinyl-coated fabric, felt, flannel, leather, or synthetic rubber-sealing strips are excessively worn or damaged, replace them as follows:

a. Spread a suitable cement on the faying surfaces. For lightweight fabric, Douglas No. 105 cement is recommended; for medium weight fabric, use Minnesota Mining and Mfg. Co. EC-833 cement. For heavy fabric and for cementing areas not visible, use Douglas 106 cement. Cement uncoated fabrics with Minnesota Mining and Mfg. Co. EC-147. Apply the cement to both faying surfaces and allow to dry until tacky (2 to 10 minutes) before joining.

b. Allow the following drying periods for cements: Douglas 105, 10 minutes; Minnesota Mining EC-833, 10 to 30 minutes; Douglas No. 106, 2 to 10 minutes.

c. If using Douglas No. 105 cement, apply a second coat and allow it to get tacky.

d. Join the faying surfaces by hand pressure or a roller.

CAUTION

Vinyl-coated fabrics are very sensitive to many adhesives, and color changes may follow application within days or over a period of months.

2-250. CEMENTING LEATHER. Leather (not oil-tanned) may be cemented as follows:

a. First clean any adjacent aluminum surfaces with a clean cloth dampened with Stoddard solvent.

b. Apply a brush coat of Minnesota Mining EC-833 cement to both faying surfaces.

c. Allow the cement to dry until tacky.

d. Join the faying surfaces with hand pressure or roller.

2-251. CEMENTING RUBBER. Cement either natural or synthetic rubber as follows:

a. Roughen the rubber faying surfaces with a medium-grit sandpaper until all the talc or parting agent is removed.

b. When cementing rubber to phenolic plastics, roughen the phenolic surface with a medium-grit sandpaper until the surface gloss is removed.

c. Clean all surfaces except leather and wood with a clean cloth dampened in Stoddard solvent.

d. Apply a brush coat of Douglas No. 101 primer cement to all surfaces except leather.

e. Allow the cement to dry for one hour.

f. Apply a brush coat of duPont Fairprene No. 4 cement to both faying surfaces.

g. Allow the cement to dry for another hour.

h. Join the faying surfaces with hand pressure.

2-252. FUSELAGE DOORS. The main cabin door on the cargo transport aircraft, aircraft 1 through 96, is a double structure installed between stations 450.5 and 535. The forward of the two doors has a detachable panel that extends to the floor. This panel may be removed for the exit of paratroopers. The forward half of the door is also equipped with an emergency release lever and may be jettisoned overboard with its detachable panel when the emergency lever is pulled. The main cabin door on the staff transport aircraft, aircraft A through D, is a single door which opens from the top and incorporates integral steps.

2-253. An inside door at station 97 leads into the pilot's compartment from the main cabin. Another door opens through the aft main cabin bulkhead into the lavatory. An outside door, located on the left side of the fuselage between stations 47 and 70, is designated as the service door.

2-254. MAIN CABIN DOOR (AIRCRAFT 1 THROUGH 96). *(See figure 2-31.)* The main cabin door is mounted on two pairs of hinges, and each half of the door opens outward. The paratrooper's panel, located in the forward half of the door, is held in place by two retaining lugs below the panel and two latches near the top and on either side of the panel. The hinge pins of the forward door are connected by rods and linkage to an emergency lever. Operation of the emergency lever removes the hinge pins, and the door may then be jettisoned. The hinges of both doors are attached to the fuselage with four screws on each half of the hinge. Sealing strips, made of hollow bulb strip synthetic rubber, are fitted around both panels at the top, bottom, and outer edges. The inner or hinge edges of both panels are sealed with airfoam rubber.

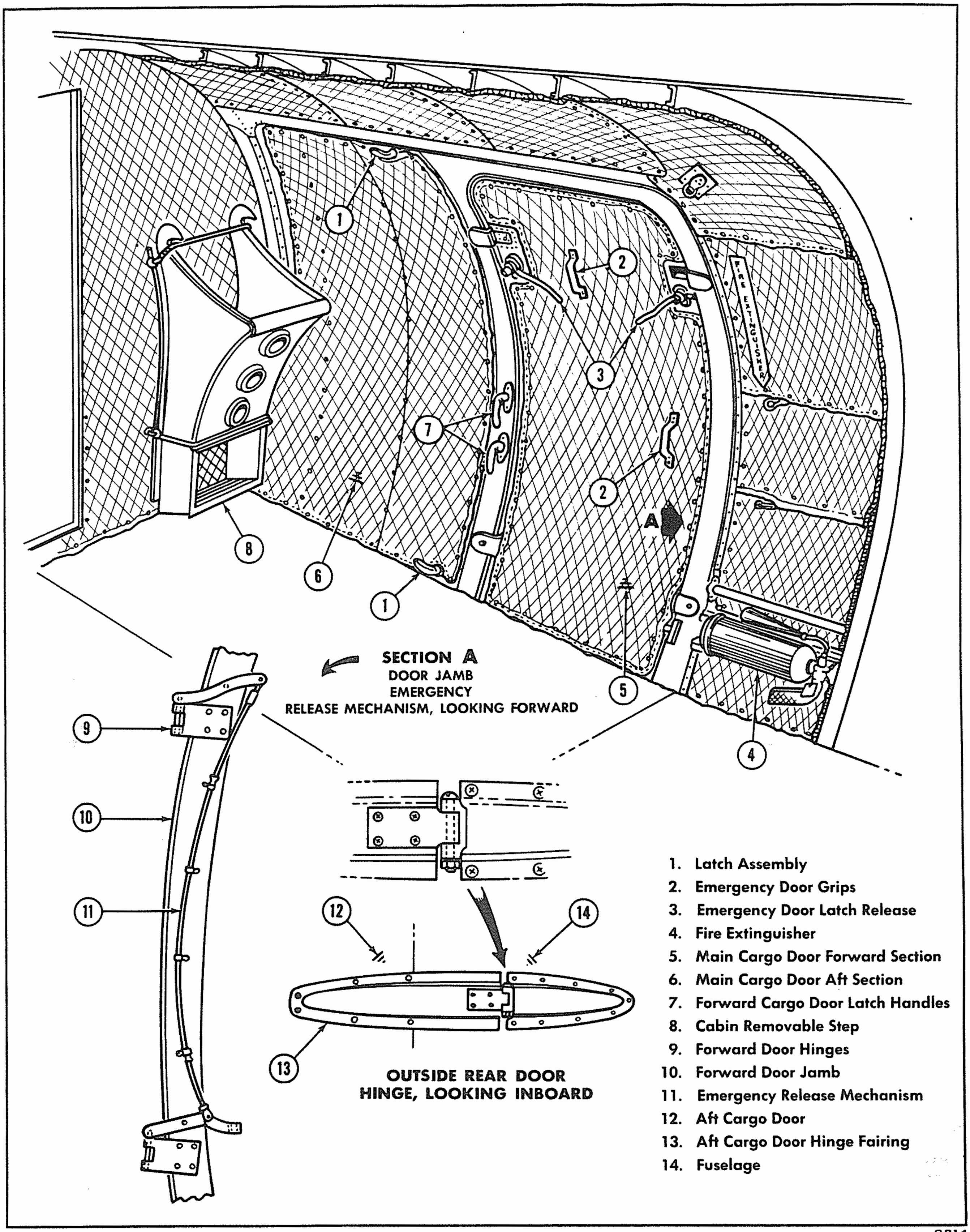

Figure 2-31. Cargo Door with Escape Panel

2-255. REMOVAL OF MAIN CABIN DOOR (AIRCRAFT 1 THROUGH 96). To remove the main cabin door, proceed as follows:

a. Open the aft section wide, until the stud on the hinge pins is aligned with the slot on the retaining collar. Pull the door up and free of the hinges.

b. The forward section of the door may be removed by breaking the safety wire and actuating the emergency lever, thus disengaging the hinge pins. Remove the panel. Either section may be removed by detaching the screws that fasten the hinges to the fuselage.

2-256. MINOR REPAIR AND REPLACEMENT OF MAIN CABIN DOOR (AIRCRAFT 1 THROUGH 96). To repair small dents in the door skin, hammer them out with a plastic hammer. Broken latch fittings must be replaced with new parts.

2-257. INSTALLATION OF MAIN CABIN DOOR (AIRCRAFT 1 THROUGH 96). Reverse the removal procedure, taking care to reset the emergency escape lever on the forward panel and to install the safety wire.

2-258. MAIN CABIN DOOR (AIRCRAFT A THROUGH D). The main cabin door on aircraft A and B is a single door hinged at the bottom, containing integral steps. When the door is opened from the top and downwards, it becomes the companionway for both entry and exit. A chain handrail extending from the door jamb, to the door supports, and to the bottom step, gives passenger support and door rigidity. On aircraft C and D, the door is built into the forward cargo door and is similar but slightly larger *(see figure 2-32)*. A safety latch, consisting of a chain and a lock-plate, will hold the door all but completely closed in case the door latching mechanism becomes disengaged during flight operations.

2-259. REMOVAL OF MAIN CABIN DOOR (AIRCRAFT A THROUGH D).

a. Remove the guard chains.

b. Support the door with shoring.

c. Remove the hinge attaching screws and remove the door.

2-260. MINOR REPAIR AND REPLACEMENT OF MAIN CABIN DOOR (AIRCRAFT A THROUGH D). To make repairs to the door exterior or the door latching mechanism, the steps must be removed from the door frame.

2-261. INSTALLATION OF MAIN CABIN DOOR (AIRCRAFT A THROUGH D). Reverse the removal procedure.

2-262. DOOR WARNING SWITCH (AIRCRAFT A THROUGH D). A door warning switch is located on the forward door jamb and is actuated by the door latching mechanism.

2-263. REPAIR AND ADJUSTMENT OF DOOR WARNING SWITCH (AIRCRAFT A THROUGH D). Repair and adjust the door warning switch by removing the four screws and back-up nuts, and adjusting the jamb nut and pushrod so that the warning light is on, except when the door is completely closed and locked.

2-264. INTERIOR FUSELAGE DOORS (AIRCRAFT 1 THROUGH 96). Three inside bulkhead doors provide passage between the compartments of the cargo transport aircraft. These doors are located as follows: one door is located in the forward bulkhead between the main cabin and the crew compartment; one is in the aft bulkhead between the main cabin and the lavatory; and one opens into the tail compartment from the lavatory.

2-265. REMOVAL OF INTERIOR FUSELAGE DOORS (AIRCRAFT 1 THROUGH 96).

a. To remove the forward bulkhead lavatory door, remove the cotter pins from the hinge pins and detach the pins; remove the door.

b. To remove the forward cabin bulkhead door and the tail compartment door, remove the piano wire pins from the hinges and remove the door.

2-266. MINOR REPAIR AND REPLACEMENT OF INTERIOR FUSELAGE DOORS (AIRCRAFT 1 THROUGH 96). To repair small dents in the door skin, hammer them out with a plastic hammer. Broken latch fittings must be replaced with new parts.

2-267. INSTALLATION OF INTERIOR FUSELAGE DOOR (AIRCRAFT 1 THROUGH 96). Reverse the removal procedure.

2-268. INTERIOR FUSELAGE DOORS (AIRCRAFT A THROUGH D). The cabin aft bulkhead sliding doors at station 413 lead into the vestibule. Two doors opening from the vestibule give access to the lavatory and the crew's sleeping quarters. A door located in the aft bulkhead of the crew's sleeping compartment permits access to the tail compartment for maintenance and inspection. For information on the removal, minor repair and replacement, and the installation of the interior fuselage doors on aircraft A through D, see paragraphs 2-264 through 2-268, preceding.

2-269. SERVICE DOOR. The service door is located on the left side of the pilot's compartment just aft of the bulkhead behind the pilot's seat. This door must not be used when the engines are running, except for smoke evacuation, as it is near the plane of propeller rotation.

2-270. REMOVAL OF SERVICE DOOR. Remove the bolts and nuts from both hinges, and remove the door.

2-271. MINOR REPAIR AND REPLACEMENT OF SERVICE DOOR. To repair small dents in the door skin, hammer them out with a plastic hammer. Broken latch fittings must be replaced with new parts.

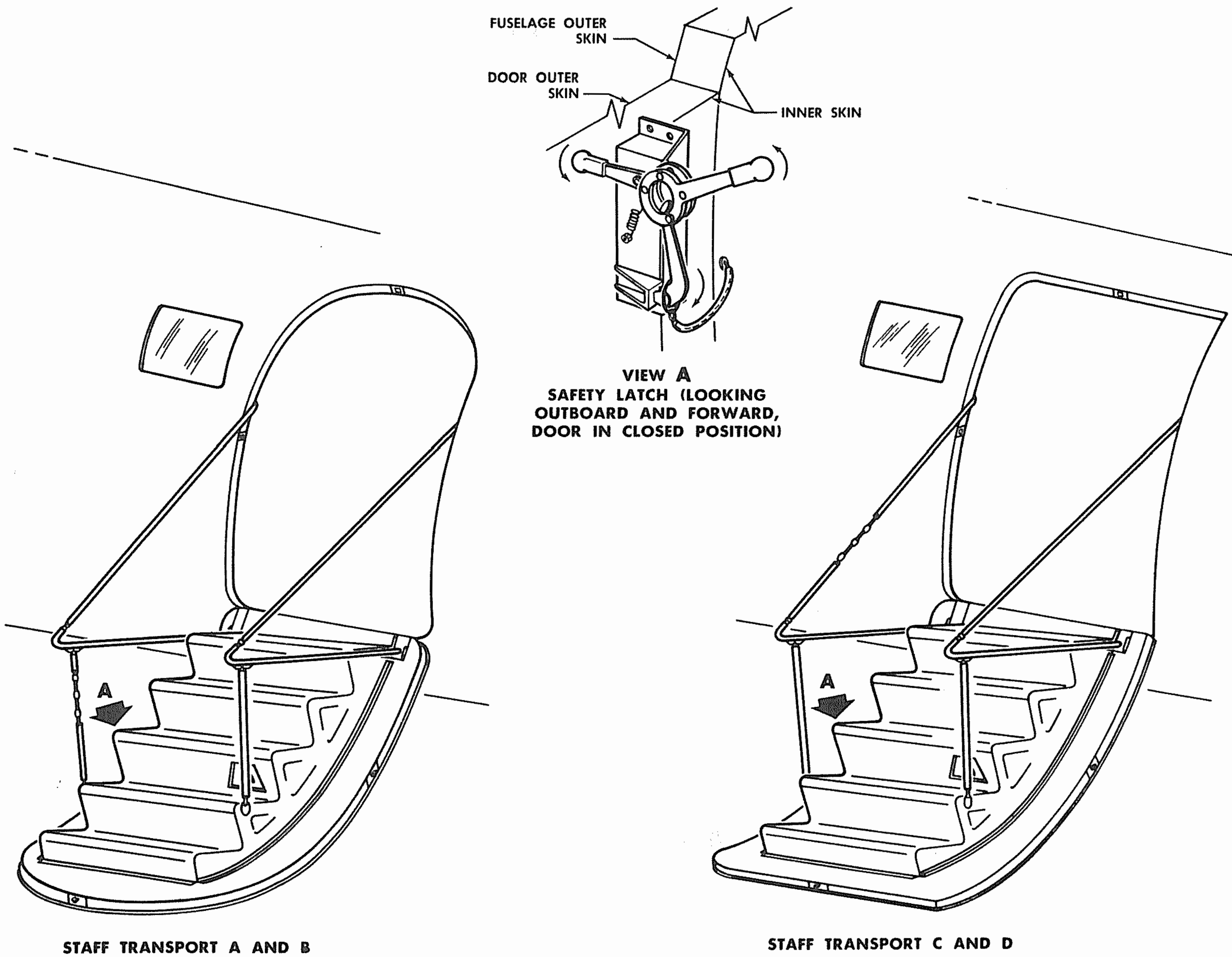

Figure 2-32. Main Entrance Door and Safety Catch (Aircraft A through D)

2-272. INSTALLATION OF SERVICE DOOR. Reverse the removal procedure.

2-273. WINDOWS.

2-274. MAIN CABIN WINDOWS (AIRCRAFT 1 THROUGH 96). *(See figure 2-33.)* Seventeen main cabin windows are each attached to the skin of the aircraft with 20 screws secured by Tinnerman nuts. A neoprene sealing ring is installed between the 3/16-inch plastic window pane and the aircraft skin, and is cemented to the skin with a fillet of aluminized EC-801 sealant.

2-275. REMOVAL OF MAIN CABIN WINDOWS (AIRCRAFT 1 THROUGH 96).

a. Working from the outside, detach the 20 screws.

b. Remove the neoprene sealing ring and the window pane as a unit.

2-276. INSTALLATION OF MAIN CABIN WINDOWS (AIRCRAFT 1 THROUGH 96). Reverse the removal procedure, adding a fillet of aluminized EC-801 sealant as necessary.

2-277. MAIN CABIN WINDOWS (AIRCRAFT A THROUGH D). *(See figure 2-33.)* Seventeen windows are installed in the main cabin area, 4 in the auxiliary exit doors and 13 in fixed frames. These are double-panel acoustic windows, held in place by means of synthetic rubber sealing rings and metal retainers. The inner pane is made from 1/4-inch plastic, and the outer pane is made from 3/8-inch plastic.

2-278. REMOVAL OF FIXED MAIN CABIN WINDOWS (AIRCRAFT A THROUGH D).

a. Remove the screws which attach the window frame inside the aircraft.

b. Remove the screws which attach the inner pane retainers to the fuselage skin outside the aircraft.

c. Remove the retainer strips, sealing rings, and the inner pane from the window frame inside the aircraft.

d. Remove the bolts and nuts which attach the outer pane and seal strips to the fuselage skin, and remove the pane.

2-279. INSTALLATION OF FIXED MAIN CABIN WINDOWS (AIRCRAFT A THROUGH D). Reverse the removal procedure.

2-280. RADIO OPERATOR'S WINDOW. A plastic window pane, 3/8 inch thick, is installed for the radio operator on the right side of the aircraft between stations 64 and 78. The lower edge of the window is four inches above the center line of the main cabin windows. The radio operator's window is held in place with a 0.072-inch aluminum retainer ring, a neoprene sealing ring, and screws in the same manner that the main cabin windows are installed (see paragraph 2-274). For removal and installation of the radio operator's window, see paragraphs 2-275 and 2-276.

2-281. LAVATORY WINDOW. The lavatory window, located on the right side of the fuselage between stations 563 and 578, is held in place with 12 screws. For removal and installation, see paragraphs 2-275 and 2-276.

2-282. EMERGENCY EXITS. The emergency exits are fitted with an escape mechanism that permits opening by the actuation of a lever beneath the window. The window pane is removable by the removal of screws and a retainer rim. Maintenance should include periodic tests to insure that the latch works smoothly and easily.

2-283. PILOT'S AND CO-PILOT'S SEATS. *(See figure 2-34.)* Bucket-type seats of magnesium, braced with aluminum-alloy, are provided for the pilot and co-pilot. The framework is attached to brackets which are bolted to the floor. The seats are inclined at a 13-degree angle. To accommodate variance in the pilots' stature, the seats are adjustable by means of two vertical tubes and two horizontal tubes. Levers located at the left of the pilot's seat and at the right of the co-pilot operate the vertical adjusting mechanisms. Short cables at the lower right side of the pilot and the lower left side of the co-pilot operate the horizontal adjusting mechanisms. Bungees behind the seats raise them when the levers release the arms over the vertical tubes. Over each inboard horizontal tube is a support to adjust the tube rollers and stabilize the seats. Each set is fitted with cushions and cushioned arm rests, and the inboard arm rests can be swung back. Standard safety belts and harnesses (NAF-1201) and inertia reels (0-3903) are provided.

2-284. REMOVAL OF PILOT'S AND CO-PILOT'S SEATS.

a. To remove the seats from the floor, remove the bolts from the rollers on the horizontal tubes.

b. To remove the buckets from the frame, remove 16 bolts, four from each of the four arms supporting the vertical tubes.

2-285. MINOR REPAIR AND REPLACEMENT OF PILOT'S AND CO-PILOT'S SEATS. For repairs in the leather seat covering, see paragraphs 2-246 and 2-250.

2-286. INSTALLATION OF PILOT'S AND CO-PILOT'S SEATS. Reverse the removal procedure.

2-287. RADIO RACK. The radio rack is located between stations 117 and 136 for the various radio items and components.

2-288. RADIO OPERATOR'S STATION. The radio compartment is located on the right side of the crew compartment, just forward of the bulkhead at station 97. A swivel chair, attached to the floor by four bolts, one in each leg, is provided adjacent to the radio operator's table. The radio operator's table is bolted to the bulkhead at station 97. A window is installed adjacent to the radio operator's station.

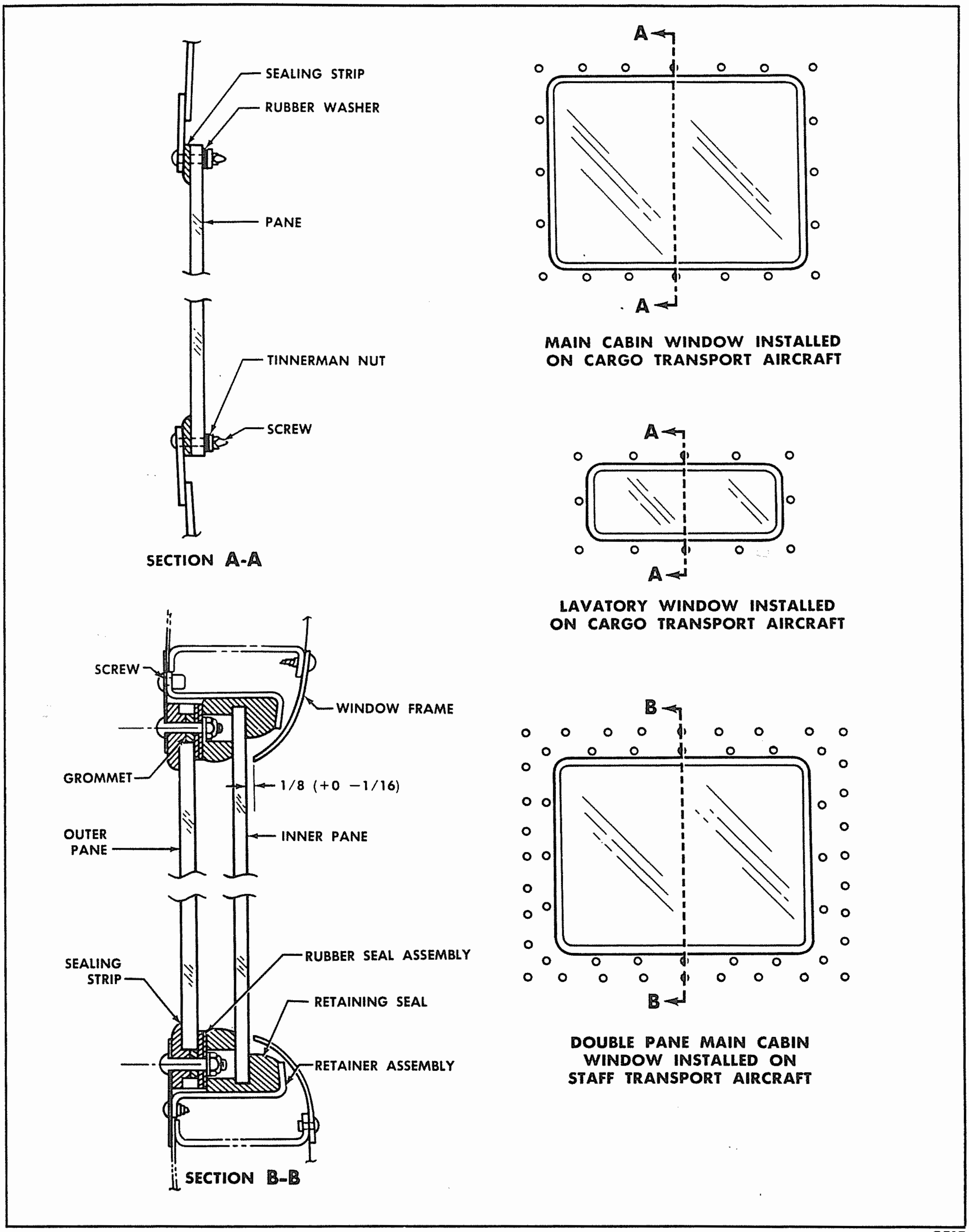

Figure 2-33. Installation of Plexiglas in Windows

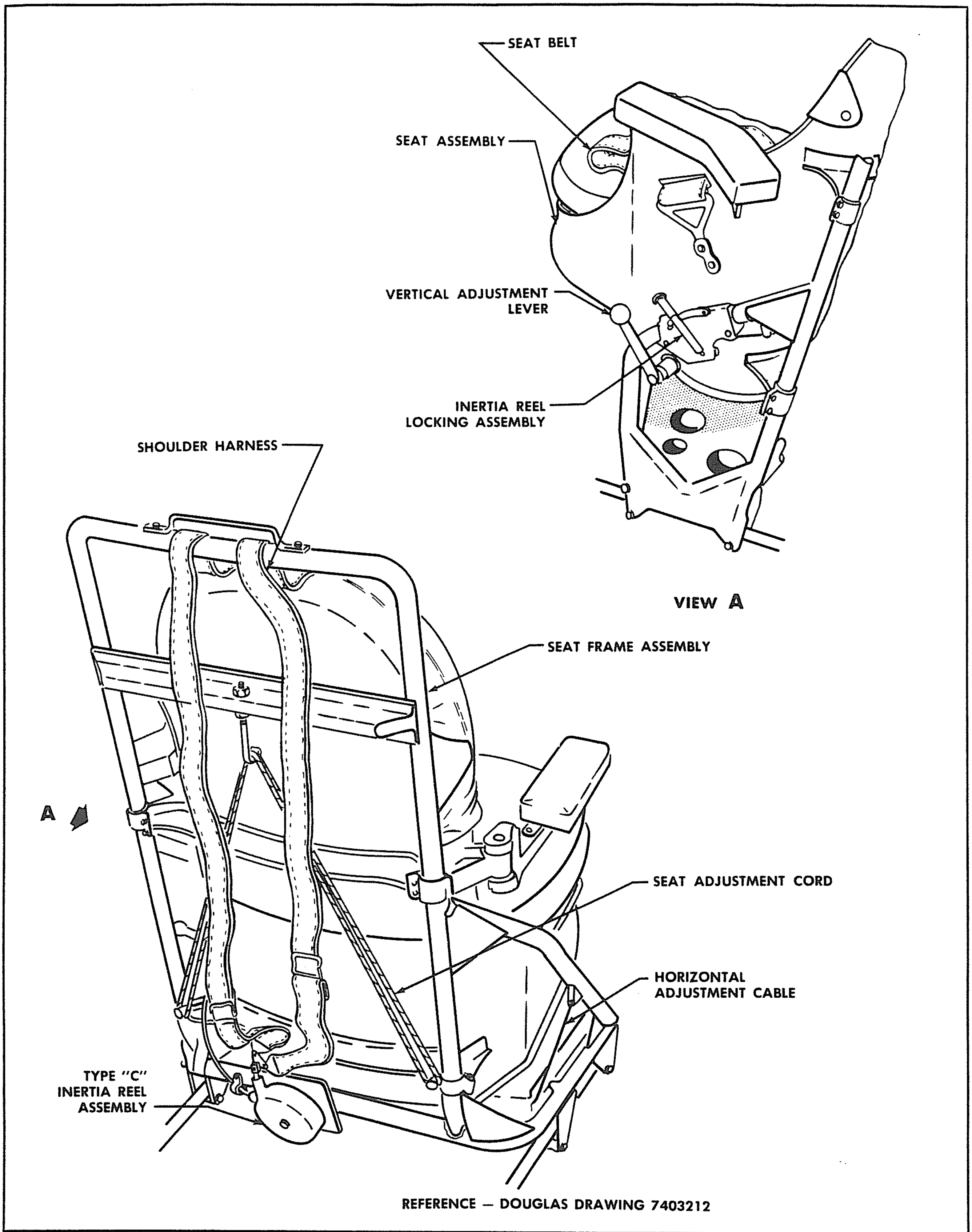

Figure 2-34. Cockpit Seats

2-289. NAVIGATOR'S STATION. The navigator's station is on the left side of the aircraft between the radio rack and the bulkhead at station 47. A removable navigator's table is installed between the radio junction box and the bulkhead at station 47. The table is hinged in order to provide access to the driftmeter. The bulkhead at station 47 is cut away to provide an adequate view through the windshield. The navigator's stool is hinged to the floor and can be folded beneath his table to allow free passage to the cockpit. On the center line of the aircraft ceiling, at station 63, is a flush circular periscopic sextant window. The rim of the periscopic sextant window is marked in increments of five degrees of the circle for the purpose of using the periscopic sextant. Adequate stowage is provided for miscellaneous navigator's equipment.

2-290. WINDSHIELD. *(See figure 2-35.)* The windshield consists of two halves, one on the right and one on the left. Each half consists of two panes: the outer pane is a sheet of clear glass $3/16$ inch thick, and the inner pane is a non-shatterable, three-ply lamination of glass and plastic $1/2$ inch thick. The inner and outer panes are $5/32$ inch apart, and through this space is circulated a current of hot air that de-ices the windshield. Each of the four panes may be removed separately. The outer ones must be removed from the outside by unscrewing a retainer ring; the inner ones may be removed from the inside by withdrawing the upper and lower pairs of hinge pins.

2-291. CLEANING AND MAINTENANCE OF WINDSHIELD. The windshield should be washed when the other aircraft windows are cleaned. To reach the enclosed surfaces of the inner and outer pairs of panes, swing the inner panes down on their lower hinges (see paragraph 2-292, step c). A small aluminum chain is rigged on each side of each inner pane to hold the panes in place at an angle convenient for washing and cleaning.

2-292. REMOVAL OF INNER PANES OF WINDSHIELD.

a. Remove the compass by disconnecting the three-point suspension bungees from hooks at the lower corners of the windshield and above the windshield on the center line of the aircraft.

b. Detach the supporting chains from both sides of the inner panes.

c. Pressing against the top of the inner windshield with the heel of the hand, withdraw the upper hinge pins from the hinges.

d. Repeat this process at the bottom of each pane to withdraw the lower hinges.

e. Remove the panes from the windshield.

2-293. INSTALLATION OF INNER PANES OF WINDSHIELD. Reverse the removal procedure.

2-294. REMOVAL OF OUTER PANES OF WINDSHIELD.

a. Detach the two windshield wiper arm-and-blade assemblies from the driving mechanism.

b. Remove the $3/8$-inch flush screws that hold the outer pane retainer ring in place.

c. Remove the outer retainer ring.

d. Remove the glass pane along with the rubber seal molding around the edges.

2-295. INSTALLATION OF OUTER PANES OF WINDSHIELD. Reverse the removal procedure. Apply water-proofing sealant before the outer retainer ring is installed.

2-296. LAVATORY (AIRCRAFT 1 THROUGH 96). *(See figure 2-36.)* The lavatory, aft of the main cabin, contains a toilet, a paper holder, a washbasin cabinet, a towel rack, and a urinal. For information on the lavatory on aircraft A through D, see paragraph 2-311.

2-297. TOILET (AIRCRAFT 1 THROUGH 96). *(See figure 2-36.)* The disposal-type toilet consists of two units, a seat and cover assembly, and an aluminum container. A large fiber tube vents the container through the right side of the fuselage. A lightweight metal urinal is provided. A rubber hose extends from the bottom of the urinal along the bulkhead and overboard through a venturi on the tail cone.

2-298. SERVICING TOILET (AIRCRAFT 1 THROUGH 96). The toilet should be removed and cleaned when necessary. Refill it with at least one quart of a recommended disinfectant.

2-299. REMOVAL OF TOILET BUCKET (AIRCRAFT 1 THROUGH 96). To remove the container from the bowl, unfasten the clips and lift the bowl. Unfasten the three trunk-type fasteners holding the container to the floor.

2-300. INSTALLATION OF TOILET BUCKET (AIRCRAFT 1 THROUGH 96). Reverse the removal procedure.

2-301. WASHBASIN (AIRCRAFT 1 THROUGH 96). The lightweight metal washbasin is attached to a fuselage stringer on the left by three screws and to the bulkhead in front by three screws. It has a recessed opening to prevent overflow, a drain nipple integral with the bowl, and a rubber stopper. On the lip of the bowl is a self-closing faucet. For drainage, a pipe is routed from the outlet of the basin to an outlet on the left side of the fuselage. A valve below the basin allows drainage of the tank through another pipe to the same overboard outlet. On the bulkhead above the basin is a paper towel holder.

2-302. REMOVAL OF WASH BASIN (AIRCRAFT 1 THROUGH 96). *(See figure 2-36.)* Remove the cabinet by removing the screws from the top of the basin, from the flange on the bulkhead, and from the wooden strip at floor level.

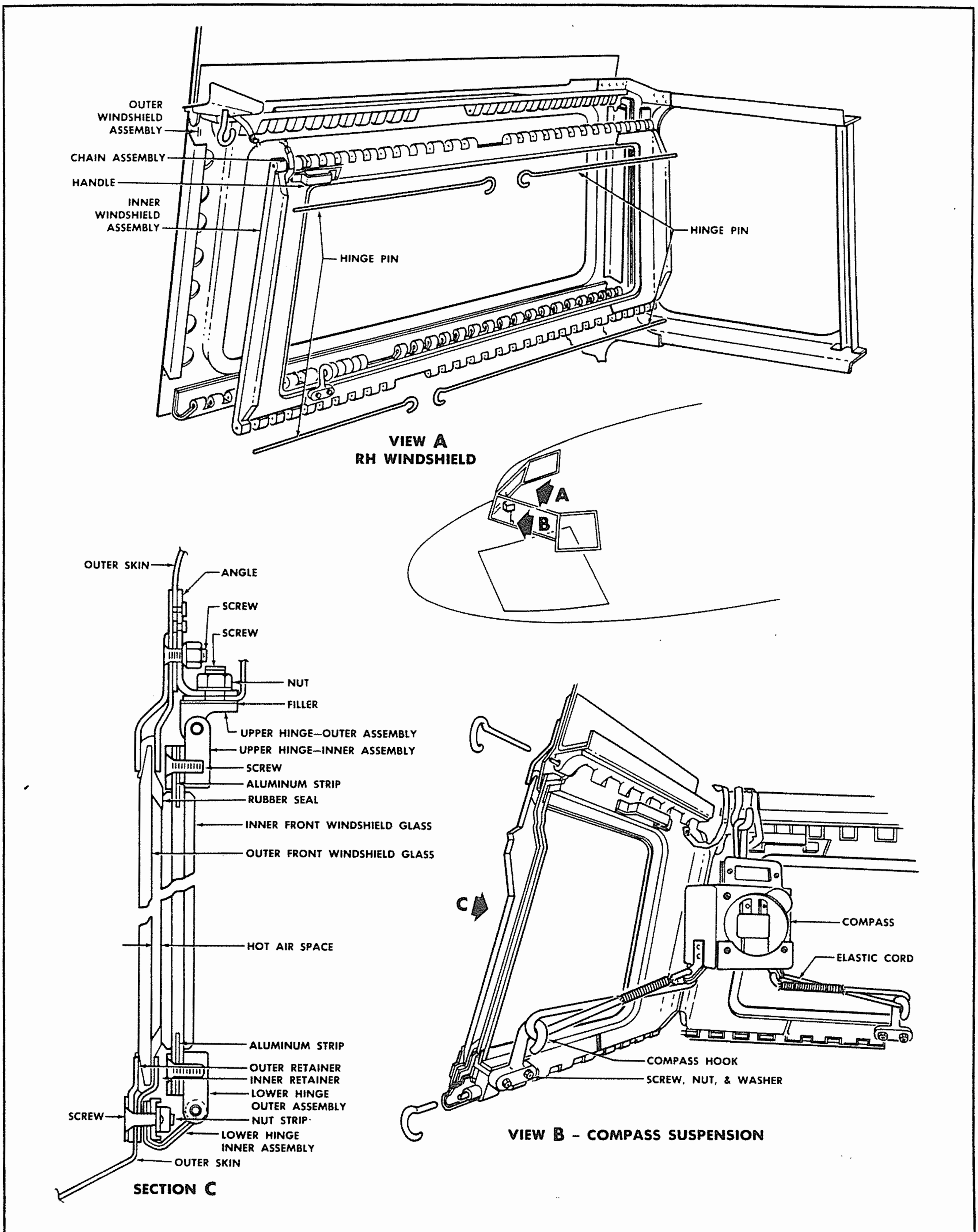

Figure 2-35. Windshield

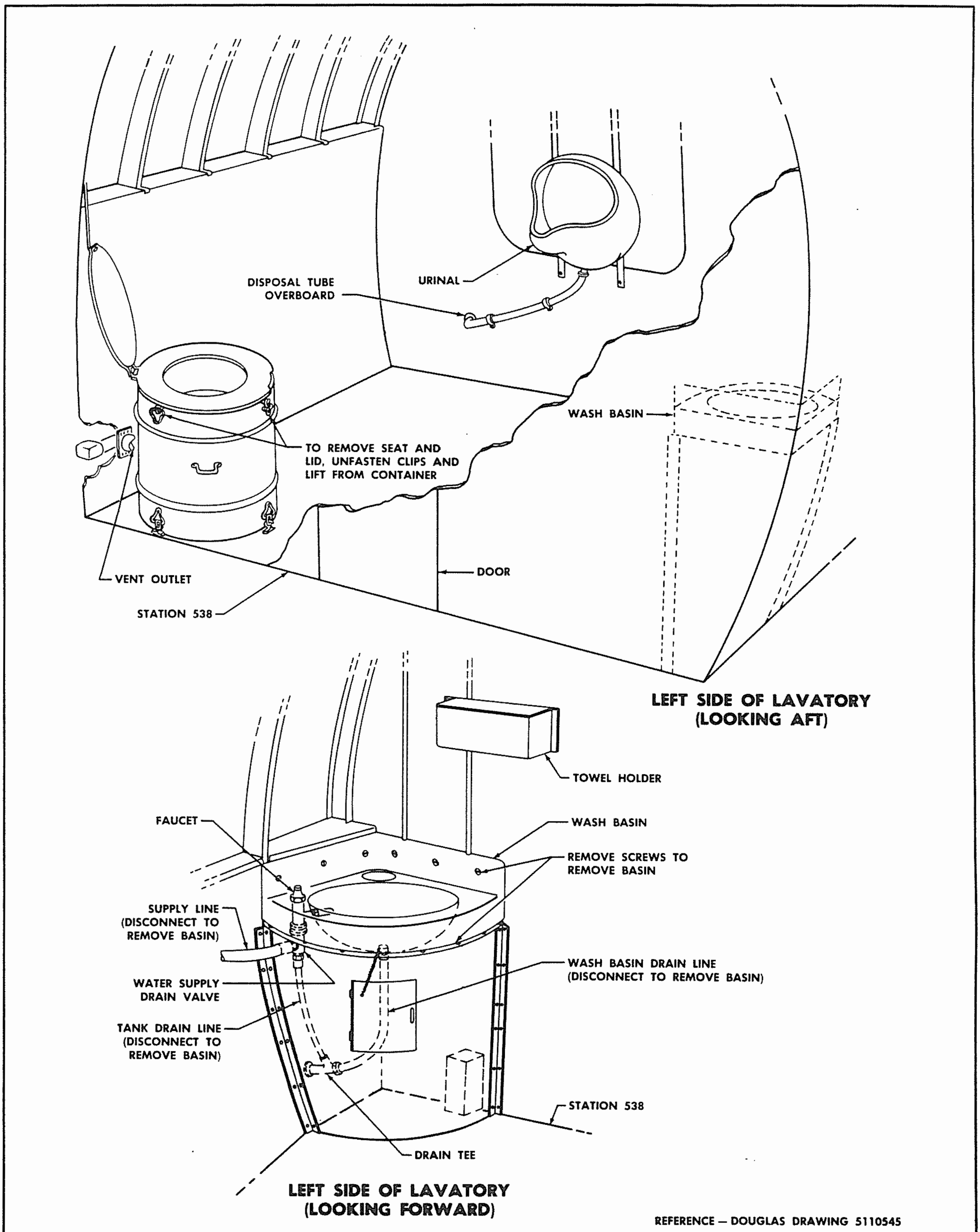

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Figure 2-36. Lavatory Equipment (Aircraft 1 through 96)

2-303. MINOR REPAIR OF WASHBASIN (AIRCRAFT 1 THROUGH 96). Sand and refinish the wash basin when necessary.

2-304. INSTALLATION OF WASHBASIN (AIRCRAFT 1 THROUGH 96). Reverse the removal procedure.

2-305. WASH-WATER TANK (AIRCRAFT 1 THROUGH 96). *(See figures 2-37 and 2-38.)* An aluminum water tank is supported by an aluminum-alloy cradle, bolted and riveted to the rear bulkhead. Two turnbuckles, attached to the arms of the cradle in front of the tank, keep the tank rigid. The cylindrical tank has a three-gallon capacity and is covered with kapok. The tank is provided with a filler neck and cap at the top and with an outlet at the bottom; the outlet is connected with a metal tube that extends along the left of the fuselage to a shutoff valve in the washbasin cabinet. The tank interior surface is covered with an asphaltum-base compound to prevent corrosion.

2-306. DRAINING WASH-WATER TANK (AIRCRAFT 1 THROUGH 96). Open the shutoff valve below the washbasin and allow the water to flow overboard.

2-307. FILLING WASH-WATER TANK (AIRCRAFT 1 THROUGH 96). The tank is replenished through the filler cap on top of the tank (see paragraphs 1-92 and 1-93).

2-308. REMOVAL OF WASH-WATER TANK (AIRCRAFT 1 THROUGH 96). To remove the tank from the cradle, release the bonding wires from the turnbuckles. Unscrew the turnbuckles and remove the pipe by loosening the nut from its connection. Slide the tank from the arms of the cradle and remove it.

2-309. MINOR REPAIR OF WASH-WATER TANK (AIRCRAFT 1 THROUGH 96). Leaks around fittings may be tightened with ordinary tools. Leaky tanks may be welded as a temporary repair, but should be replaced as soon as possible.

2-310. INSTALLATION OF WASH-WATER TANK (AIRCRAFT 1 THROUGH 96). Reverse the removal procedure.

2-311. LAVATORY (AIRCRAFT A THROUGH D). *(See figure 2-38.)* The lavatory is located aft of the bulkhead at station 413 on the right side. It contains a toilet, a paper holder, a washbasin, and a towel rack. For information on aircraft 1 through 96, see paragraph 2-296.

2-312. TOILET (AIRCRAFT A AND B). A Weber toilet tank is installed in the lavatory.

2-313. TOILET (AIRCRAFT C AND D). A Wickland bucket-type toilet is installed in the lavatory.

2-314. SERVICING TOILET (AIRCRAFT A THROUGH D). The toilet is serviced through a service panel on the bottom right side of the aircraft, at station 440. A hinged access door covers the service panel.

2-315. WASHBASIN (AIRCRAFT A THROUGH D). *(See figure 2-38.)* A washbasin is installed in the right aft corner of the lavatory compartment. Individual faucets supply hot or cold water to the basin, and a drain is provided through the bottom of the fuselage.

2-316. WASH-WATER TANKS (AIRCRAFT A THROUGH D). *(See figure 2-38.)* Two water tanks are installed to provide water to the lavatory washbasin and the buffet sink *(see figure 2-38)*. These tanks are installed between the walls of the partition separating the lavatory from the buffet area. One of the tanks is covered with insulation, and contains a heating element controlled by a switch on the buffet service panel.

2-317. REMOVAL OF WATER TANKS (AIRCRAFT A THROUGH D).

a. Remove the partition lining panel adjacent to the toilet.

b. Disconnect the water fill and drain connections from the tank; disconnect the electrical connections from the heating element in the hot water tank.

c. Remove the attaching clamps, and remove the tank from the aircraft.

CAUTION

Exercise care to avoid damage to the tank insulation.

2-318. INSTALLATION OF WASH-WATER TANKS (AIRCRAFT A THROUGH D). Reverse the removal procedure.

2-319. MAIN CABIN SEATING EQUIPMENT (AIRCRAFT 1 THROUGH 96). On cargo and troop transport aircraft, folding benches for 35 men are provided for installation along the sides of the main cabin. These structures are supported by troop bench attachment fittings secured to the side of the fuselage, and by fittings secured to the floor structure. On each aircraft, the following types of troop benches are installed: four 4-man troop benches; five 3-man troop benches, and two 2-man troop benches. In addition to the folding benches, collapsible high-density personnel seats are installed, seating 30 persons *(see figure 2-39)*. The seats are arranged along both sides of the cabin, facing forward, providing a center aisle. When these seats are installed, the folding troop benches and litter side wall fittings are removed and carried as loose equipment in the aircraft.

2-320. REMOVAL OF FOLDING BENCHES (AIRCRAFT 1 THROUGH 96). *(See figure 2-40.)* The canvas web benches for 35 troops or other personnel may be folded and stowed against the sides of the cargo compartment to make room for the installation of lit-

(Continued on Page 197)

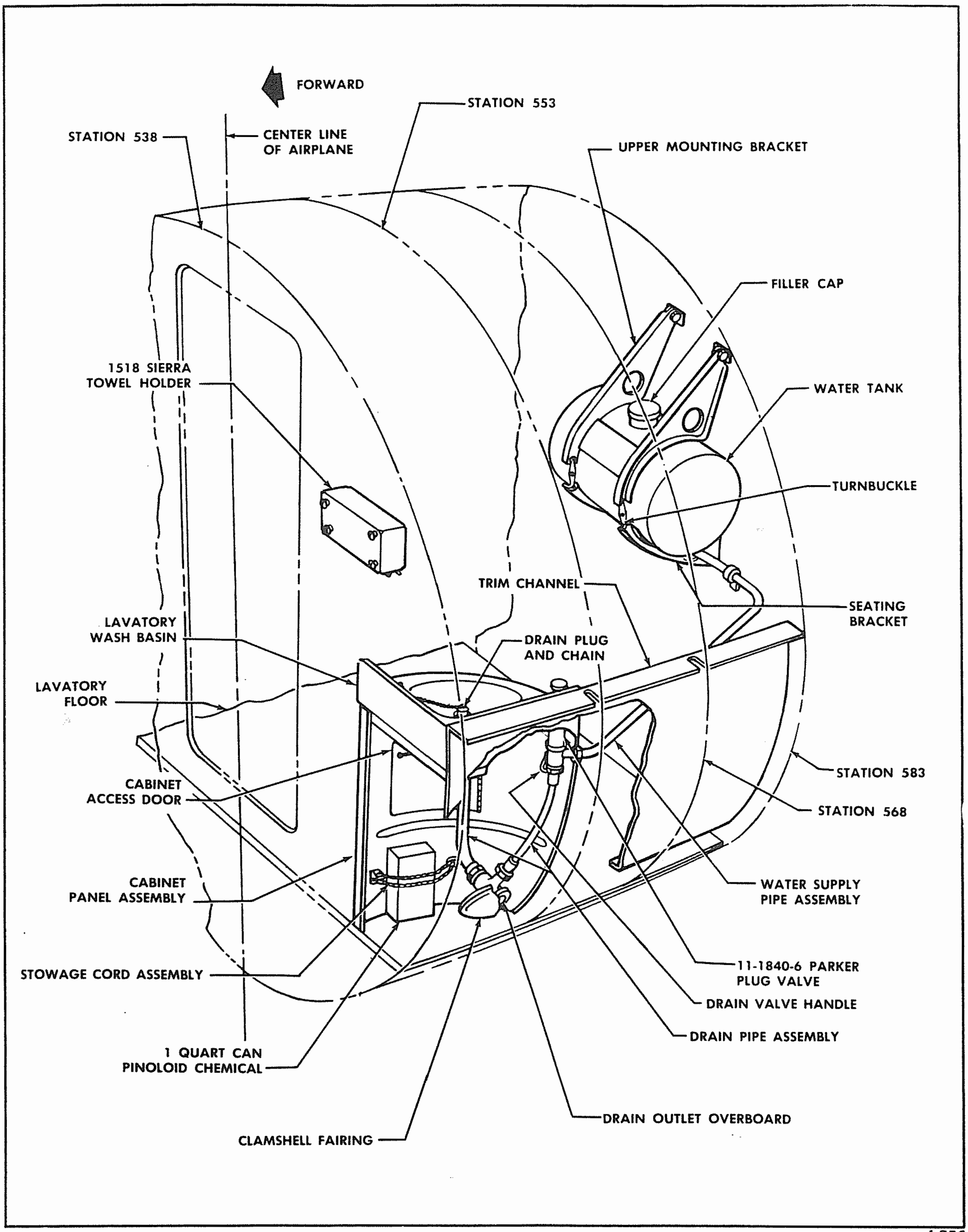


Figure 2-37. Wash-Water System (Aircraft 1 through 96)

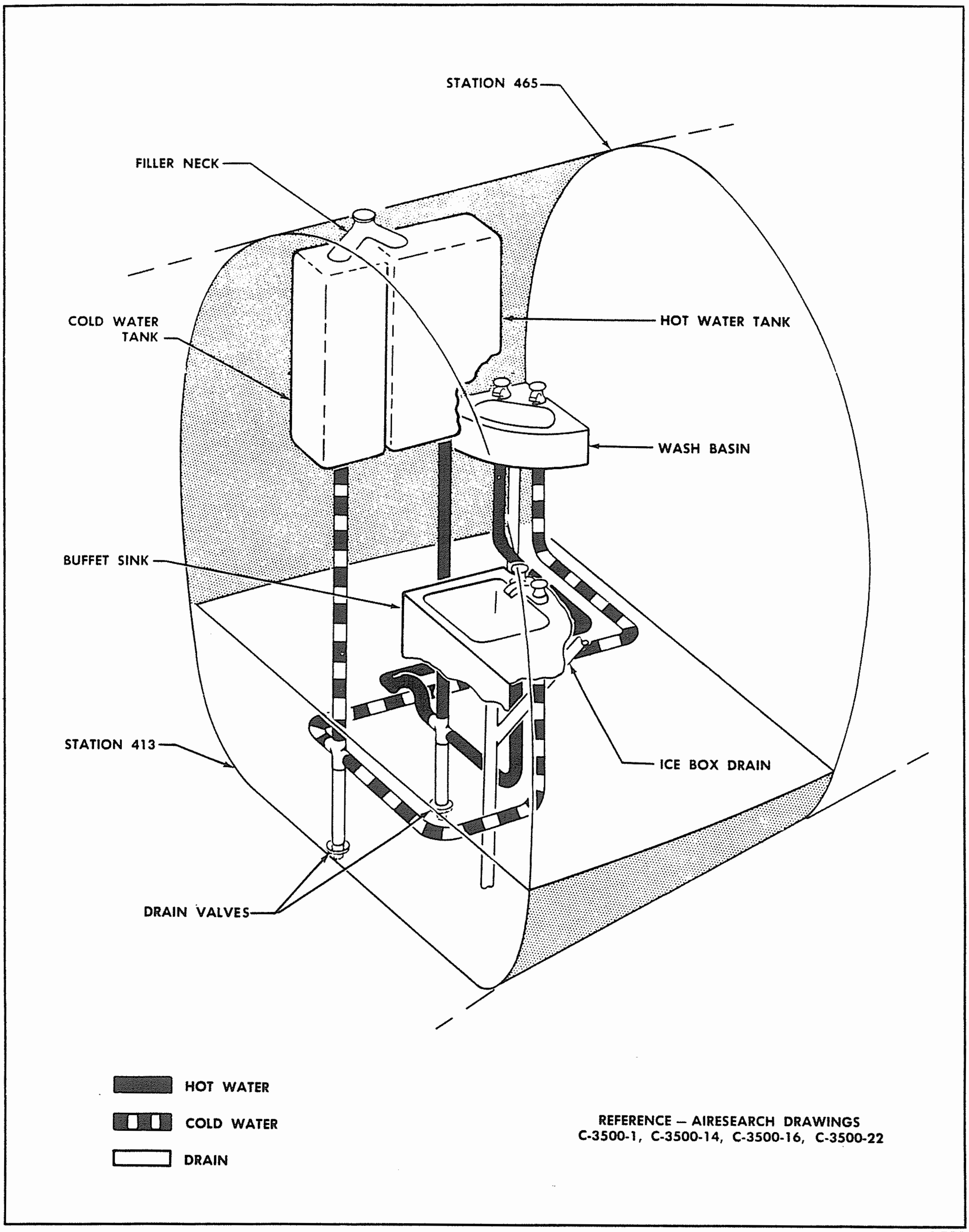


Figure 2-38. Wash-Water System (Aircraft A through D)

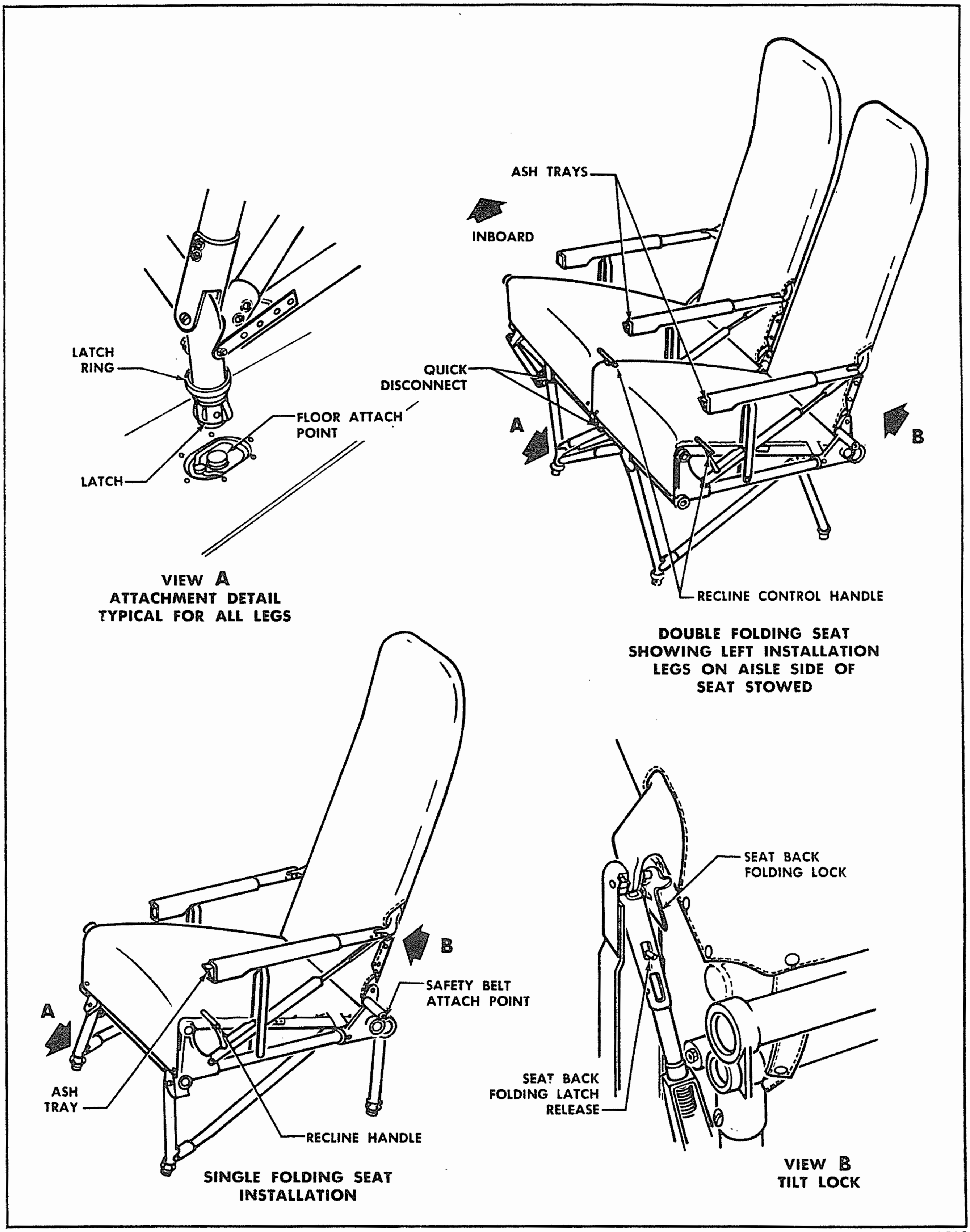


Figure 2-39. Collapsible Personnel Seats (Aircraft 1 through 96)

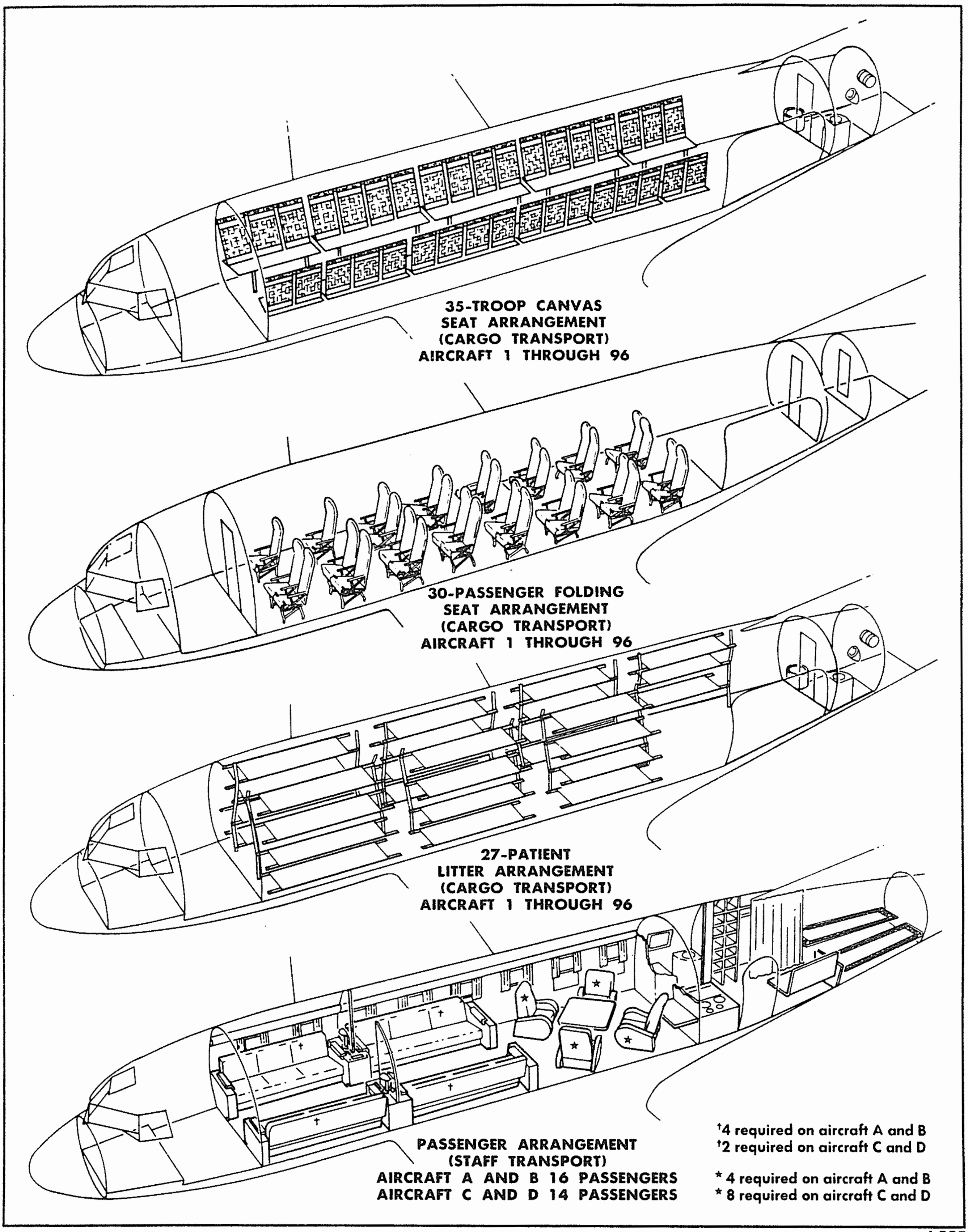


Figure 2-40. Interior Arrangements

(Continued from Page 192)

ters and or the loading of cargo. The folded benches provide protection against damage to the sides of the cabin. Tubes supporting these seats are bolted to the frame of the aircraft, as are the litter-supporting frames. To remove the canvas benches, actuate the quick-release fasteners on the bench legs at the floor by pressing the spring lugs. Unhook the canvas seat backs from the upper tubes. Remove the bolts attaching the 31 fittings that clamp the lower tube to the fuselage, and remove the tube and benches. The benches, then, may be removed from the tube if desired.

2-321. INSTALLATION OF FOLDING BENCHES (AIRCRAFT 1 THROUGH 96). Reverse the removal procedure.

2-322. REMOVAL OF COLLAPSIBLE PASSENGER SEATS (AIRCRAFT 1 THROUGH 96). *(See figure 2-39.)* To remove collapsible passenger seats from the aircraft, operate the fitting at each leg attach point by depressing the spring-loaded pins and pulling up on the ring; the attaching dogs will then expand permitting the leg to be removed from the floor fitting. Collapse the seat by folding the legs inward to the seat bottom, and fold the back forward and down to the seat bottom by operating the seat folding lock adjacent to the arm rest and seat back brace.

2-323. MINOR REPAIR AND REPLACEMENT OF COLLAPSIBLE PASSENGER SEATS (AIRCRAFT 1 THROUGH 96). The passenger seat should operate through five positions of reclining angles. A positive locking mechanism, consisting of a compression spring, an operating handle, and a locking pin, should be checked for wear at the latching points. The seat covering snaps on the frame, and may be removed for cleaning and repairing.

2-324. INSTALLATION OF COLLAPSIBLE PASSENGER SEATS (AIRCRAFT 1 THROUGH 96). Reverse the removal procedure.

2-325. MAIN CABIN SEATING EQUIPMENT (AIRCRAFT A AND B). *(See figure 2-40.)* The main cabin is fitted with leather, vinyl, and cloth upholstered seats for 16 passengers. Four of these seats are Hardman aircraft swivel executive type, arranged around a removable table with hinged flaps, in a section of the cabin between stations 289 and 413. In the forward end of the cabin are four divans, each providing space for three passengers. Each divan is 80 inches long and 33 inches wide, and can be utilized as a sleeping space. Seats are hinged on the outboard edge and may be raised to afford access to storage space beneath. Swivel chair bases are of top grain leather. Seats and divans are upholstered in nylon fabric. To level the divan seat cushion for sleeping, lift the inboard edge and pull the cushion inboard. The outboard edge will rise and the seat cushion will become level.

2-326. MAIN CABIN SEATING EQUIPMENT (AIRCRAFT C AND D). On these aircraft, the accommodations are installed for 14 passengers in 8 swivel chairs and 2 divans; floor fittings are installed for the movable table, permitting the table to be used in either of 2 swivel chair groupings. For the description and handling of the divan, see paragraph 2-325.

2-327. REMOVAL OF MAIN CABIN SEATING EQUIPMENT (AIRCRAFT A THROUGH D). The four swivel seats (see paragraph 2-329) are attached to the floor beams with four 1/4-inch bolts, and require no special tools for removal. The swivel seats will not go through the door and, therefore, must be dismantled for removal from the aircraft by removing the seat from the pedestal base, the back from the side frames, and the cushion and arm rests from the remaining structure. The three-passenger divans in the forward end of the cabin have hinged seats and may be removed with ordinary tools.

2-328. INSTALLATION OF MAIN CABIN SEATING EQUIPMENT (AIRCRAFT A THROUGH D). Reverse the removal procedure.

2-329. MAIN CABIN MISCELLANEOUS EQUIPMENT (AIRCRAFT 1 THROUGH 96). The main cabin is equipped with canvas web straps and fittings *(see figure 2-41)* on which can be swung 27 litters in 3 tiers of 4 each, one tier of 3 each on the right side of the aircraft, and 3 tiers of 4 each on the left side. Canvas bags are attached to the frame between longerons No. 4 and No. 6 near the litter ends for the storage of canvas straps that support the litters. The straps are rolled up when not in use. Sidewall fittings consist of 14 hangers (No. X43A2455), 14 strap assemblies (No. 43G1953), and 42 bracket assemblies (No. 43B2330). When the collapsible passenger seats are installed in aircraft 1 through 96, the 42 bracket assemblies (No. 43B2330) are removed and stowed in a bag in the tail compartment. When the litters are installed, the troop benches are folded back to make room for them. Interior Fiberglas insulation blankets line the fuselage and are attached with snap-on fasteners. Two thermos jug containers are attached to the forward bulkhead, and one thermos assembly is attached to the aft bulkhead. Three first aid kits are installed, one in the flight compartment and two in the main cabin on the forward and aft bulkheads; the first aid kits are attached to the frame of the aircraft with sheet metal screws. One fire extinguisher is located on the left side of the fuselage near the floor between stations 429 and 450. A static line is rigged overhead for the use of paratroopers. Its fittings are bolted to the fuselage, to the left of the center line at stations 97 and 520. Cargo tie-down fittings are provided on the floor and side walls. Load adjusters are mounted on the forward side of the partition at station 47.

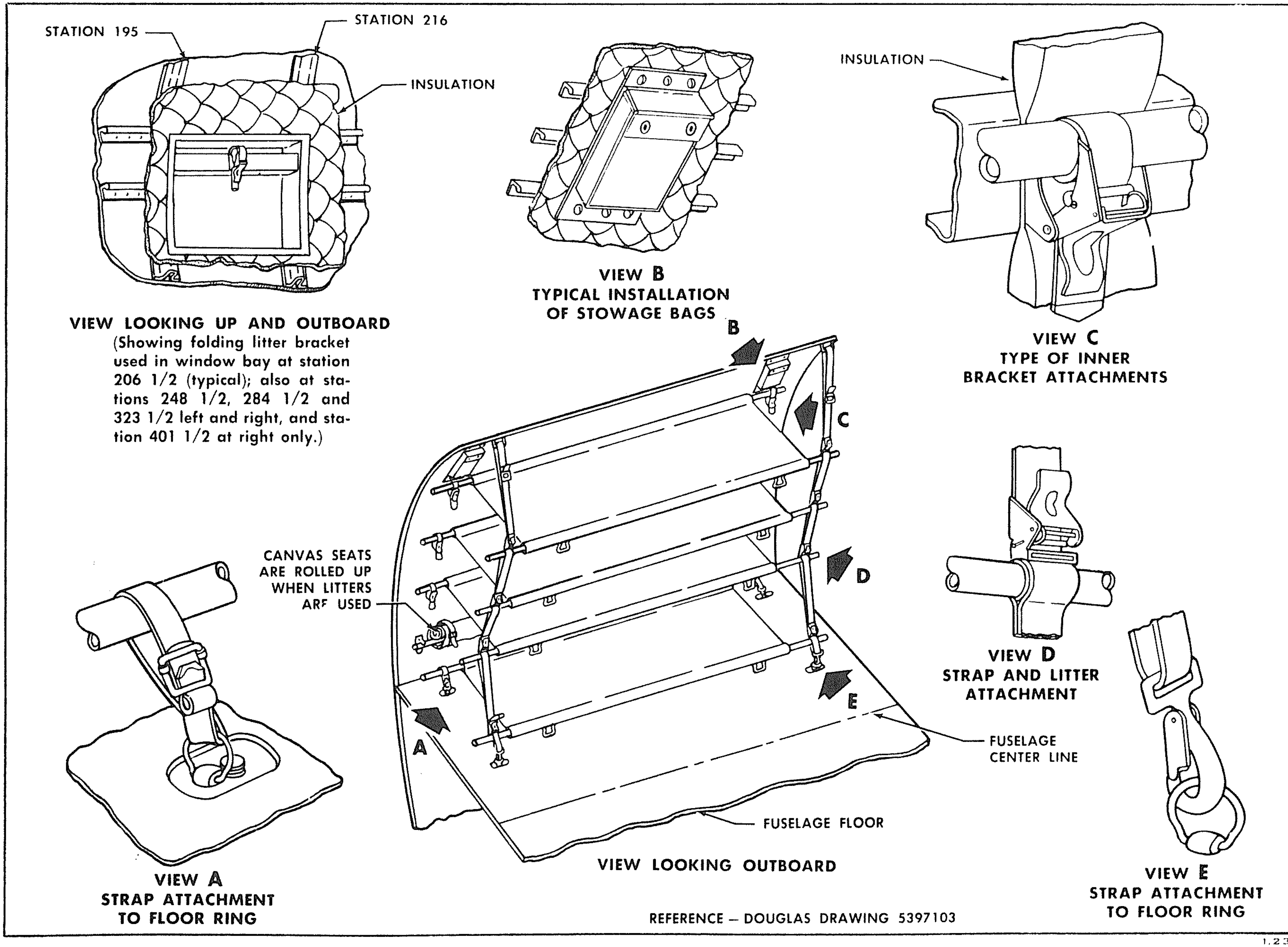


Figure 2-41. Canvas Belt Suspension of Litters in Cargo Compartment

2-330. MAIN CABIN MISCELLANEOUS EQUIPMENT (AIRCRAFT A THROUGH D). The aircraft is equipped with a folding table around which are installed the four swivel seats (see paragraphs 2-325 and 2-326). The table has hinged flaps and can be installed against the right aft main cabin bulkhead to provide desk space for the occupant of the right aft swivel seat. An executive's desk is bolted to the left side of the bulkhead at station 413, and is available to the occupant of the left aft swivel seat. Individual lap trays, attachable to the chair arms, are provided for each of the four swivel seats. When not in use, the trays are stowed two on each side of the aisle on the forward side of the bulkhead, at station 538. Aft of the bulkhead at station 413, a vestibule is included. The main cabin door is hinged on the lower edge; it may be lowered so that the steps, integral with the inside of the door, can form a companionway. Aft of the main door is a folding bench for two flight attendants or other crew members. On the right side of the vestibule, aft of the lavatory between stations 466 and 486, is a parachute rack with a capacity of 16 units. Aft of the parachute rack, against the aft bulkhead of the vestibule, is a coat room in which cradles are installed for the two oxygen cylinders that supply the passengers and two attendants. Aft of the vestibule is a compartment containing two bunks hinged to the sides of the fuselage and supported by two legs that fit into fittings on the floor. The head lining in the main cabin is turquoise Boltaflex; the curtains are of all-wool gabardine, and the lower panels are tan leather. The aft bunk compartment is lined with white Fiberglas sheet.

2-331. BUFFET (AIRCRAFT A THROUGH D). *(See figure 2-42.)* The buffet, located on the left side of the aircraft, aft of station 413, includes a top-loading refrigerator that has a capacity of 25 pounds and it can use either ice or dry ice. Two hot cups, four Stanley stainless steel vacuum bottles, and two hot plates are provided. A chime and signal light are installed for the purpose of calling flight attendants. The call buttons for the chime and signal light are located adjacent to individual lights in the main cabin area. An interphone handset is installed on the aisle side of the buffet. A window, equipped with vinyl curtains which match the main cabin head lining, is installed at approximately station 428 on the left side of the aircraft. Stowage space is provided under the buffet for glasses, service ware, and for lunches.

2-332. FUSELAGE LINING (AIRCRAFT 1 THROUGH 96). *(See figure 2-43.)* The interior fuselage walls, including the crew compartment, are covered with a ½-inch thick Fiberglas blanket with a vinyl-coated fabric on the inside, and a sheet of aluminum foil cemented to a vinyl film on the outside of the blanket. The blankets are attached by means of snap fasteners. Window flaps of the same material are stowed behind the lining between the windows, and may be installed in accordance with the instructions stenciled on the flaps.

2-333. FUSELAGE LINING (AIRCRAFT A THROUGH D). *(See figure 2-43.)* The main cabin is soundproofed with a 1½-inch Fiberglas blanket between the frames, and 3-inch Fiberglas blankets over the frames. The crew compartment interior fuselage walls are covered with a ½-inch thick Fiberglas blanket, with a vinyl-coated fabric and a sheet of aluminum foil cemented to the bottom of the blanket. The blanket is attached by means of snap fasteners and studs.

2-334. FLOOR COVERING (AIRCRAFT 1 THROUGH 96). *(See figure 2-44.)* The main cabin floor covering, between stations 97 and 450, consists of 45½-inch Geneerco No. 2 floor panels that are supported by five longitudinal beams centered approximately 20 inches apart. These longitudinal beams act as combination skid rails and attach locations for cargo-passenger seat tie-down fittings, spaced 10 inches apart in a forward and aft direction.

2-335. FLOOR COVERING (AIRCRAFT A THROUGH D). The floor is covered by a woolen, rubber-backed aircraft carpet, over a rubber waffle-weave vibration pad.

2-336. TAIL COMPARTMENT. The tail compartment is located aft of the lavatory and provides access to the control cables. Entrance to this compartment is through a door in the aft bulkhead of the lavatory.

2-337. TAIL CONE. *(See figure 2-45.)* The tail cone is attached to the aft end of the fuselage and is removable. The tail cone provides a favorable aerodynamic contour to the aft end of the fuselage. Structural mounting for the tail lights is also located on the tail cone.

2-338. REMOVAL OF TAIL CONE. To remove the tail cone, proceed as follows:

a. Disconnect the electrical wiring.

b. Disconnect the drain pipe to the urinal venturi.

c. Remove the screws attaching the rear end of the stabilizer fairing to the tail cone.

d. Remove the screws attaching the tail cone to the fuselage.

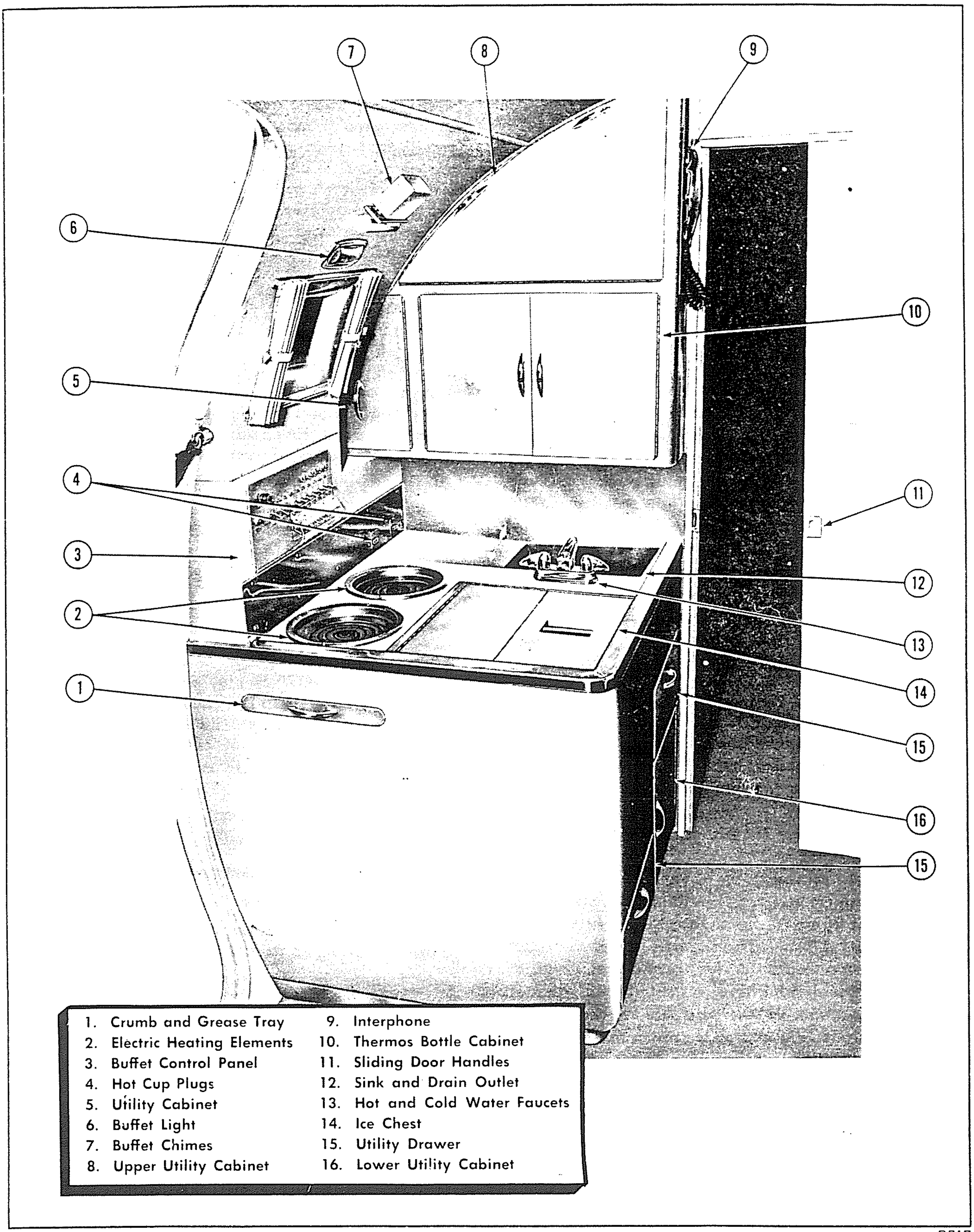
e. Remove the tail cone by pulling it aft.

2-339. INSTALLATION OF TAIL CONE. Reverse the removal procedure.

2-340. STOWAGE OF LOOSE EQUIPMENT. For information on the stowage of loose equipment, see figure 2-46.

2-341. FUSELAGE BOLT TORQUE VALUES. For information on the fuselage bolt torque values, see paragraph 2-453.

(Continued on Page 205)



1. Crumb and Grease Tray
2. Electric Heating Elements
3. Buffet Control Panel
4. Hot Cup Plugs
5. Utility Cabinet
6. Buffet Light
7. Buffet Chimes
8. Upper Utility Cabinet
9. Interphone
10. Thermos Bottle Cabinet
11. Sliding Door Handles
12. Sink and Drain Outlet
13. Hot and Cold Water Faucets
14. Ice Chest
15. Utility Drawer
16. Lower Utility Cabinet

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Figure 2-42. Buffet Installation

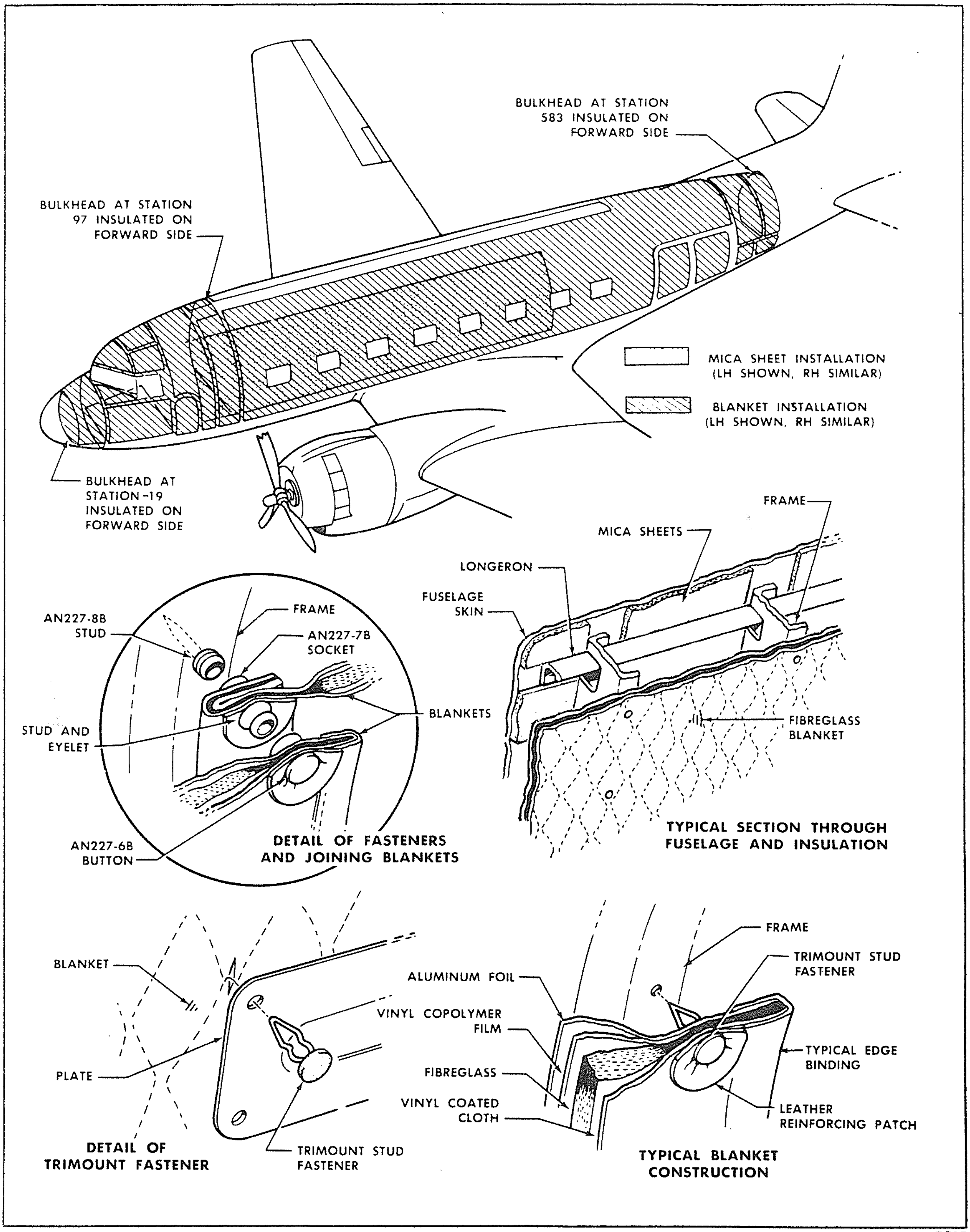

Figure 2-43. Soundproofing Installation (Aircraft 1 through 96)

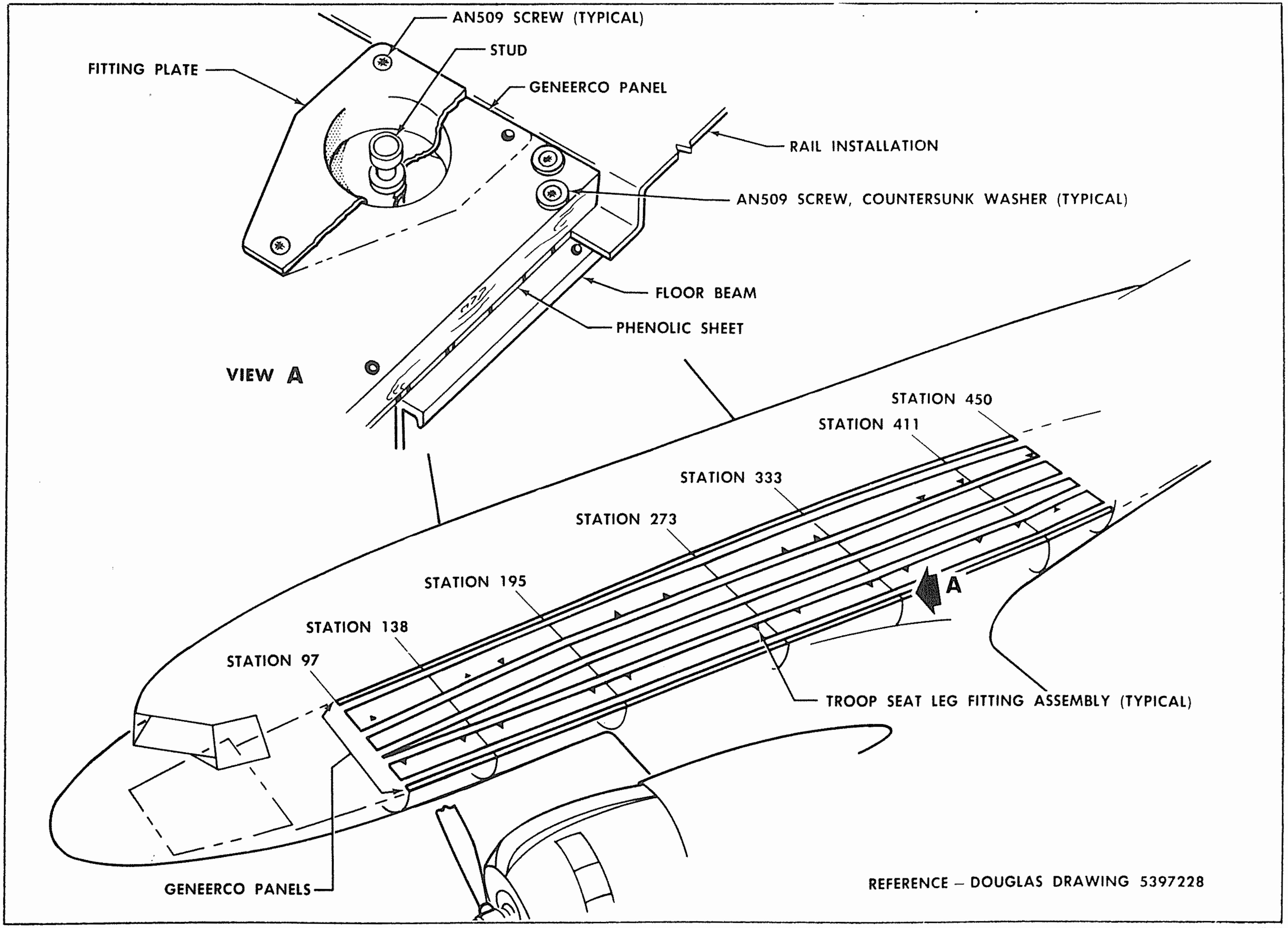

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Figure 2-44. Floor Installation – Main Cargo Compartment

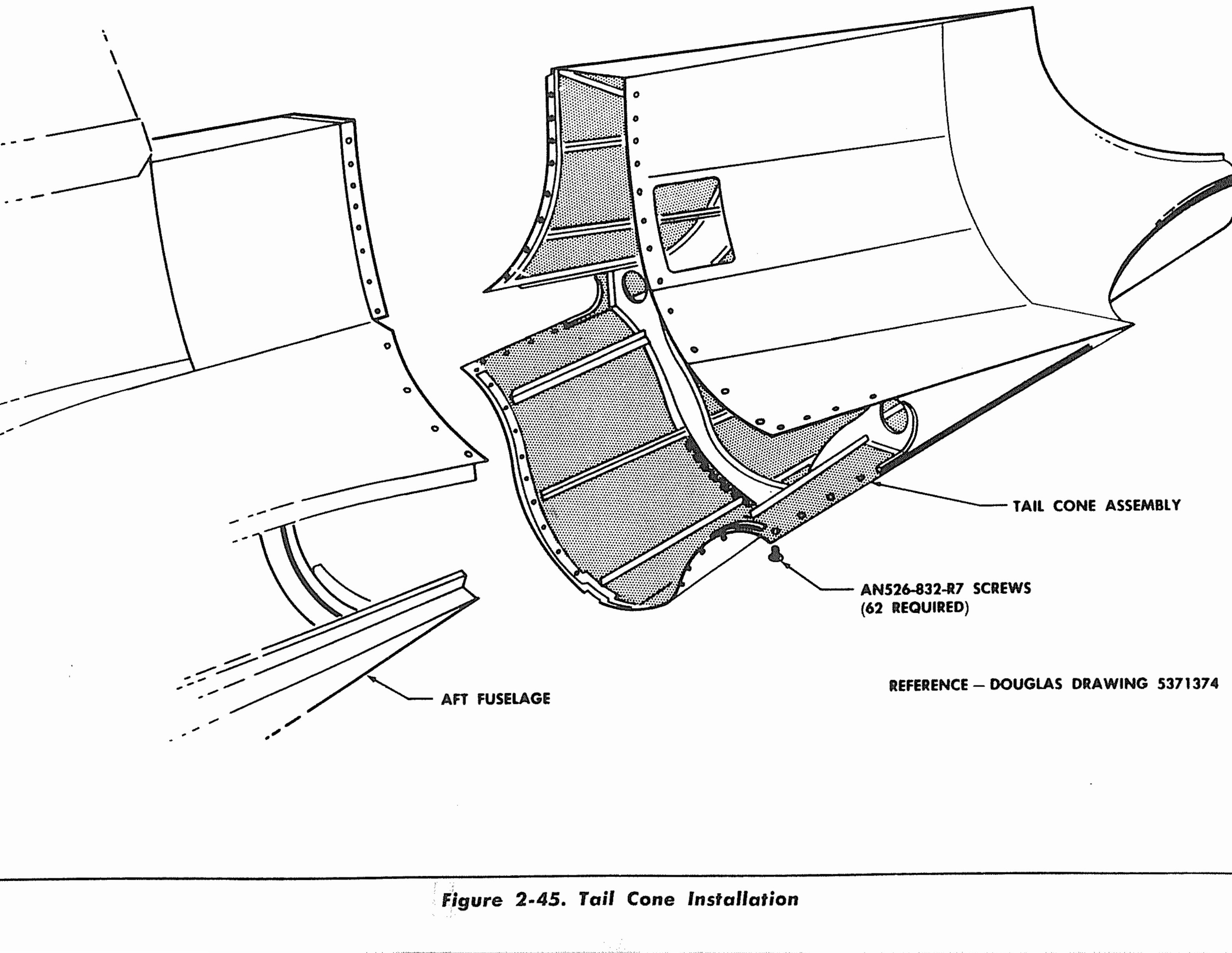
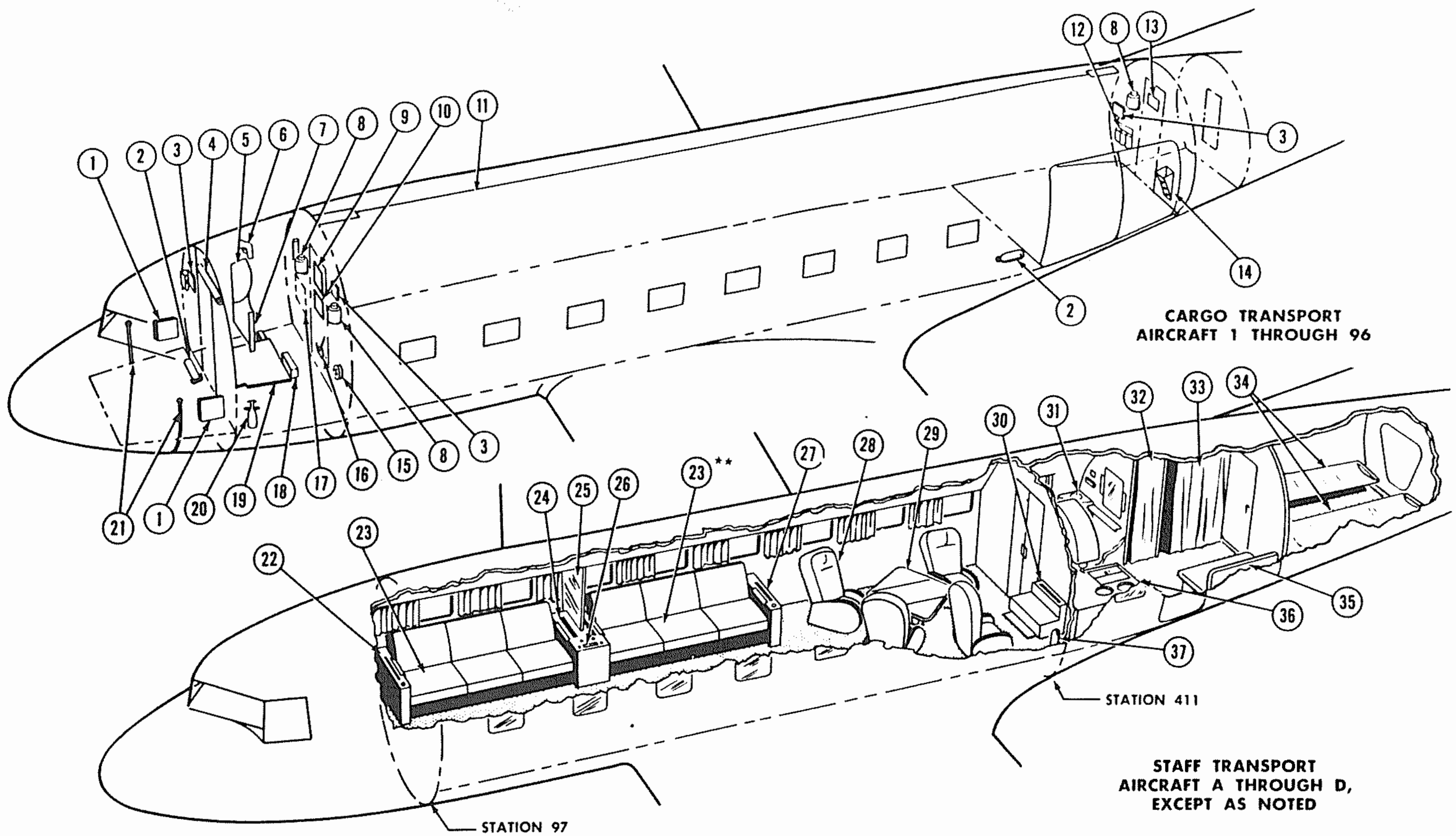


Figure 2-45. Tail Cone Installation



1. Map Case
2. Fire Extinguisher
3. First Aid Kit
4. Glare Curtain
5. Shroud Installation
6. Pyrotechnic Pistol Mount
7. Pistol Stowage
8. Jug Installation
9. Fuel System Diagram
10. Wiring Diagram Holder
11. Parachute Static Line Cable
12. Stowage Bag
13. Litter Installation Placard
14. Cabin Step
15. Snatch Block Stowage
16. Ax
17. Troop Bench Installation Placard
18. Navigator's Tool Holder
19. Navigator's Table
20. Sextant Stowage Installation
21. Relief Tube
22. Forward End Table (Typical)
*23. Divan (Typical)
24. Cabinet (Typical)
25. Lucite Partition (Typical)
26. Cabin Radio
27. Aft End Table (Typical)
†28. Hardman Seat
29. Removable Table
30. Desk
31. Lavatory
32. Parachute Rack
33. Coat Room
34. Crew Bunks
35. Crew Seat*
36. Buffet
37. Fire Extinguisher

*2 required on Aircraft C and D.
†8 required on Aircraft C and D.
**LH and RH divan replaced by noted seats (item 28) at this location

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Figure 2-46. Miscellaneous Equipment

(Continued from Page 199)

2-342. ALIGHTING GEAR.

2-343. DESCRIPTION. The complete hydraulically actuated alighting gear consists of a main alighting gear assembly, installed in each of the nacelles, and a tail wheel assembly installed in the fuselage, aft of station 624.

2-344. The main alighting gear wheels, when retracted, are completely enclosed by mechanically actuated doors. An alighting gear safety latch and a solenoid-operated latch prevent the inadvertent retraction of the alighting gear while the aircraft is on the ground. Each main alighting gear wheel is provided with a single-disc, self-adjusting multiple spot brake. An alighting gear control valve, located on the hydraulic power panel, directs hydraulic pressure to the alighting gear actuating cylinders, to extend or retract the alighting gear assemblies, as required.

2-345. The tail alighting gear, when in the retracted position, is only partially enclosed. In the extended position, the tail wheel may be swiveled 120 degrees on each side of the aircraft center line, or it may be locked in the trailing position to assist the pilot during the take-off or landing runs. The tail wheel lock lever is located on the control pedestal in the pilots' compartment.

2-346. The alighting gear hydraulic system is composed of the control valve, the thermal relief valve, the main and tail gear actuating cylinders, and the alighting gear hydraulic system pressure gage *(see figure 2-47)*. Hydraulic fluid at system pressure is directed to the alighting gear components from the pressure regulator to extend or retract the gear. Application of system pressure to the top of the piston causes the alighting gear to extend, and pressure applied to the lower side of the piston causes the gear to retract. A thermal relief valve is installed in the system to prevent damage to the retracting mechanism, or to its attaching structure, due to excessively high temperatures or from gust loads. The alighting gear hydraulic system pressure gage is installed on the hydraulic panel in the flight compartment.

2-347. TROUBLE SHOOTING OF ALIGHTING GEAR.

Trouble	*Probable Cause*	*Remedy*
a. Alighting gear does not respond to operation of control valve handle although constant pressure is indicated.	Faulty control valve.	Check adjustment of or replace control valve.
	Faulty actuating cylinder.	Check operation of actuating cylinder.
b. Shock struts do not operate smoothly.	Low air pressure.	Check air pressure (see paragraph 1-99).
c. Hard impact shock on landing.	Incorrect inflation of shock struts.	Check for leakage of air or fluid from shock struts. For correct filling and inflating procedure, see paragraph 1-99.
d. Fluid leakage around piston of shock strut.	Packing leakage, possibly due to pitted or rough surface on piston causing damage to packing.	Replace packing. Smooth-finish critical piston area with fine wet-or-dry sandpaper and oil, or replace parts if severely damaged.
e. Alighting gear solenoid does not function properly.	Switch on right alighting gear oleo is out of adjustment.	Adjust or replace switch. Adjust or replace solenoid.
f. Alighting gear indicator and warning lights do not operate properly.	Mechanical safety latch broken or out of adjustment.	Check and adjust safety latch *(see figure 2-48)*.
	Switches inoperative or out of adjustment.	Check switches, and adjust or replace if necessary.
g. Main gear doors do not close completely.	Improper adjustment of linkage.	Adjust door linkage. Check for full retraction of alighting gear.
h. Grinding noise in wheel when aircraft is towed or taxied.	Wheel bearing failure.	Remove bearing and install a new part.
i. When air brake handle is turned, brakes respond but indicated air pressure continues to fall.	External leakage in fittings or pipes.	Paint fittings and pipes with soapy solution. Air bubbles indicate leakage. Tighten fittings or replace damaged parts.
	Air leakage through brake shuttle valves.	Replace parts as necessary.

Note

Leakage in air brake system can be checked by disconnecting the hydraulic pipe from the shuttle valve, and by noting the leakage in the valve with the brakes released and with the brakes set.

j. When brakes are applied, pressure fluctuates appreciably. Pressure regulator operates at very short intervals or may stay closed, allowing pressure to remain low.	External leakage in brake system.	Inspect pipes, fittings, and units for external leakage.
	Internal leakage in brake control valve.	If no leakage is visible, check for trouble in brake control valve.

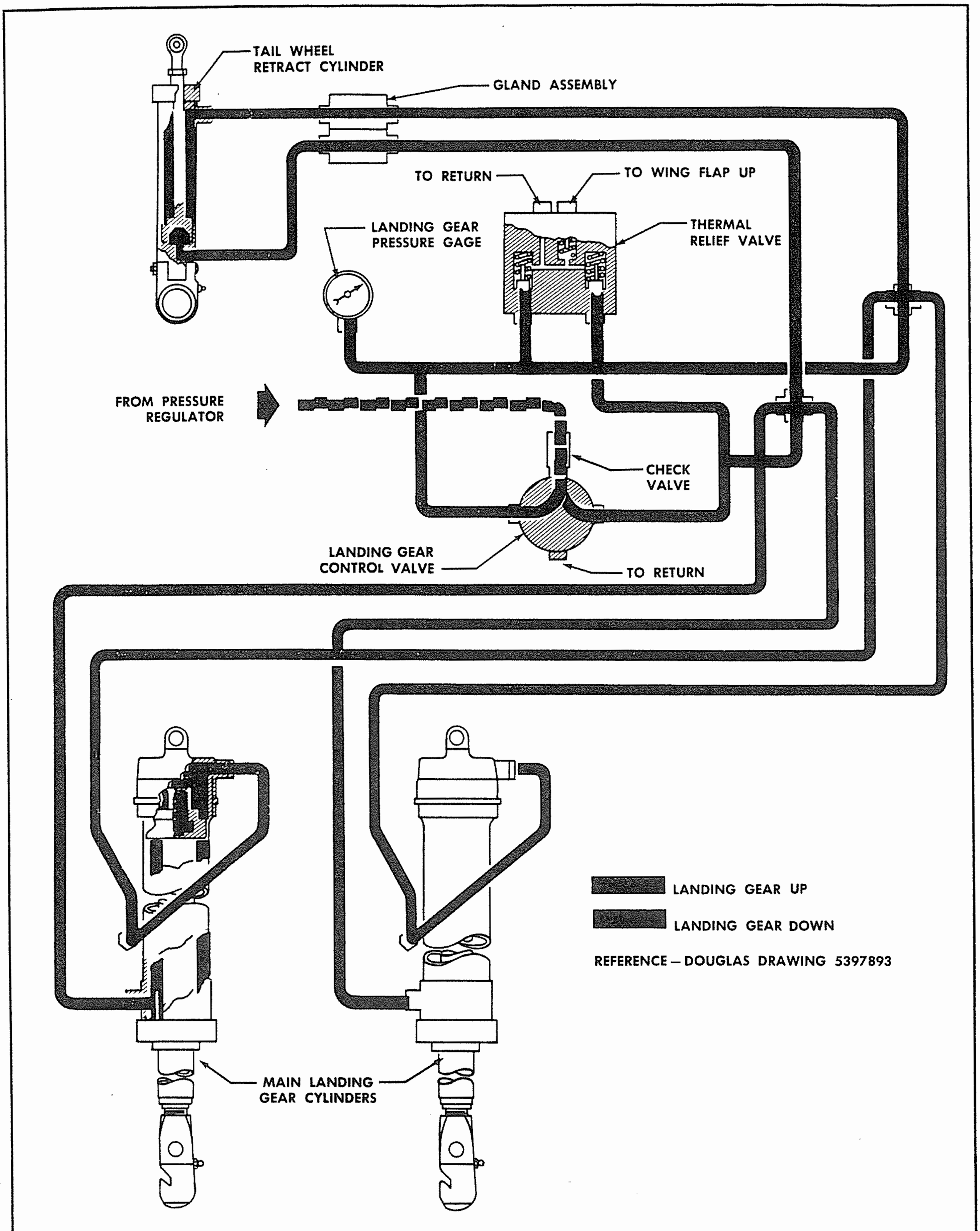


Figure 2-47. Alighting Gear Hydraulic System

Trouble	Probable Cause	Remedy
k. Brakes do not respond or respond only partially to application of brakes. Pedals offer little or no resistance to toe pressure. No change in hydraulic pressure.	Faulty adjustment or mechanical failure between brake pedals and control valve.	Isolate trouble by inspecting the mechanical system, and repair or adjust as necessary.
	Faulty adjustment of brake control valve.	Adjust the brake control valve.
	Hydraulic brake pressure low.	Check adjustment of brake control valve.
	Spot linings worn out.	Replace the spot linings when pin has receded to a depth of ¼ inch, measured from the shoulder of torque nut with brakes applied or parked.
	External leak in system.	Check for leaks. If piston seal is worn, replace it.
	Loss of fluid around automatic adjusting pin.	Tighten the lock nut to 25 foot-pounds torque.
l. Dragging brakes.	Binding of pedal linkage.	Check and free the point of binding.
	Parking brake improperly adjusted.	Check and adjust parking brake properly so that binding is eliminated and brake works freely.
	Dirty or broken return springs in brake cause piston to fail to return.	Disassemble brake and clean it with denatured alcohol. Replace the return springs. Refill system and bleed.
	No clearance between brake lining and disc.	Loosen adjusting pin lock nut. Pull adjusting pin out as far as possible. Insert 0.015-inch shim between lining and disc. Tighten lock nut to 25 foot-pounds torque. Remove the shim.
	Brake valve improperly adjusted.	Adjust brake valve.
m. Alighting gear fails to retract.	Improper adjustment or improper operation of control valve.	Replace control valve. **Note** If only one gear fails to retract, check for restriction in pipe.
n. Alighting gear fails to latch in down position when hydraulic system pressure is zero.	Bungee damaged or broken.	Replace bungee.

2-348. MAIN ALIGHTING GEAR. Each main alighting gear wheel is mounted on an axle which is clamped between the two air-oil shock absorber struts. The upper ends of the two shock struts are connected to a rigid truss, the upper fittings of which are hinged to the forward face of the front spar in each nacelle. Triangular links are attached to the lower ends of each of the shock strut piston tubes. The aft corners of the triangular links are bolted to rods, which in turn are bolted to shock strut cross bracing. The lower corners of the triangular links are bolted to the rear brace yoke, which is hinged at the upper end to the wing structure. The hydraulic actuating cylinder piston rod is connected to the upper truss, and the actuating cylinder is attached to the nacelle structure aft of the firewall. When the actuating cylinder piston is retracted, the upper truss is pulled forward and upward into the nacelle. The shock absorber and wheel assembly, being hinged to the lower ends of the upper truss, are guided upward into the nacelle by the rear brace yoke. When the alighting gear is fully retracted, the projecting ends of the axles are held against rubber bumpers, located in the sides of the nacelles. An elastic shock cord bungee is provided at the top of the alighting gear installation to balance the weight of the alighting gear during retraction or extension.

2-349. REMOVAL OF MAIN ALIGHTING GEAR. To remove a main alighting gear assembly, proceed as follows:

a. Remove cargo and fuel so that the gross weight corresponds to the requirements shown in figure 1-19.

b. Install the wing jacks (see paragraph 1-64). Make certain that the wing jacks engage the single point jack pads on the lower surface of the inner wing section, inboard of the nacelle, under the front spar.

c. Remove the bungee (see paragraph 2-409).

d. Disconnect the alighting gear actuating cylinder piston rod from the upper truss by removing the attaching bolt and raising the safety latch.

e. Disconnect and cap the brake hydraulic and air pipes at the upper end of the rear brace yoke.

f. If a suitable stand is available to support the gear, each main alighting gear may be removed as an assembly. Remove the bolts that attach the shock struts to the upper truss and the bolt that attaches the upper end of the rear brace yoke to the wing structure; remove the alighting gear assembly. The upper truss may then be removed by removing the attaching bolts.

g. If no suitable stand is available to support the gear, remove the wheel and axle assembly (see paragraph 2-447). Then remove the rear brace yoke by removing the attaching bolts. Remove the shock strut assemblies from the upper truss and remove the truss.

2-350. INSTALLATION OF MAIN ALIGHTING GEAR. Reverse the removal procedure.

2-351. ADJUSTMENT OF MAIN ALIGHTING GEAR AFTER INSTALLATION.

a. Make a careful check for adequate clearance between the alighting gear and the nacelle structure. Jack up the aircraft, according to the procedure given in paragraph 1-64. Slowly retract the alighting gear by operating the hydraulic hand pump.

b. Carefully inspect the alighting gear mechanical safety latch (see paragraph 2-357).

c. Tighten the axle clamp bolt nuts (two on each axle) to a torque of 270 to 320 foot-pounds.

d. Tighten the bushing bolt nuts (two on each shock strut assembly) to a torque of 23 to 25 foot-pounds.

e. Check the action of the alighting gear solenoid latch.

f. Check the action of the alighting gear warning horn system and the alighting gear indicator and warning light system.

g. Check the action of the main alighting gear retracting mechanism.

2-352. OPERATION OF ALIGHTING GEAR CONTROLS.

2-353. GENERAL. The alighting gear is controlled by the alighting gear control valve, located on the hydraulic control panel. Hydraulic fluid circulates through the control valve transmitting the pressure that retracts and extends the alighting gear. For detailed information on the alighting gear control valve, see paragraph 2-405.

2-354. To retract the alighting gear, observe the following procedure: Hold the mechanical safety latch control handle in the LATCH RAISED position, and move the alighting gear control valve handle to the UP position. This action causes the control valve to direct hydraulic fluid, under pressure, to the lower end of the actuating cylinder, retracting the piston rod. As the piston rod retracts, the alighting gear is pulled into the nacelle and the alighting gear door mechanism operates to close the doors under the retracted wheels.

Note

There are no mechanical locks to hold the alighting gear in the UP position. Only fluid pressure in the actuating cylinder retains the alighting gear in the retracted position.

2-355. To extend the alighting gear, proceed as follows:

a. Make certain that the mechanical safety latch control handle is in the SPRING LOCK position.

b. Place the alighting gear control valve handle in the DOWN position. The alighting gear control valve will direct hydraulic fluid, under pressure, to the upper end of the alighting gear actuating cylinder. The piston rod will extend, forcing the alighting gear to the down position, after opening the alighting gear doors.

2-356. ALIGHTING GEAR SAFETY DEVICES. Certain devices incorporated in the alighting gear system are intended to lock the alighting gear in the down position and to prevent inadvertent retraction while the aircraft is on the ground. These devices include the mechanical safety latch system, the alighting gear solenoid latch, the alighting gear safety pins, and the alighting gear position indicator and warning light system.

2-357. MECHANICAL SAFETY LATCH CONTROLS. *(See figure 2-48.)* The mechanical safety latch control system incorporates a two-way cable system which extends from the mechanical safety latch control handle, located on the floor between the pilot's and co-pilot's seats, to the mechanical safety latch assemblies in the nacelles. The control handle has three positions: LATCH RAISED, which releases the alighting gear for retraction; SPRING LOCK, which receives and latches the alighting gear actuating cylinder hook; and POSITIVE LOCK, for safety-locking the actuating cylinder piston rod hook when the gear is extended completely. The control cable is connected by linkage to a catch and dog on the alighting gear control valve. The catch and dog prevent the control valve from being removed into the UP position if the mechanical safety latch is in the SPRING LOCK or POSITIVE LOCK position.

2-358. MECHANICAL SAFETY LATCH ASSEMBLY. *(See figure 2-48.)* A spring-loaded, cable-operated safety latch assembly is provided in each nacelle, on the forward face of the wing front spar, to lock the alighting gear in the extended position. The unit automatically latches when the gear is fully extended, by engaging a slot in the lower end of the actuating cylinder rod. Latches for both gear are controlled simultaneously by the safety latch control cables.

2-359. REMOVAL OF MECHANICAL SAFETY LATCH ASSEMBLY. To remove the mechanical safety latch assembly, proceed as follows:

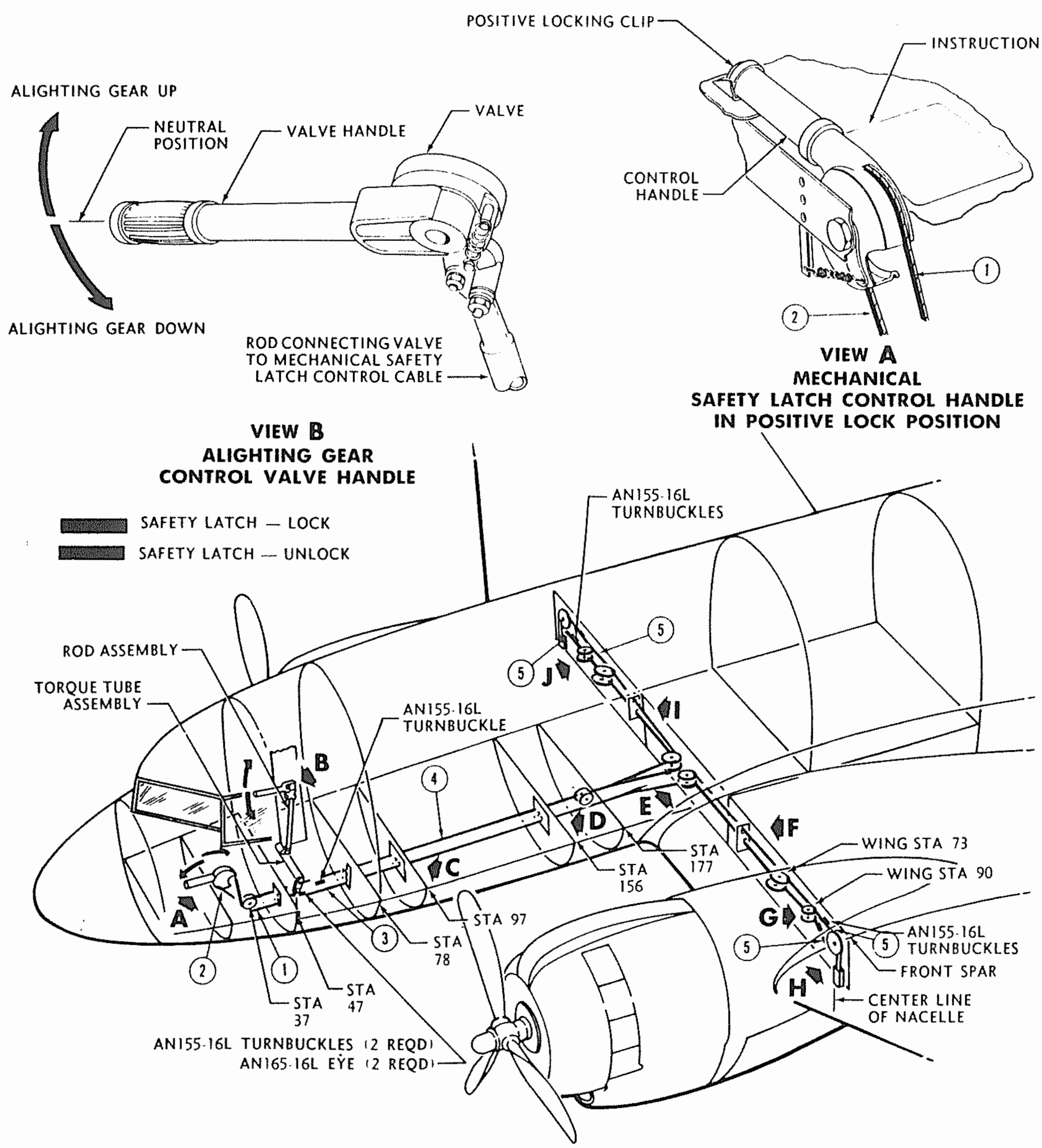
a. Remove the safety wire and disconnect the two cables at the safety latch.

b. Remove the cotter pins, nuts, and bolts at the upper attachment point.

c. Remove the safety wire and remove four machine screws at the lower end of the assembly.

d. Remove the four bolts and nuts, and remove the guide at the lower end of the mechanical safety latch.

(Continued on Page 215)

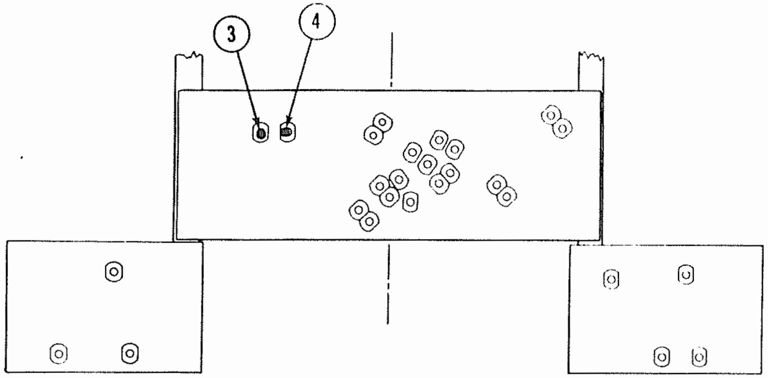


Notes:

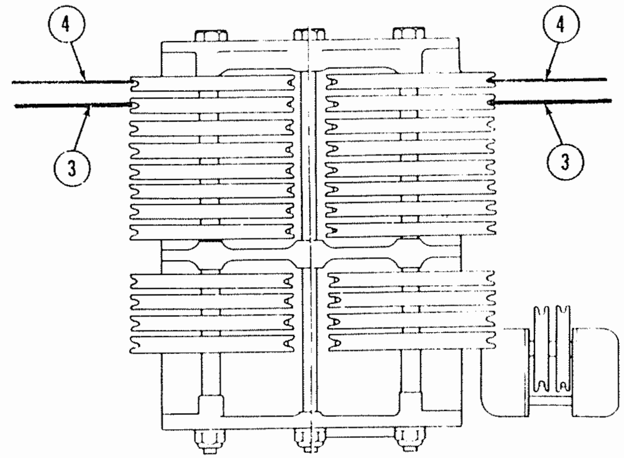
1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 6.
2. For cable assembly details, see Sheet 6 and 7.
3. Tension cables in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
4. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart, paragraph 2-453.
5. For Rigging and Adjustment Diagram, see Sheet 5.
6. Reference — Douglas drawing 5367889.

A—Safety Latch Handle,	see Sheet 1
B—Control Valve Handle,	see Sheet 1
C—Fairleads — Station 97,	see Sheet 2
D—Fairlead — Station 156,	see Sheet 2
E—Pulley Brackets — Front Spar,	see Sheet 2
F—Fairlead — Left Wing Station 45,	see Sheet 3
G—Pulley Brackets — Left Wing Front Spar,	see Sheet 3
H—Safety Latch Assembly,	see Sheet 1
I—Fairlead — Right Wing Station 45,	see Sheet 4
J—Pulley Brackets — Right Wing Front Spar,	see Sheet 4

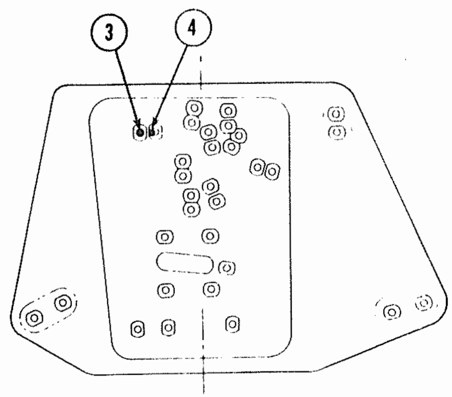
Figure 2-48 (Sheet 1 of 7 Sheets). Mechanical Safety Latch Control System — Key Drawing



VIEW C
FAIRLEADS AT STATION 97
LOOKING FORWARD



VIEW E
PULLEY BRACKETS AT STATION 22
(FRONT SPAR) LOOKING AFT



VIEW D
FAIRLEAD AT STATION 156
LOOKING FORWARD

SAFETY LATCH-LOCK
SAFETY LATCH-UNLOCK

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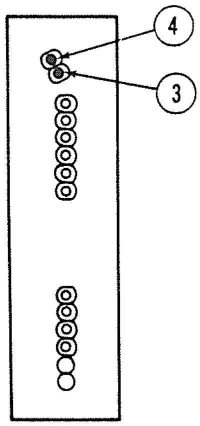
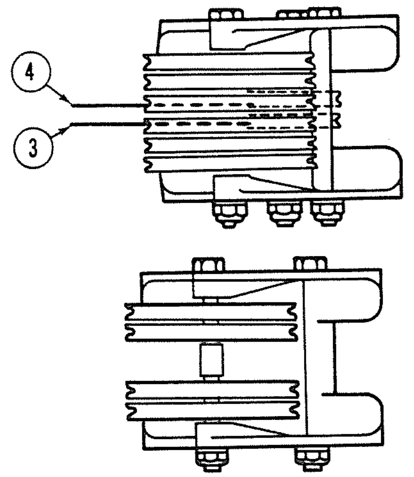
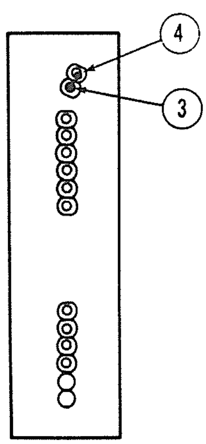
Figure 2-48 (Sheet 2 of 7 Sheets). Mechanical Safety Latch Control System — Fairleads, Stations 97 and 156; Pulley Brackets, Front Spar



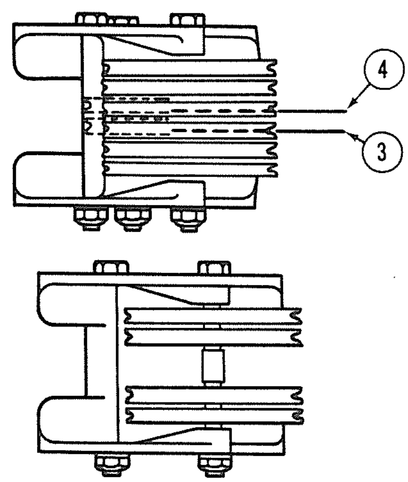
VIEW F
FAIRLEAD AT LEFT WING
STATION 45 LOOKING INBOARD



VIEW G
PULLEY BRACKETS AT FRONT
SPAR IN LEFT NACELLE
LOOKING AFT



VIEW I
FAIRLEAD AT RIGHT WING
STATION 45 LOOKING INBOARD



VIEW J
PULLEY BRACKETS AT FRONT
SPAR IN LEFT NACELLE
LOOKING AFT

SAFETY LATCH LOCK

SAFETY LATCH UNLOCK

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Figure 2-48 (Sheet 3 of 7 Sheets). Mechancial Safety Latch Control System — Fairleads, Left and Right Wing Stations 45; Pulley Brackets, Left and Right Nacelle Front Spars

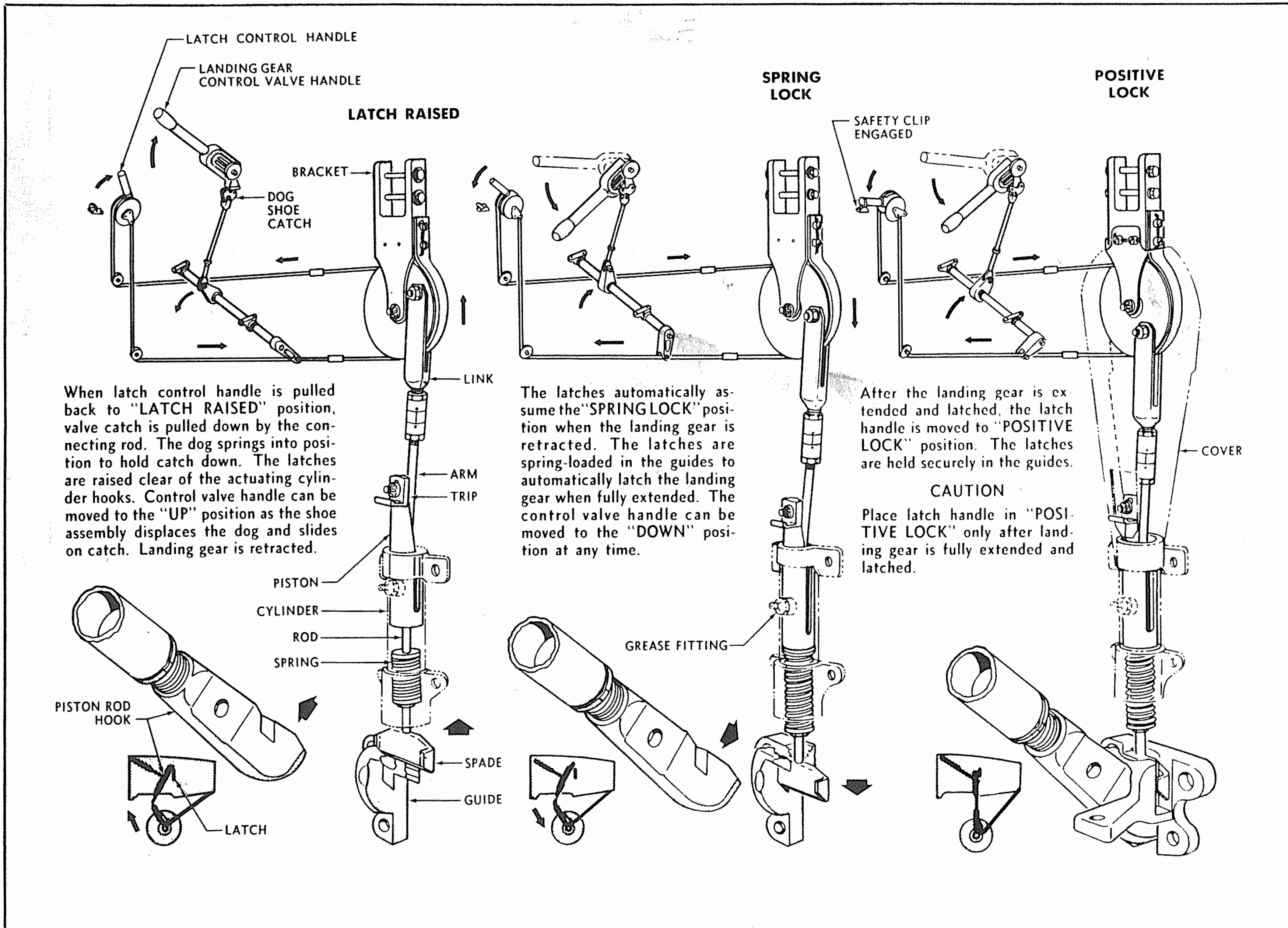

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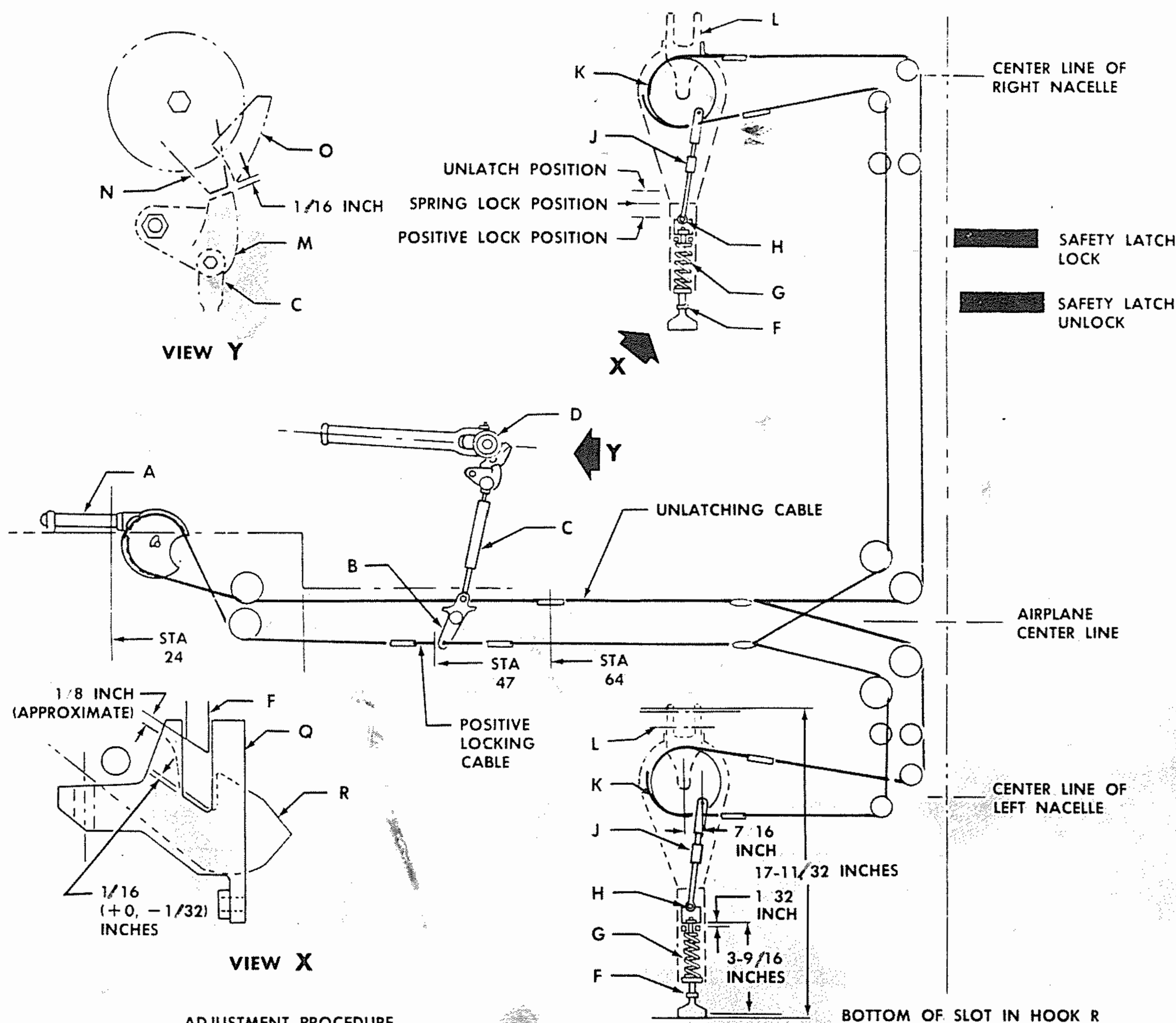
Figure 2-48 (Sheet 4 of 7 Sheets). Mechanical Safety Latch Control System — Latch Sequence



ADJUSTMENT PROCEDURE

NACELLE SECTION LATCH INSTALLATION SEQUENCE

1. Lower alighting gear.
2. With hydraulic pressure in retract strut, adjust hook (R) to locate sides of slots of hook and guide (Q) equally spaced, as shown in View E. Stake washer to lock.
3. Adjust guide sideways to suit hook and adjust vertically to obtain 1/16 (+0, −1/32) inches shown in View E.
4. Locate bracket (L) to 17 11/32 inches.
5. Adjust latch (F) to 3 9/16 inches and tighten lock nut.
6. With latch (F) in hook (R) slot compress spring (G) 1/32 inch and locate cylinder (H).
7. Adjust turnbuckle assembly (J) to give 7/16 inch dimension at pulley (K) with pin (H) in positive lock position, and tighten lock nuts. There must be at least 1/4 inch thread engagement at both ends of turnbuckle.

LATCH CONTROL INSTALLATION SEQUENCE

1. Before installing rod (C), even all cable turnbuckles to bury approximately equal lengths of thread.
2. With latch handle (A) in SPRING LOCK position, readjust turnbuckles as necessary to put 30 pounds maximum tension in cables at temperature of 70 F (21 C).
3. With latch handle (A) in LATCH RAISED position, adjust nacelle cable turnbuckles so that latches (F) are at equal heights (approximately 1/8 inch) above hook (R).
4. With latch handle (A) in POSITIVE LOCK position, adjust turnbuckles at station 52 so that tension in positive lock cable is 20 to 30 pounds greater than tension in unlatch cable.
5. With latch handle (A) in SPRING LOCK position, adjust fore and aft turnbuckles of locking cable at arm (B) so that arm is at 90 degrees with floor line.
6. Install rod (C) with alighting gear control valve (D) in NEUTRAL position and latch handle (A) held full back in LATCH RAISED position. Adjust rod (C) to give 1/16 inch clearance between dog (N) and cam (M).
7. With alighting gear control valve (D) in UP position, and cam (M) resting on shoe (O), latches (F) must clear hooks (R) by 3/32 inch minimum.

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Figure 2-48 (Sheet 5 of 7 Sheets). Mechanical Safety Latch Control System — Adjustment Diagram

MECHANICAL SAFETY LATCH CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)
1	2369624-501	1	A	22 3/4	3/32 dia 7x7 flex	RA2487-3	AN669L3 LH		
2	2369624-503	1	B	33 5/8	3/32 dia 7x7 flex	RA2487-3	AN669L3 RH		
3	4367762-1	1	C	(L_1) 267 1/4 (L_2) 159 3/8 (L_3) 107 7/8	3/32 dia 7x7 flex	AN669L3 LH (5) AN669L3 LH	28-2-G	28-2-G	AN669L3 LH
4	4367762-501	1	C	(L_1) 262 1/8 (L_2) 148 3/4 (L_3) 113 3/8	3/32 dia 7x7 flex	AN669L3 LH (5) AN669L3 LH	28-2-G	28-2-G	AN669L3 LH
5	2369624-1	4	D	9 5/8	3/32 dia 7x7 flex	AN669L3 RH	RA2487-3		

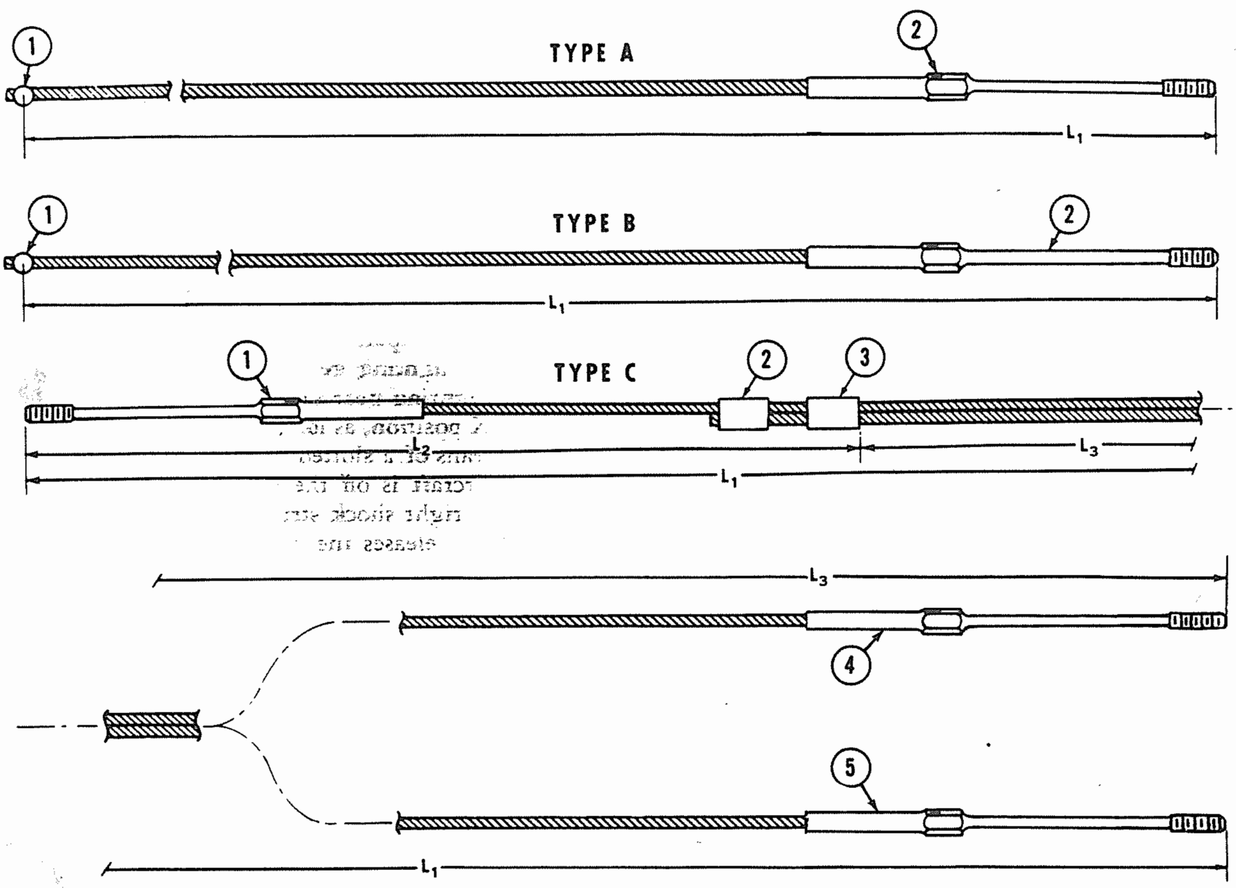

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Figure 2-48 (Sheet 6 of 7 Sheet). Mechanical Safety Latch Control System — Cable Chart and Cable Assemblies

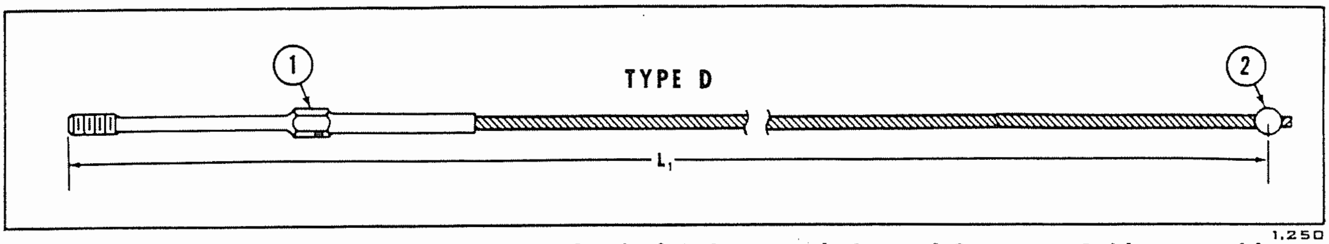


Figure 2-48 (Sheet 7 of 7 Sheets). Mechanical Safety Latch Control System — Cable Assembly

(Continued from Page 208)

2-360. INSTALLATION OF MECHANICAL SAFETY LATCH ASSEMBLY.

a. Install the guide at the lower end of the mechanical safety latch by replacing the four bolts and nuts.

b. Secure the lower end of the assembly with four machine screws, and safety.

c. Install the nuts, bolts, and cotter pins at the upper attachment point.

d. Connect the turnbuckles at the safety latch.

e. Adjust the mechanical safety latch controls (see paragraph 2-364).

f. Safety the turnbuckles.

2-361. MECHANICAL SAFETY LATCH CONTROL CABLES. *(See figure 2-48.)* The mechanical safety latch control cables extend from the control lever in the cockpit, aft through the lower part of the fuselage to the front spar pulley bracket assembly, and then outboard to the latch assembly in each nacelle.

2-362. REMOVAL OF MECHANICAL SAFETY LATCH CONTROL CABLES.

a. Remove the necessary access doors and floor panels.

b. Disconnect the mechanical safety latch control cables at the turnbuckles outboard of wing station 90. Thread the cables.

c. Disconnect the cables from the safety latch pulleys in the nacelles, and remove them from the aircraft.

d. Disconnect the mechanical safety latch control cables at the turnbuckles aft of station 47.

e. Disconnect the forward control cables from the mechanical safety latch control handle, located on the floor between the pilot's and co-pilot's seats. Remove the cables.

f. Draw the intermediate cables through the floor panel opening adjacent to the hydraulic control panel, removing the cable guard pins and grommets as necessary. Pull the cables slowly to prevent chipping of the micarta pulleys. Use two men to perform this operation; one is to pull the cables and the other to guide the cable ends through the pulley brackets and fairleads.

2-363. INSTALLATION OF MECHANICAL SAFETY LATCH CONTROL CABLES.

a. Route the intermediate mechanical safety latch control cables aft through the fuselage. Replace the guard pins and fairlead grommets as necessary.

b. Connect the forward control cable to the mechanical safety latch control handle on the floor between the pilot's and co-pilot's seats.

c. Connect the cables with turnbuckles aft of station 47.

d. Attach the cables to the mechanical safety latch pulleys in the nacelles.

e. Connect the cables outboard of wing stations 10 with turnbuckles.

f. Safety all turnbuckles.

g. Replace the access doors and floor panels.

2-364. ADJUSTMENT AND OPERATIONAL TESTS OF MECHANICAL SAFETY LATCH CONTROLS. For a complete description of the procedure to be followed in adjusting and testing the mechanical safety latch controls, see figure 2-48.

2-365. ALIGHTING GEAR SOLENOID LATCH. A solenoid-operated latching device, installed adjacent to the alighting gear latch lever, automatically holds the alighting gear safety latch lever in the POSITIVE LOCK position, as long as the aircraft is on the ground, by means of a slotted latch plate. When the weight of the aircraft is off the shock struts, a switch, attached to the right shock strut, energizes the solenoid which in turn releases the alighting gear latch lever. When the alighting gear reaches the fully retracted position, the switch de-energizes the solenoid.

2-366. ALIGHTING GEAR GROUND SAFETY PINS. *(See figure 2-49.)* The alighting gear ground safety pins are installed in the alighting gear linkage to prevent the inadvertent retraction of the alighting gear while the aircraft is on the ground. When not in use, the safety pins are stowed in the canvas bag on the forward face of station 538 bulkhead.

2-367. ALIGHTING GEAR POSITION INDICATOR AND WARNING LIGHT SYSTEM. The alighting gear position indicator is installed on the right side of the pilots' main instrument panel. The indi-

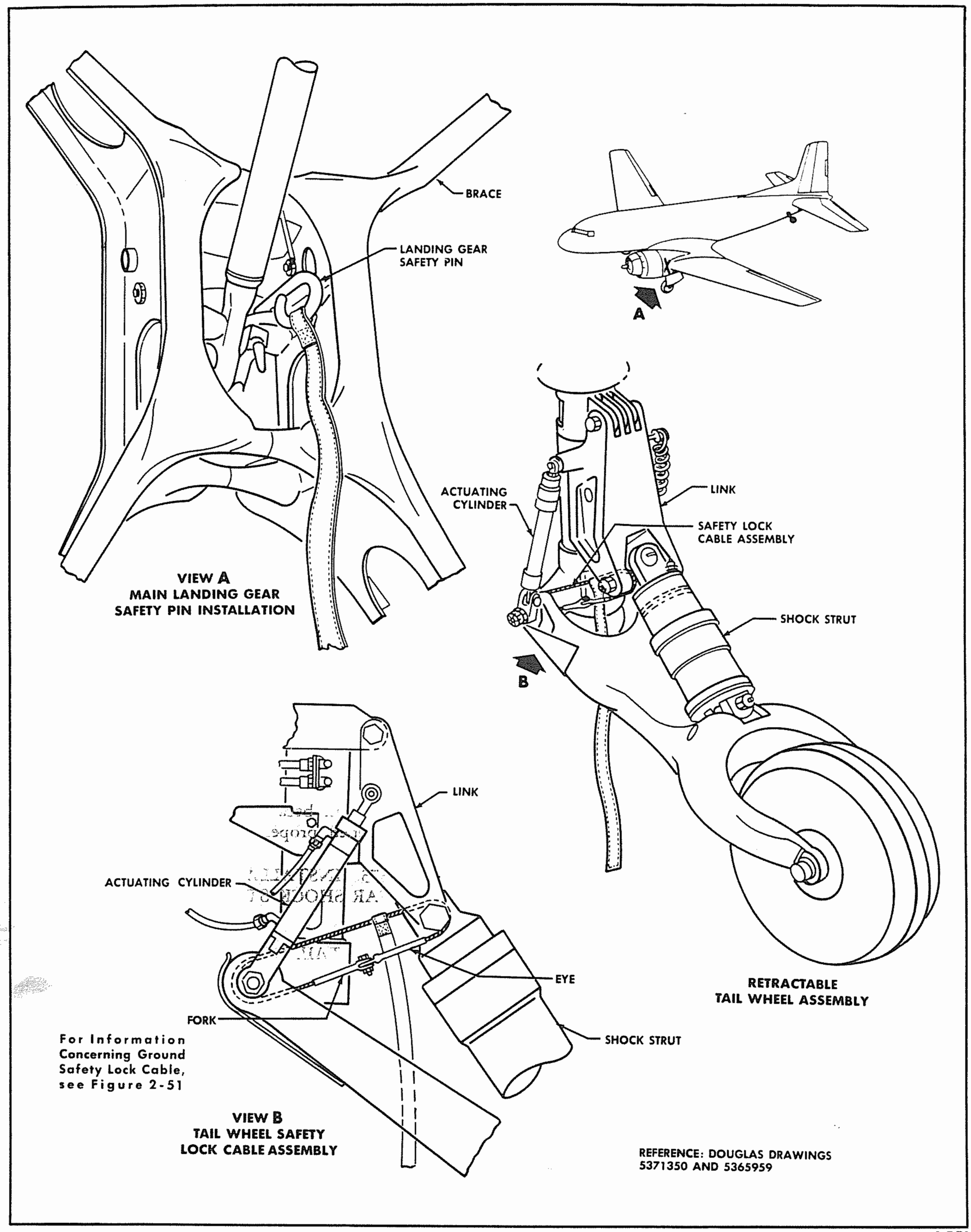


Figure 2-49. Alighting Gear Safety Pin Installation

cator incorporates three windows, one for each alighting gear, with markers which appear behind the windows to indicate the positions of the respective alighting gear legs. A single red light is installed adjacent to the position indicator to serve as a warning when the alighting gear is in an unsafe position. When the aircraft is on the ground, and the power is ON, an alighting gear wheel silhouette will appear in each of the indicator windows. When the aircraft is in the air, and the alighting gear control is moved to the UP position, the red warning light will come on and stay on until the alighting gear is fully retracted, as indicated by a red and white striped marker, which will appear in each of the indicator windows.

2-368. ALIGHTING GEAR WARNING HORN SYSTEM. An alighting gear warning horn, installed on the left wall of the pilots' compartment, will sound a warning if either or both of the throttles are retarded past the one-quarter-open position when the alighting gear is retracted or in any intermediate position.

2-369. MAIN ALIGHTING GEAR SHOCK STRUTS. *(See figure 2-50.)* Each main alighting gear wheel is mounted between two air-oil shock absorber struts, which absorb the alighting and taxiing loads. Alighting loads are absorbed mainly by forcing the hydraulic fluid in the strut through an orifice in the lower chamber of the cylinder; taxiing loads, however, are absorbed mainly by the compression of the air in the upper chamber of the cylinder. The main alighting gear wheel axle is clamped at each end by a fitting connected to the piston rods of the two struts. Rigid bracing connects the upper ends of the two struts, which in turn are connected to the upper truss. A rigid tube is installed between the two struts as an aid to the balancing of the air pressure in the two members.

2-370. SERVICING MAIN ALIGHTING GEAR SHOCK STRUTS. To service the main alighting gear shock struts, proceed as follows:

a. Before and after each flight, clean the piston rods of the alighting gear shock struts with a cloth soaked in hydraulic fluid. If hard dirt or grit is encountered, use kerosene as a solvent to loosen the dirt. Re-lubricate the piston rods with hydraulic fluid after the dirt is removed.

b. For filling and inflating procedures, see paragraph 1-99.

c. After the shock strut is inflated, test the air valve, valve core, and filler plug carefully by using soapy water.

2-371. REMOVAL OF MAIN ALIGHTING GEAR SHOCK STRUT. To remove the main alighting gear shock strut, make certain that the alighting gear ground safety pins are installed, and proceed as follows:

a. Jack up the aircraft (see paragraph 1-64).

b. Release the air pressure in the shock strut by loosening one filler plug until all air has escaped.

c. Disconnect and remove the balance pipe from the shock strut to be removed.

d. Disconnect the rear brace strut from the shock strut.

e. Remove the axle clamp bolt from the shock strut to be removed.

f. Remove the bolt that connects the shock strut upper brace.

g. Remove the bolt that attaches the shock strut to the upper truss, and remove the shock strut from the aircraft.

h. Remove the upper brace arm and the triangular links from the shock strut before disassembly.

2-372. MINOR REPAIR AND REPLACEMENT OF MAIN ALIGHTING GEAR SHOCK STRUTS. A seepage of hydraulic fluid around the filler plug or valve will usually indicate a leak. Tighten the filler plug and/or replace the air valve as necessary. Take care to prevent damage caused by excessive tightening of the air hose connection at the air valve. If the lip of the air valve body is damaged, install a new part. Use air valve caps at all times, and tighten to a torque of 15 to 20 inch-pounds with a small wrench. If fluid leakage at the shock strut occurs, tighten the packing nut firmly. If moderate tightening does not stop the fluid leakage, replace the packing. If the shock strut is damaged or is not functioning properly, replace it with a serviceable unit.

Note

If tightening of the strut packing becomes necessary, release the air pressure in the strut, because the packing cannot be tightened properly against the air pressure.

2-373. INSTALLATION OF MAIN ALIGHTING GEAR SHOCK STRUTS. Reverse the removal procedure.

2-374. TAIL GEAR.

2-375. The tail gear consists of a spindle, fork and wheel assembly, an air-oil shock absorber, retracting cylinder, locking mechanism, and a static ground conductor. The tail gear is designed to be swiveled 120 degrees on each side of the aircraft center line, or it may be locked in the trailing position during take-off or alighting runs. The tail gear lock lever is located on the control pedestal in the cockpit.

2-376. REMOVAL OF TAIL GEAR. To remove the tail gear, proceed as follows:

a. Raise the tail of the aircraft (see paragraph 1-68).

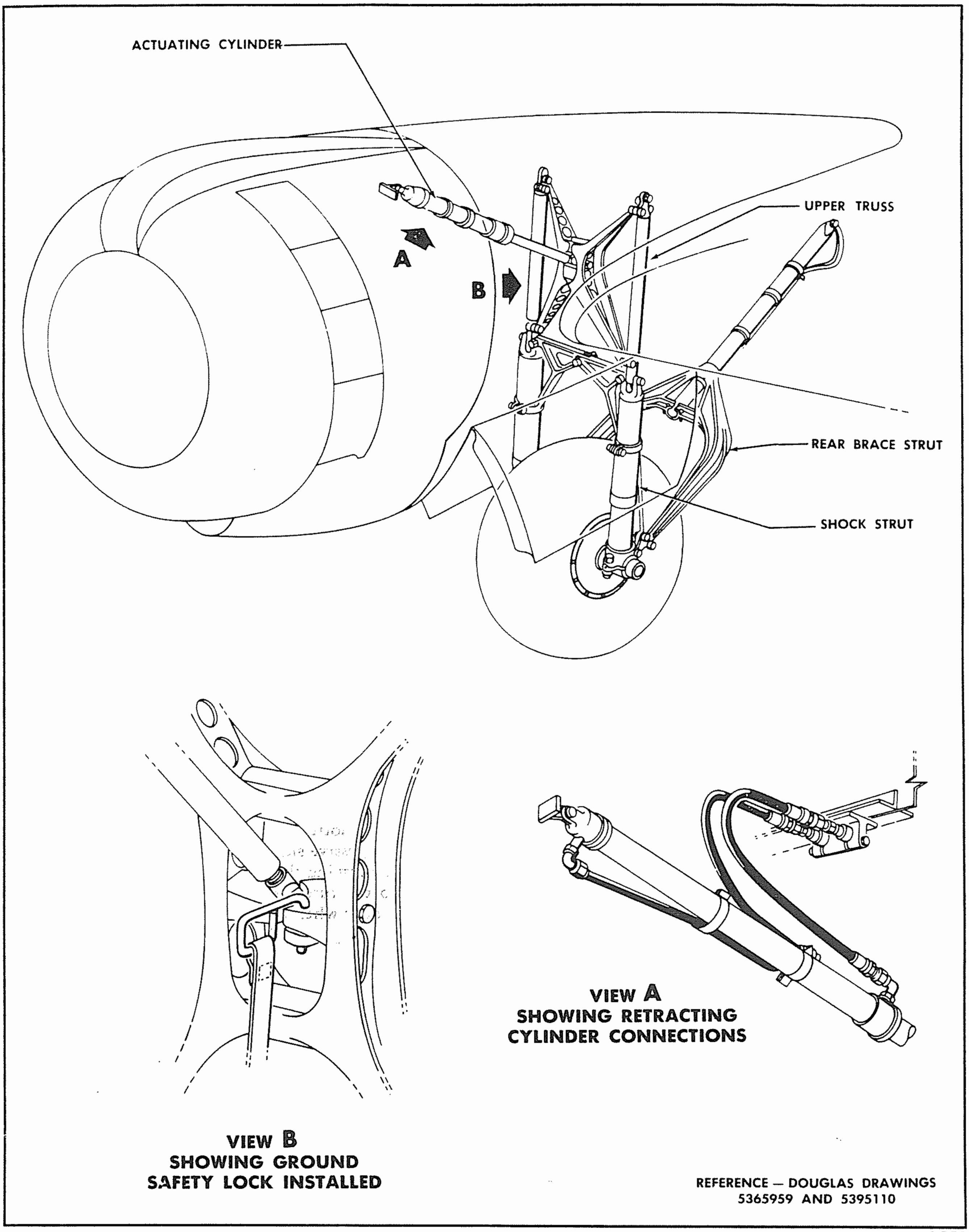


Figure 2-50. Main Alighting Gear

b. Disconnect the tail gear lock cable and spring from within the fuselage, aft of the lavatory compartment.

c. Support the tail wheel assembly with a bar inserted through the hollow link pin in the fork, directly below the shock absorber and tail gear fork attachment, after removing the nut, washers, and bolt from the pin.

d. Remove the fairing from the fork.

e. Remove the nuts at the spindle center bearing, and lower the tail gear out of the aircraft.

2-377. MINOR REPAIR AND REPLACEMENT OF TAIL GEAR. Periodic inspection will determine the general condition and security of all parts. Upper and center anti-friction bearings should be greased at 200- to 300-hour intervals.

2-378. INSTALLATION OF TAIL GEAR. Add preload on upper ball and socket *(see figure 2-51).* Reverse the removal procedure.

2-379. ADJUSTMENT AFTER INSTALLATION OF TAIL GEAR. *(See figure 2-51.)* To adjust the tail gear, proceed as follows:

a. With the tail wheel gear extended and no load on the wheel, the link assembly must stop on the spindle assembly. The center line of the bolt connecting the shock absorber strut and the link must be overcenter $1/16$ ($\pm 1/32$) inch.

b. Adjust the length of the retracting cylinder so that the piston just bottoms, with the centers of the bolt holes of the cylinder trunnion and the cylinder center line in line.

c. Screw the retracting cylinder rod end inward a minimum of $1\frac{1}{2}$ turns and install the bolt.

d. With the tail wheel retracted, adjust the stroke of the retracting cylinder, if necessary, by moving the retracting cylinder rod end the required amount to maintain $\frac{1}{8}$-inch clearance between the link and the spindle bearing housing.

CAUTION

The rod end must not be lengthened beyond its initial adjustment because this could cause bottoming of the piston in the cylinder.

e. Tighten the nut at the upper end of the tail wheel fork to a torque of 135 (± 7) foot-pounds.

f. Safety the turnbuckles.

2-380. TAIL GEAR ACTUATING CYLINDER. *(See figure 2-49.)* The tail gear actuating cylinder is incorporated in the assembly to permit partial retraction of the tail wheel. The cylinder end of the actuating cylinder is bolted to the tail gear fork assembly, while the piston rod end is attached to the link assembly.

2-381. REMOVAL OF TAIL GEAR ACTUATING CYLINDER. To remove the tail gear actuating cylinder, proceed as follows:

a. Support the tail of the aircraft on a jack (see paragraph 1-68).

b. Extend the tail gear.

c. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and drain the hydraulic fluid from the cylinder.

d. Disconnect and cap the flexible hoses at the elbows of both cylinder ports.

e. Remove the trunnion mounting bolt at the cylinder end of the actuating cylinder.

f. Remove the bolt attaching the piston rod end of the actuating cylinder to the tail wheel link assembly, and remove the cylinder.

2-382. INSTALLATION OF TAIL GEAR ACTUATING CYLINDER. Reverse the removal procedure with the following additional step: complete the adjustment procedure as described in paragraph 2-379.

2-383. TEST OF TAIL GEAR ACTUATING CYLINDER. For a complete description of the test procedure for the tail gear actuating cylinder, see paragraph 3-64.

2-384. TAIL GEAR LOCKING MECHANISM. The tail gear lock assembly is cable-operated by a lever located below the throttle quadrant on the control pedestal. The spring-operated lock at the tail wheel is a key and slot mechanism, with the slot contained on the spindle and the key hinged to the tail gear housing assembly. The key is pulled from the slot by the cable assembly that connects to the control knob in the cockpit. A spring re-locks the assembly when the control knob is released. An aluminum-alloy pin secures the slotted fitting to the spindle, and, in the event of an excessive side load on the tail wheel, the pin will shear, allowing the tail gear to move freely before damage to the fuselage structure can result. The lock operates only when the tail gear is in the trailing position.

2-385. REMOVAL OF TAIL GEAR LOCKING MECHANISM. To remove the tail gear locking mechanism, proceed as follows:

a. Remove the spring and cable attachments.

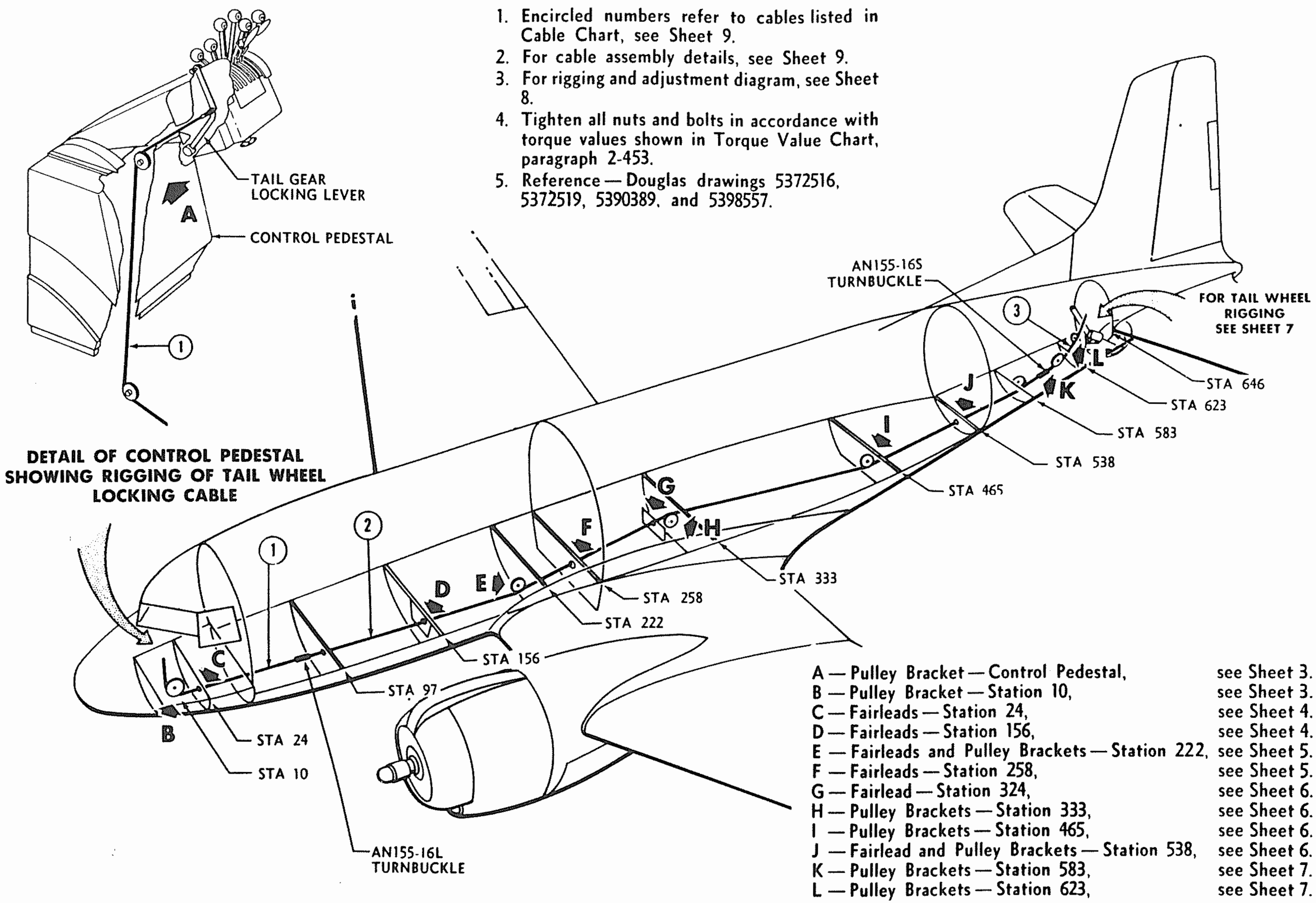
b. Remove the lock hinge nut and bolt, and remove the lock.

2-386. INSTALLATION OF TAIL GEAR LOCKING MECHANISM. Reverse the removal procedure, with the following additional steps:

a. Place the lever on the control pedestal in the LOCK position. Adjust the cable forward turnbuckle so that the threads are buried, and safety it with lockwire.

(Continued on Page 229)

Notes:
1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 9.
2. For cable assembly details, see Sheet 9.
3. For rigging and adjustment diagram, see Sheet 8.
4. Tighten all nuts and bolts in accordance with torque values shown in Torque Value Chart, paragraph 2-453.
5. Reference — Douglas drawings 5372516, 5372519, 5390389, and 5398557.



A — Pulley Bracket — Control Pedestal, see Sheet 3.
B — Pulley Bracket — Station 10, see Sheet 3.
C — Fairleads — Station 24, see Sheet 4.
D — Fairleads — Station 156, see Sheet 4.
E — Fairleads and Pulley Brackets — Station 222, see Sheet 5.
F — Fairleads — Station 258, see Sheet 5.
G — Fairlead — Station 324, see Sheet 6.
H — Pulley Brackets — Station 333, see Sheet 6.
I — Pulley Brackets — Station 465, see Sheet 6.
J — Fairlead and Pulley Brackets — Station 538, see Sheet 6.
K — Pulley Brackets — Station 583, see Sheet 7.
L — Pulley Brackets — Station 623, see Sheet 7.

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Figure 2-51 (Sheet 1 of 9 Sheets). Tail Gear Locking Control System — Key Drawing

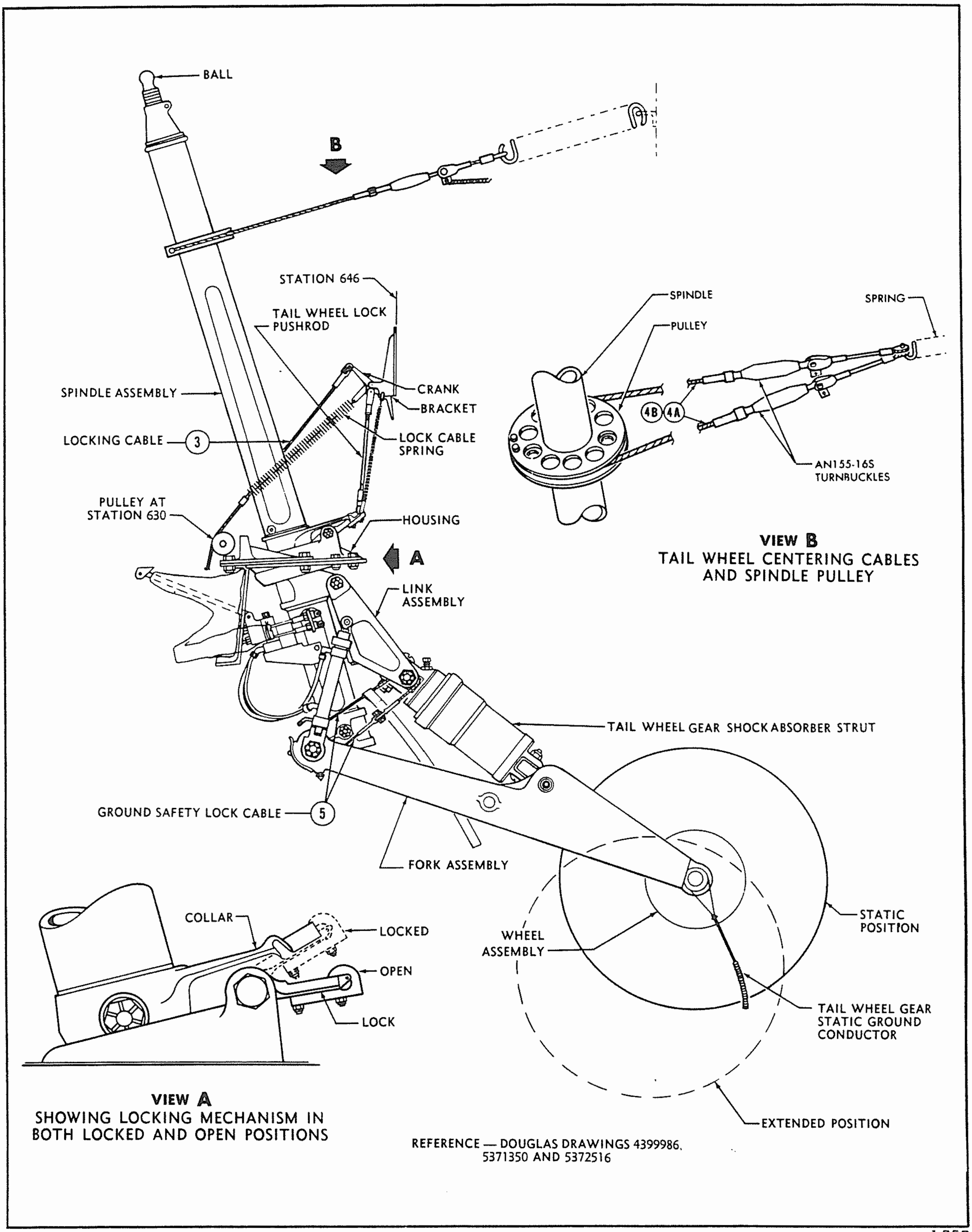
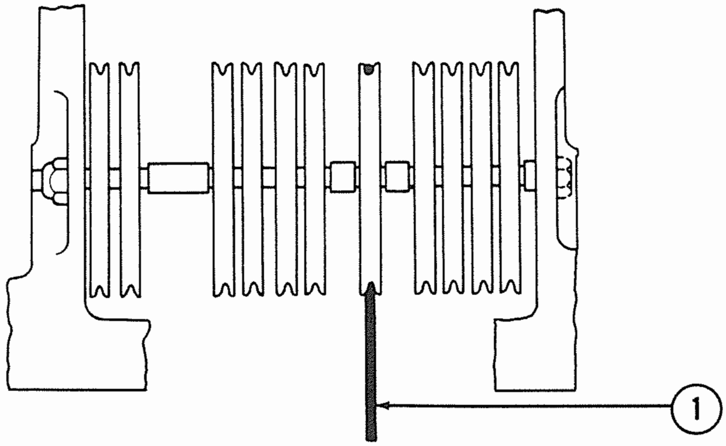
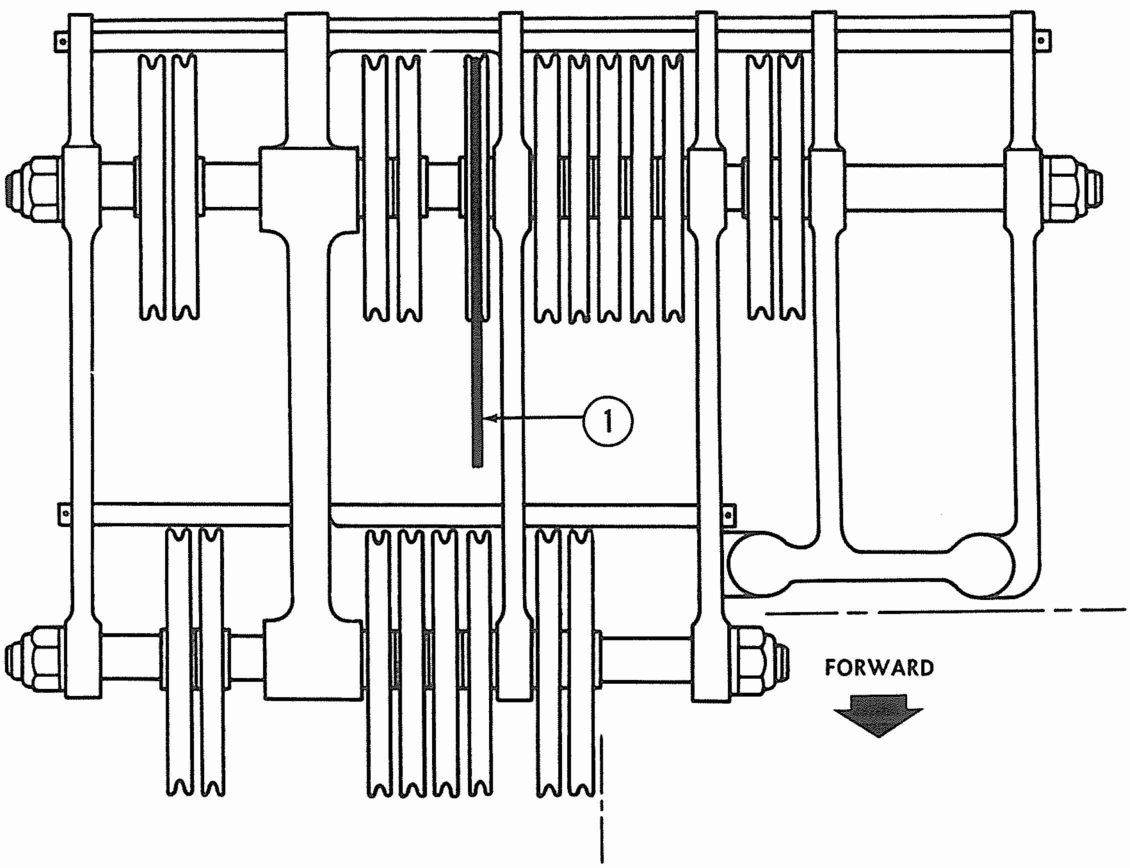


Figure 2-51 (Sheet 2 of 9 Sheets). Tail Gear Locking Control System — Tail Wheel Rigging



VIEW A — UPPER PULLEY BRACKET
IN CONTROL PEDESTAL
(LOOKING FORWARD)



VIEW B — PULLEY BRACKET
AT STATION 10
(BOTTOM VIEW)

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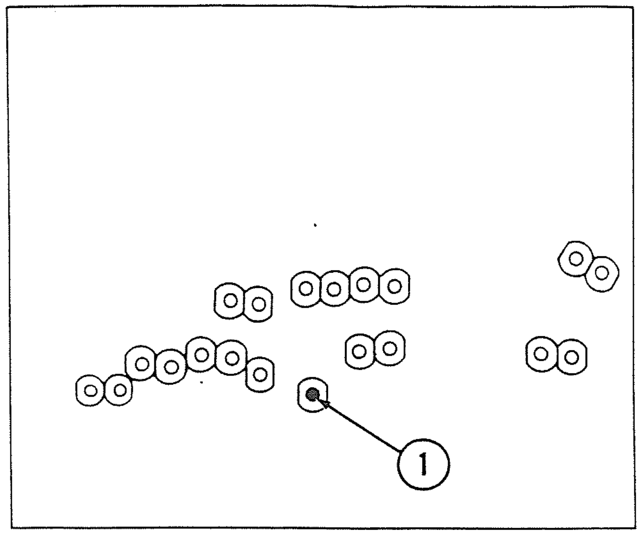
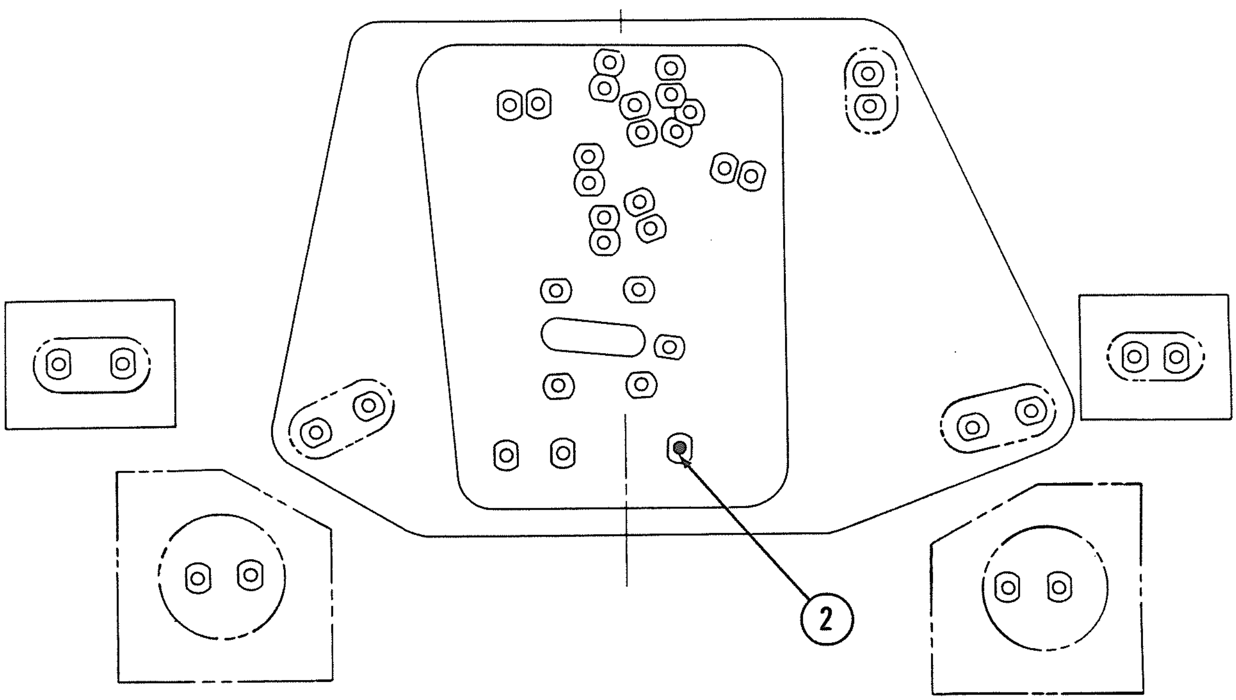
Figure 2-51 (Sheet 3 of 9 Sheets). Tail Gear Locking Control System — Pulley Brackets, Control Pedestal and Station 10

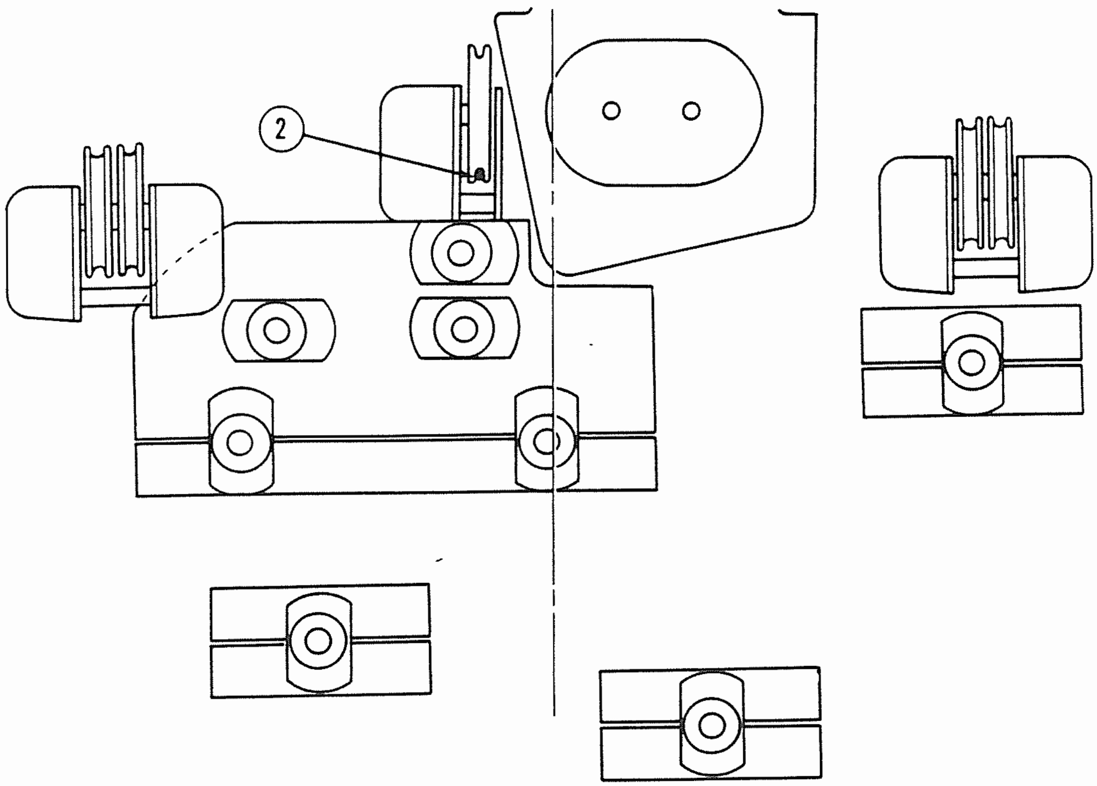


VIEW C — FAIRLEAD
AT STATION 24
(LOOKING FORWARD)

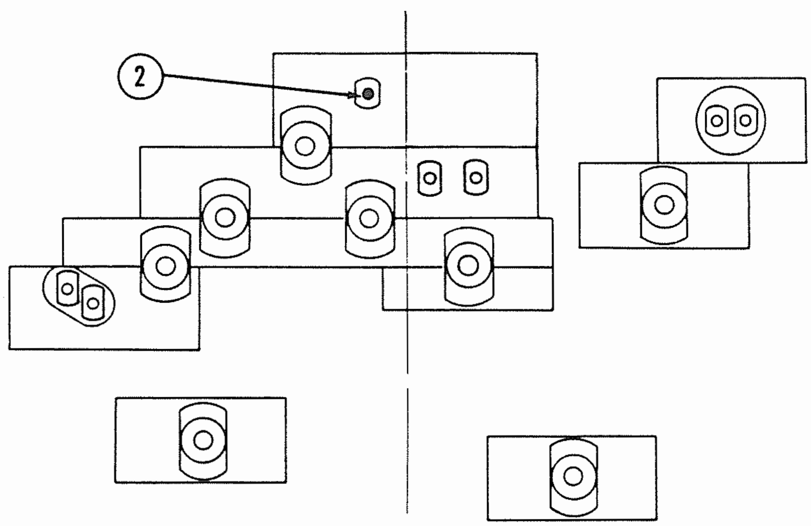


VIEW D — FAIRLEADS
AT STATION 156
(LOOKING FORWARD)

Figure 2-51 (Sheet 4 of 9 Sheets). Tail Gear Locking Control System — Fairleads, Stations 24 and 156



VIEW E — PULLEY BRACKETS AT STATION 222
(LOOKING AFT)



VIEW F — FAIRLEADS AT STATION 258
(LOOKING AFT)

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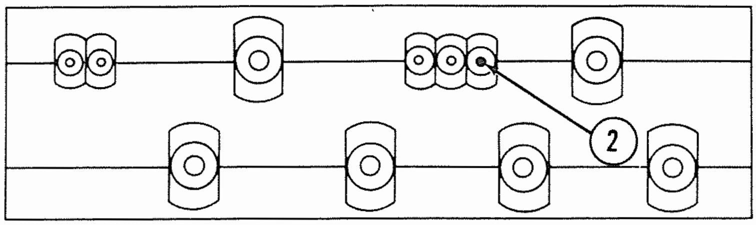
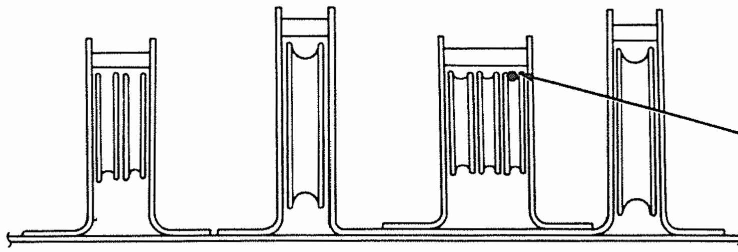
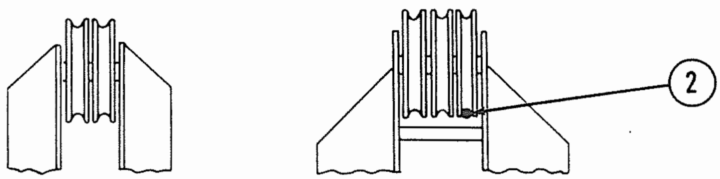
Figure 2-51 (Sheet 5 of 9 Sheets). Tail Gear Locking Control System — Fairleads, Stations 222 and 258



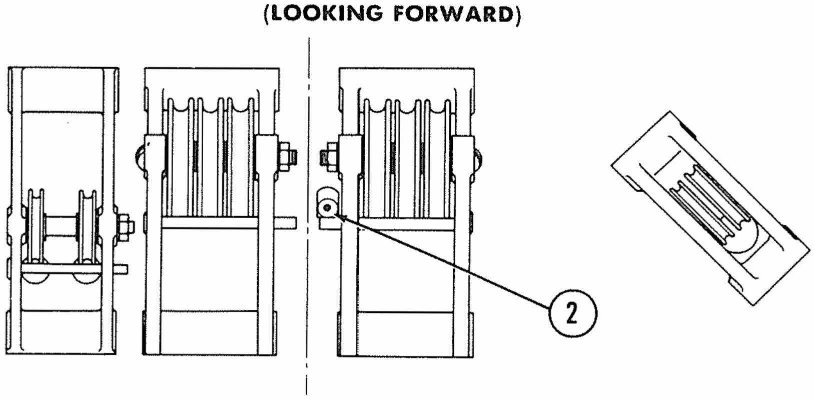
VIEW G — FAIRLEAD
AT STATION 324
(LOOKING FORWARD)



VIEW H — PULLEY BRACKETS
AT STATION 333
(LOOKING FORWARD)



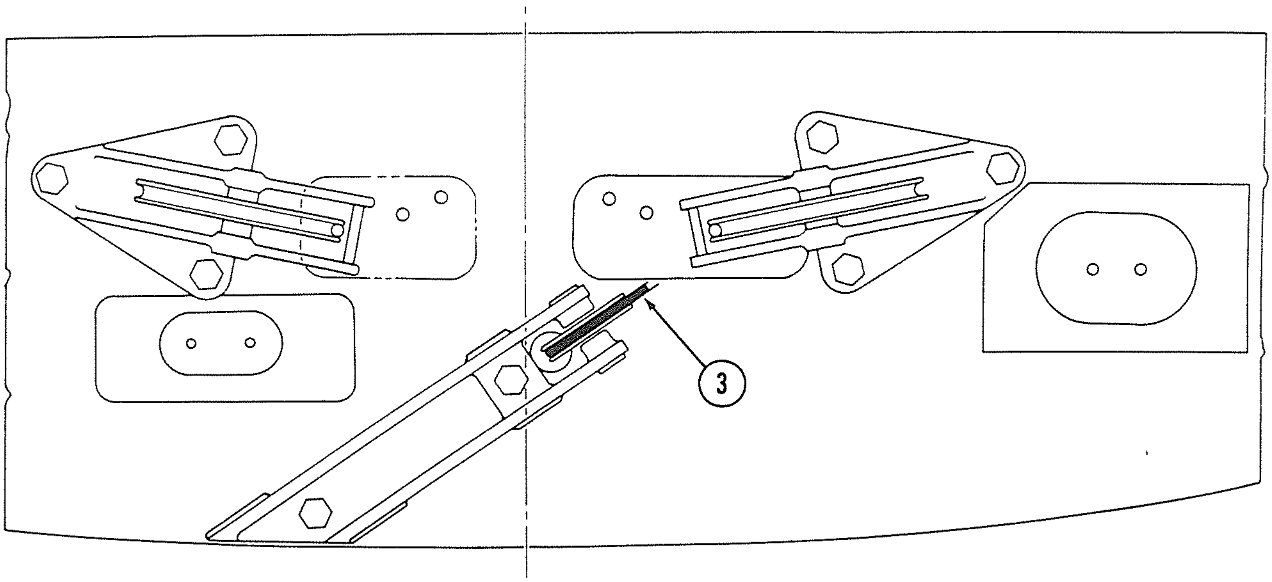
VIEW I — PULLEY BRACKETS
AT STATION 465
(LOOKING FORWARD)



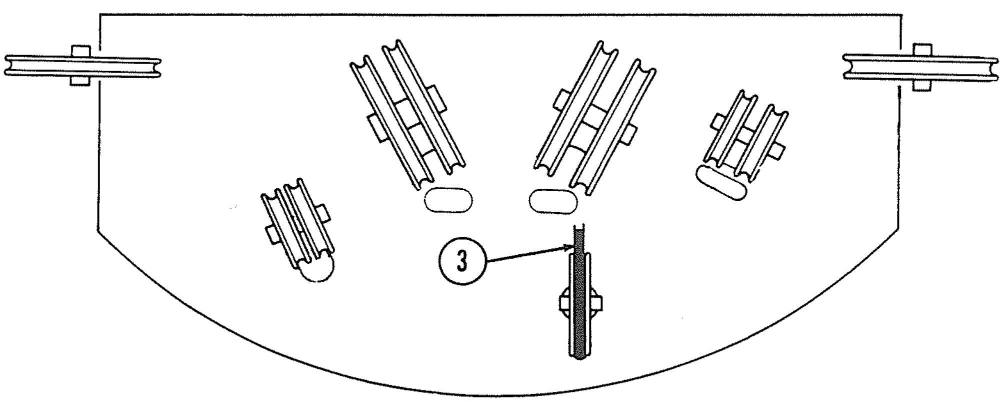
VIEW J — PULLEY BRACKETS AND
FAIRLEAD AT STATION 538
(LOOKING FORWARD)

1,257

Figure 2-51 (Sheet 6 of 9 Sheets). Tail Gear Locking Control System — Fairlead, Station 324; Pulley Brackets, Stations 333 and 465; Pulley Brackets and Fairlead, Station 538



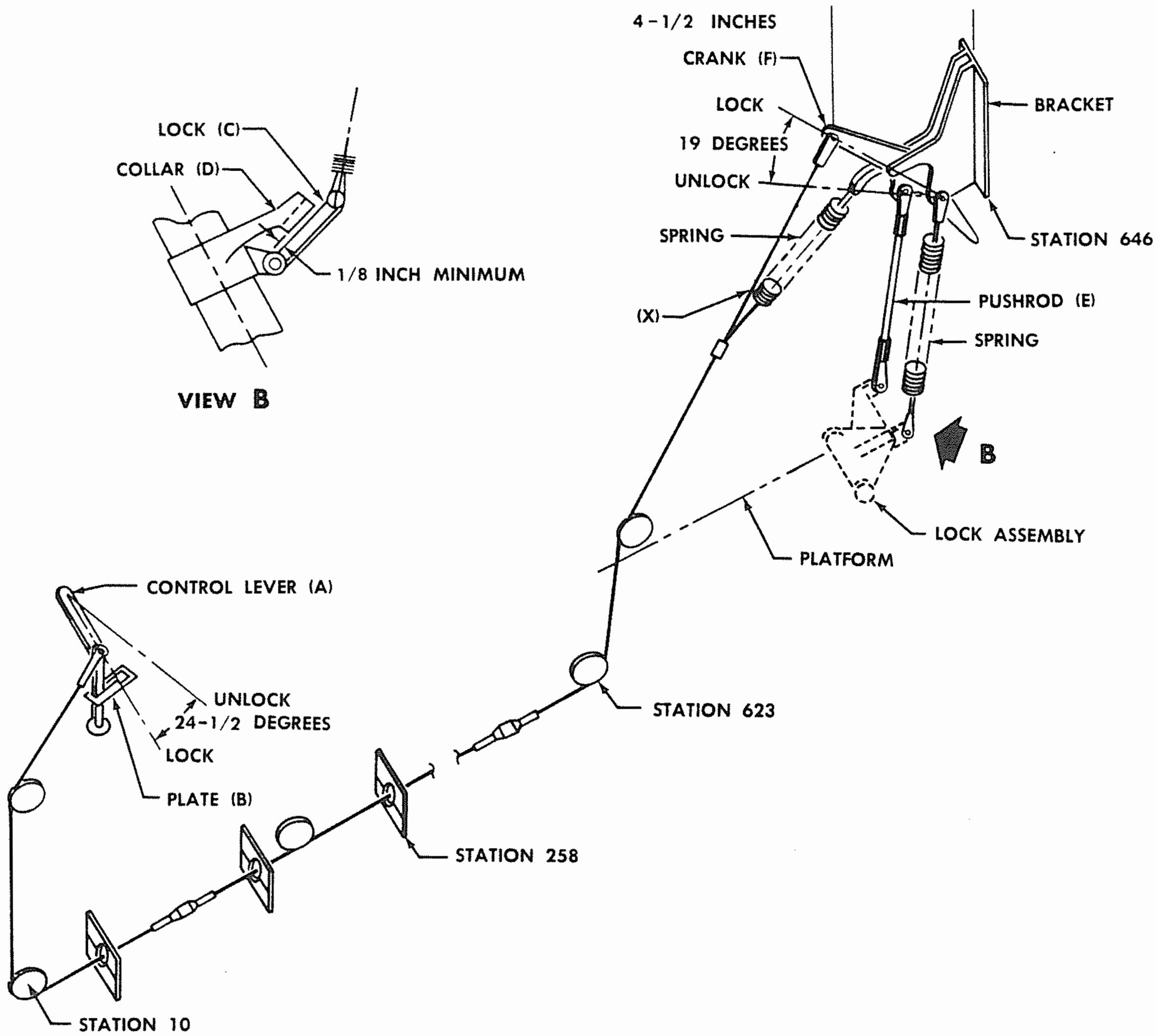
VIEW K — PULLEY BRACKETS
AT STATION 583
(LOOKING FORWARD)



VIEW L — PULLEY BRACKETS
AT STATION 623
(LOOKING FORWARD)

1,258

Figure 2-51 (Sheet 7 of 9 Sheets). Tail Gear Locking Control System — Pulley Brackets, Stations 583 and 623



ADJUSTMENT PROCEDURE

1. Set the cockpit lever (A) in the LOCK position, full forward against bottom of slot in plate (B).
2. With lock (C) bottoming in the notch in collar (D), adjust pushrod (E) to obtain the 4½ inch locating dimension for crank (F) as shown.
3. Adjust turnbuckles uniformly throughout system so that the portion of the aft cable at (X) is just slack.
4. Set the cockpit control lever (A) in the UNLOCK position, full aft in notch in plate (B). The clearance between lock (C) and collar (D) at the latch notch should be ⅛ inch minimum as shown in View B.
5. Safety the turnbuckles.

10,366

Figure 2-51 (Sheet 8 of 9 Sheets). Tail Gear Locking Control System — Adjustment Diagram

TAIL GEAR LOCKING MECHANISM CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)
1	2118996-501	1	A	123	3/32 dia 7x7 flex	1116245	S-2049219-16D-3L		
2	3391412-523	1	B	509 1/2	3/32 dia 7x7 flex	AN669L3 RH	AN669S3 LH		
3	2367227	1	C	(L_1) 58 1/2 (L_2) 12 1/4	3/32 dia 7x7 flex	AN669S3 RH	28-2-G	AN381-3-12	NAS426-3
*4A	4399986-1	2	D	(L_1) 16 1/8 (L_2) 10 1/2	3/32 dia 7x7 flex	AN100-4 (5) 4399986-15	18-2-G (6) AN658-3	AN669S3 LH (7) 18-2-G	AN165-16S (8) AN100-4
†4B	4399986-503	2	D	(L_1) 16 1/8 (L_2) 9 1/4	3/32 dia 7x7 flex	AN100-4 (5) 4399986-15	18-2-G (6) AN658-3	AN669S3 LH (7) 18-2-G	AN165-16S (8) AN100-4
5	2391538	1	E	27	1/4 dia 7x19 flex	AN667-8	2391538-5	AN668-8	

*Aircraft A, B, and 1 through 8 †Aircraft C, D, and 9 through 96

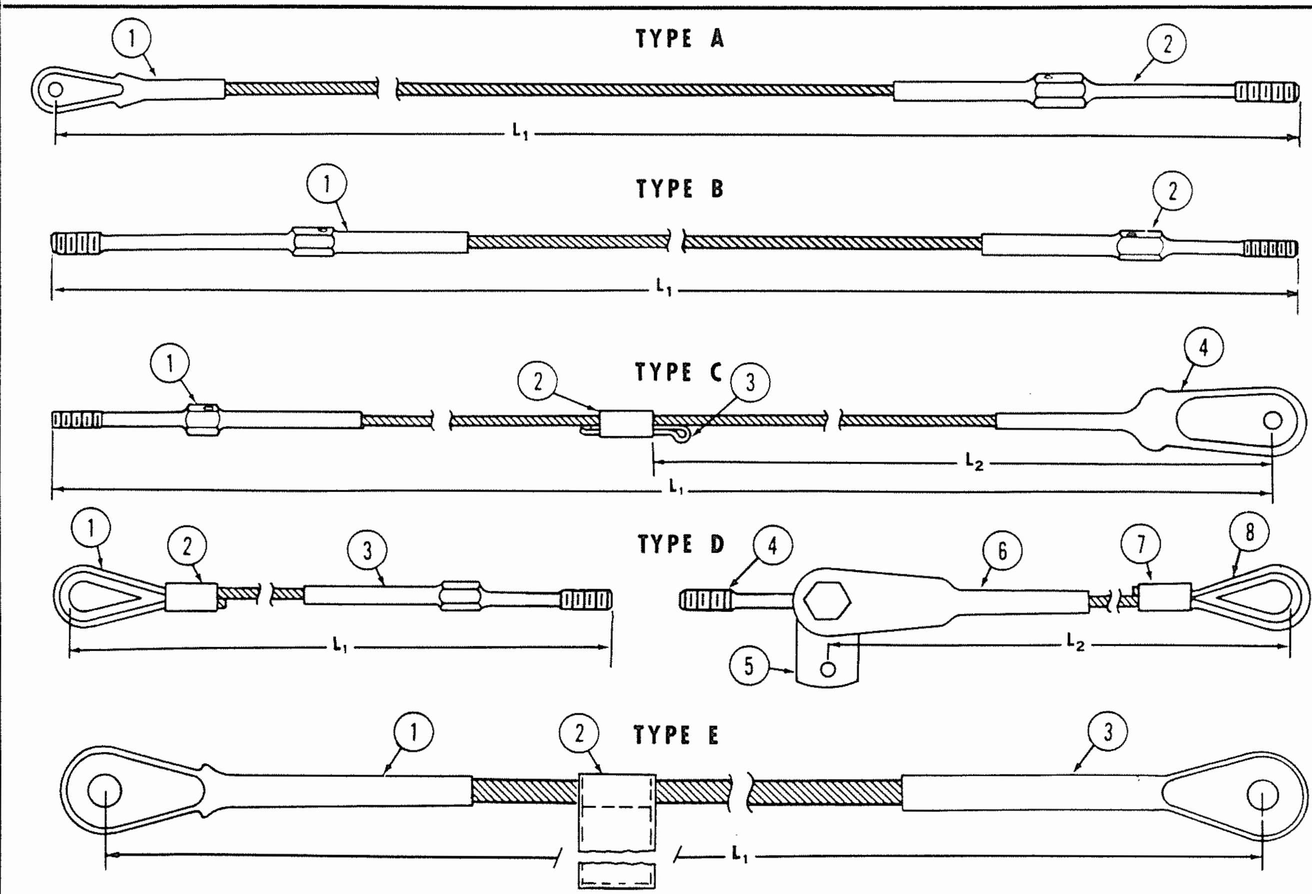


Figure 2-51 (Sheet 9 of 9 Sheets). Tail Gear Locking Control System — Cable Chart and Cable Assemblies

(Continued from Page 219)

b. Adjust the cable aft turnbuckle to assure good bottoming of the lock, consistent with minimum slack in the cable.

c. Place the lever in the UNLOCK position and check for a 1/8-inch minimum clearance between the lock and the mating part on the spindle, and a 1/32-inch minimum clearance between the aft end of the cable and the pulley.

d. Safety the cable aft turnbuckle after the cable is properly adjusted.

2-387. TAIL GEAR LOCKING CONTROLS. *(See figure 2-51.)* The tail gear locking controls consist of a cable system which extends from the control pedestal, in the cockpit, aft through the lower part of the fuselage, to the tail gear locking mechanism. A single-cable system, which employs a return spring installed slightly aft of the tail gear, is used.

2-388. TAIL GEAR LOCKING CONTROL CABLES. *(See figure 2-51.)* The tail gear locking control cables extend from the tail gear locking lever on the control pedestal, aft through the fuselage, to the tail gear locking mechanism crank, located slightly aft of the tail gear.

2-389. REMOVAL OF TAIL GEAR LOCKING CONTROL CABLES.

a. Remove the necessary floor panels and exterior access doors. Remove the access door from the left side of the control pedestal.

b. Disconnect the turnbuckles at stations 97 and 609. Thread the aft end of the intermediate cable and the aft control cable.

c. Disconnect the forward control cable from the lever in the control pedestal, and thread the cable.

d. Remove the necessary guard pins and grommets, and pull the forward and intermediate tail gear locking control cables out of the aircraft through the forward lower fuselage access door at station 97. Pull the cables slowly to prevent chipping of the micarta pulleys. Use two men to perform this operation, one to pull the cables and the other to guide the cable ends through the fairleads and pulley brackets.

e. Disconnect the aft control cable from the tail wheel lock pushrod lever at station 646, and release the return spring.

f. Remove necessary guard pins, and pull the aft cable through the fuselage tail compartment.

2-390. INSTALLATION OF TAIL GEAR LOCKING CONTROL CABLES.

a. Connect the aft tail gear locking cable to the tail wheel lock pushrod lever at station 646, and attach the return spring. Route the control cable forward.

b. Insert the intermediate control cable through the access door at station 97, and route the cable aft, replacing guard pins and grommets as necessary.

c. Insert the forward control cable through the access door at station 97, and route the cable forward through the fuselage, replacing guard pins and grommets as necessary. Connect the forward cable end to the lever in the control pedestal.

d. Connect the cables with turnbuckles at stations 97 and 609. Tension all cables in accordance with the Cable Rigging Tension Chart *(see figure 2-9)*.

e. Adjust the tail gear locking controls (see paragraph 2-391).

f. Safety the turnbuckles.

g. Replace the floor panels and access doors.

2-391. ADJUSTMENT AND OPERATIONAL TEST OF TAIL GEAR LOCKING CONTROLS. *(See figure 2-51.)*

a. Set the tail gear locking control lever, on the control pedestal, in the LOCK position (full forward against the bottom of the slot).

b. With the lock assembly bottoming in the notch in the collar, adjust the tail wheel lock pushrod to obtain a 4½-inch clearance between the end of the crank arm and the base of the bracket on the aircraft structure at station 646.

c. Adjust the cable with the turnbuckles, forward of stations 97 and 623, so that the portion of the cable between the end of the crank arm and the lock cable spring is just slack.

d. Move the control lever on the pedestal to the UNLOCK position. There should be a minimum clearance of 1/8-inch between the tail gear lock and the collar at the latch notch.

e. Safety the turnbuckles.

2-392. TAIL GEAR CENTERING CABLES. *(See figure 2-51.)* The two tail wheel centering cables are attached to the tail wheel spindle pulley and extend aft to a common spring, which is attached at fuselage station 664.

2-393. REMOVAL OF TAIL GEAR CENTERING CABLES.

a. Disconnect the two forward cables from the tail wheel spindle pulley by removing the two bolts and nuts.

b. Disconnect the two aft cables from the common spring.

2-394. INSTALLATION OF TAIL GEAR CENTERING CABLES.

a. Connect the two aft cables to the common spring.

b. Connect the two forward cables to the tail wheel spindle pulley.

c. Safety the turnbuckles.

2-395. ADJUSTMENT OF TAIL GEAR CENTERING CABLES. To adjust the tail gear centering cables, proceed as follows: With the wheel centered to ±1 degree and unlocked, adjust the cable lengths to 12⅞ (±⅛) inches and safety the turnbuckles with safety wire.

2-396. TAIL GEAR GROUND SAFETY LOCK CABLE. The tail gear ground safety lock cable is installed at the tail gear to prevent inadvertent retraction of the tail gear while the aircraft is on the ground. The cable assembly may be removed or installed with one bolt and nut.

2-397. TAIL GEAR STATIC GROUND CONDUCTOR. The tail gear static ground conductor is installed at the tail gear axle.

2-398. ALIGHTING GEAR HYDRAULIC SYSTEM.

2-399. For information on the alighting gear hydraulic system, see paragraph 2-346, preceding.

2-400. MAIN ALIGHTING GEAR ACTUATING CYLINDERS. *(See figures 2-47 and 2-50.)* The main alighting gear actuating cylinders are installed between the nacelle structure aft of the firewall and the alighting gear upper truss. When hydraulic fluid, under pressure, is directed to the upper end of the retracting cylinder, the piston rod extends and forces the alighting gear to extend. When hydraulic fluid, under pressure, is directed to the lower end of the retracting cylinder, the piston rod retracts, pulling the alighting gear up into the nacelle.

2-401. REMOVAL OF MAIN ALIGHTING GEAR ACTUATING CYLINDERS.

a. Install ground safety pins in both alighting gear upper trusses.

b. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

c. Disconnect the flexible hoses leading to the actuating cylinder at the connections located at the top of the nacelle. Cap the exposed ends of the hoses and connectors at the top of the nacelle.

d. The actuating cylinders must be supported from inside the nacelle while the bolt that attaches the cylinder to the nacelle structure is removed. This bolt can be reached through the access door in the top of the nacelle, aft of the firewall.

e. Remove the bolt that attaches the actuating cylinder piston rod hook to the alighting gear upper truss.

f. Move the alighting gear mechanical safety latch control handle to the LATCH RAISED position, and remove the alighting gear actuating cylinder assembly from the aircraft.

2-402. INSTALLATION OF MAIN ALIGHTING GEAR ACTUATING CYLINDERS. Reverse the removal procedure.

2-403. TEST OF MAIN ALIGHTING GEAR ACTUATING CYLINDERS. For a complete description of the test procedure for the main alighting gear actuating cylinders, see paragraph 2-404.

2-404. ADJUSTMENT AFTER INSTALLATION OF MAIN ALIGHTING GEAR ACTUATING CYLINDERS. Retract the alighting gear slowly by operating the hydraulic hand pump. Make a careful check for adequate clearance between the alighting gear and the nacelle structure. Carefully inspect the alighting gear mechanical safety latch (see paragraph 2-357).

2-405. ALIGHTING GEAR CONTROL VALVE. *(See figure 2-47.)* The alighting gear control valve is located on the hydraulic system power panel, immediately above the hydraulic hand pump. The two-position control valve is used to extend or retract the alighting gear.

2-406. REMOVAL OF ALIGHTING GEAR CONTROL VALVE.

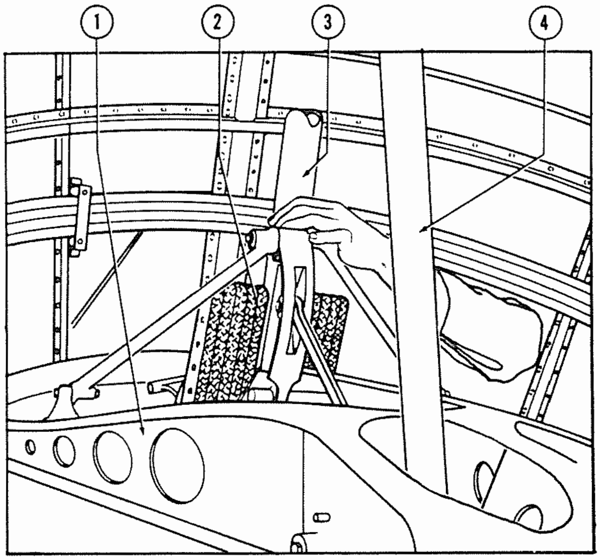
a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

b. Disconnect and cap the four hydraulic hoses leading to the alighting gear control valve elbows.

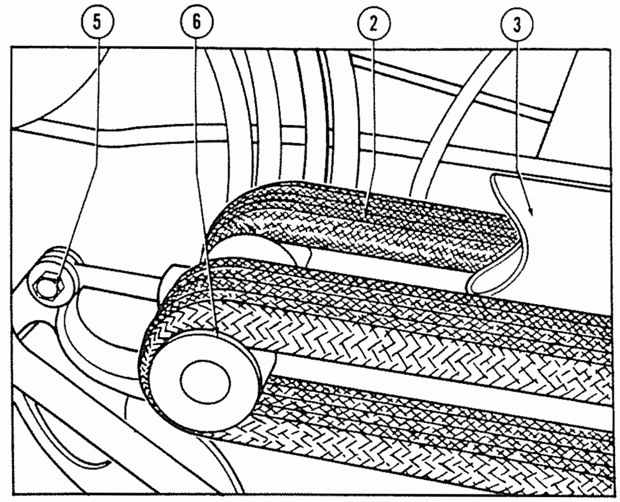
c. Remove the bolt, nut, and cotter pin that attach the rod to the alighting gear safety latch catch assembly on the control valve. Remove the bolts that attach the valve to the panel, and remove the valve.

2-407. INSTALLATION OF ALIGHTING GEAR CONTROL VALVE. Reverse the removal procedure.

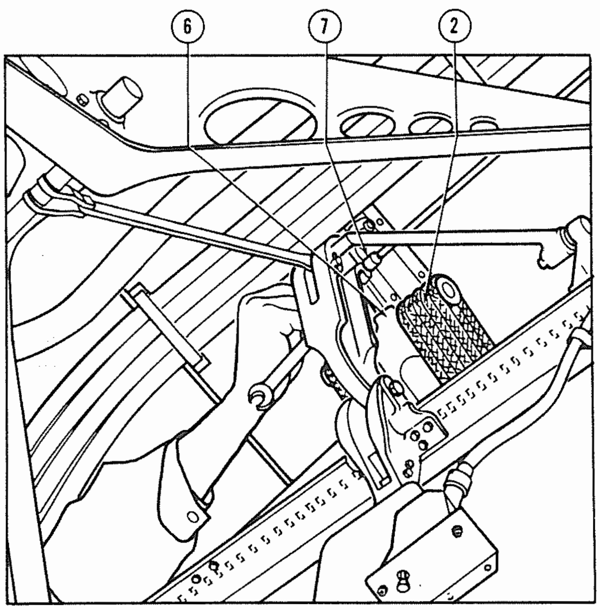
2-408. MAIN ALIGHTING GEAR BUNGEE. *(See figure 2-52.)* The bungee consists of 10 loops of elastic shock cord, mounted on a forward yoke that is connected to the upper truss by a rod, and on an aft yoke mounted in the rear of the nacelle. The bungee cords are extended by the extension of the alighting gear, and assist in balancing the displaced weight. The bungee also assists the actuating cylinder during the alighting gear retraction cycle. The date of manufacture of the bungee can be determined by colored threads, woven



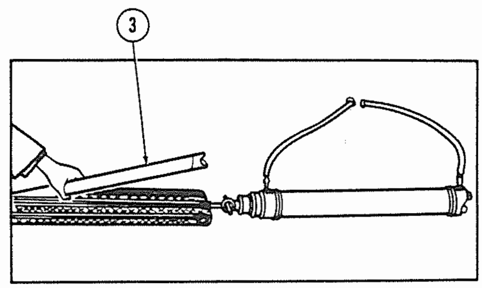
INSTALLING BUNGEE SPREADER BAR



BUNGEE SPREADER BAR IN PLACE BEFORE RETRACTION OF LANDING GEAR

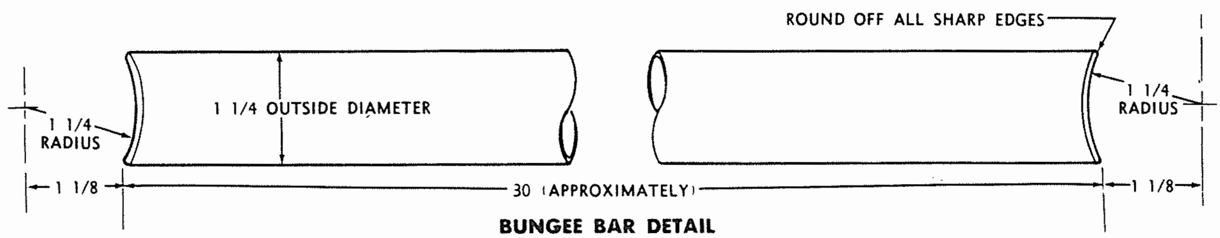


REMOVING BOLT FROM BUNGEE FORWARD YOKE



SPREADING BUNGEE WITH HYDRAULIC JACK

1. Landing Gear Upper Truss
2. Bungee Shock Cords
3. Bungee Spreader Bar
4. Landing Gear Actuating Cylinder Piston Rod
5. Bungee Forward Yoke Attaching Bolt
6. Bungee Forward Yoke
7. Socket Wrench



BUNGEE BAR DETAIL

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Figure 2-52. Bungee Spreader

into the outer cover; two threads for a year and one for a quarter, as follows:

Years		*Quarters*	
1949	yellow	January-March	red
1950	black	April-June	blue
1951	green	July-September	green
1952	red	October-December	yellow
1953	blue	(repeat cycle)	

Note

Do not use bungee elastic shock cords in critical aircraft structures longer than two years.

2-409. REMOVAL OF MAIN ALIGHTING GEAR BUNGEE. To remove the main alighting gear bungee, proceed as follows:

a. Fabricate a bungee spreader bar, if one is not available, by cutting a round notch in the end of a one-inch tube or pipe *(see figure 2-52).*

b. Support the aircraft on wing jacks (see paragraph 1-64).

c. With the alighting gear extended, place one end of the spreader bar on the center of the bungee rear yoke. The bungee spreader bar may be applied through the access doors in the upper surface of the nacelle *(see figure 2-52).*

d. Reduce the hydraulic system pressure to zero by operating the wing flaps. Move the safety latch control handle to the LATCH RAISED position, and the alighting gear control valve handle to the UP position.

e. Retract the alighting gear slowly, by operating the hydraulic hand pump, until the forward end of the spreader bar engages the bungee forward yoke, thus relieving the pull on the yokes *(see figure 2-52).*

f. Remove the bolts that attach the bungee yokes, and remove the spreader bar, shock cord, and fittings as a unit *(see figure 2-52).* Lower the alighting gear slowly or support it from the ground, as the alighting gear will fall with the bungee removed.

g. Use a hydraulic jack or other suitable equipment to stretch the shock cords sufficiently to free the spreader bar *(see figure 2-52).*

h. Remove the spreader bar.

i. Relieve the tension on the shock cords. The bungee yokes and fittings may then be removed from the shock cords.

2-410. MINOR REPAIR AND REPLACEMENT OF MAIN ALIGHTING GEAR BUNGEE. Visually inspect the bungee shock cords at each prescribed inspection period, and replace whenever necessary. Any bungee shock cord showing marked irregularity in diameter is considered defective. Check the dating colors on the bungee shock cord and replace after two years of service.

2-411. INSTALLATION OF MAIN ALIGHTING GEAR BUNGEE. Reverse the removal procedure.

2-412. MAIN ALIGHTING GEAR DOORS AND ACTUATING MECHANISM. The main alighting gear, when retracted, is completely enclosed by mechanically actuated, hinged doors. The mechanism consists of two links, each connecting one of the doors to the rear brace strut. A force exerted on the links, during extension or retraction of the alighting gear, will cause the doors to open or close.

2-413. REMOVAL OF MAIN ALIGHTING GEAR DOORS AND ACTUATING MECHANISM. To remove the main alighting gear doors and actuating mechanism, proceed as follows:

a. Remove the bolts at the ends of the linkage, and remove the linkage.

b. Remove the pin from the piano-type hinge, and remove the door.

2-414. INSTALLATION OF MAIN ALIGHTING GEAR DOORS AND ACTUATING MECHANISM. Reverse the removal procedure.

2-415. ADJUSTMENT OF MAIN ALIGHTING GEAR DOORS AND ACTUATING MECHANISM. To adjust the main alighting gear doors and actuating mechanism, proceed as follows:

a. Support the aircraft on wing jacks (see paragraph 1-64).

b. Disconnect one door for access to permit adjustments on linkage for opposite door.

c. Raise the alighting gear by operating the hydraulic hand pump, allowing the eyebolts on the alighting gear bracket to be free to swivel.

d. With the doors fully closed, tighten the nuts on the eyebolts.

2-416. BRAKE SYSTEMS.

2-417. DESCRIPTION. The brake system can be operated by either the hydraulic system, which is the system normally used, or by the emergency air brake system, which can be used in the event of hydraulic system failure. A shuttle valve, installed at the brake assembly, directs either hydraulic pressure or air pressure to the brake, depending upon which system is being used to apply the brakes.

2-418. MAIN ALIGHTING GEAR HYDRAULIC BRAKES. *(See figures 2-53 and 2-54.)* The main alighting gear brake assembly consists of a single-disc, self-adjusting spot brake, which is operated by main system hydraulic pressure controlled by a valve connected

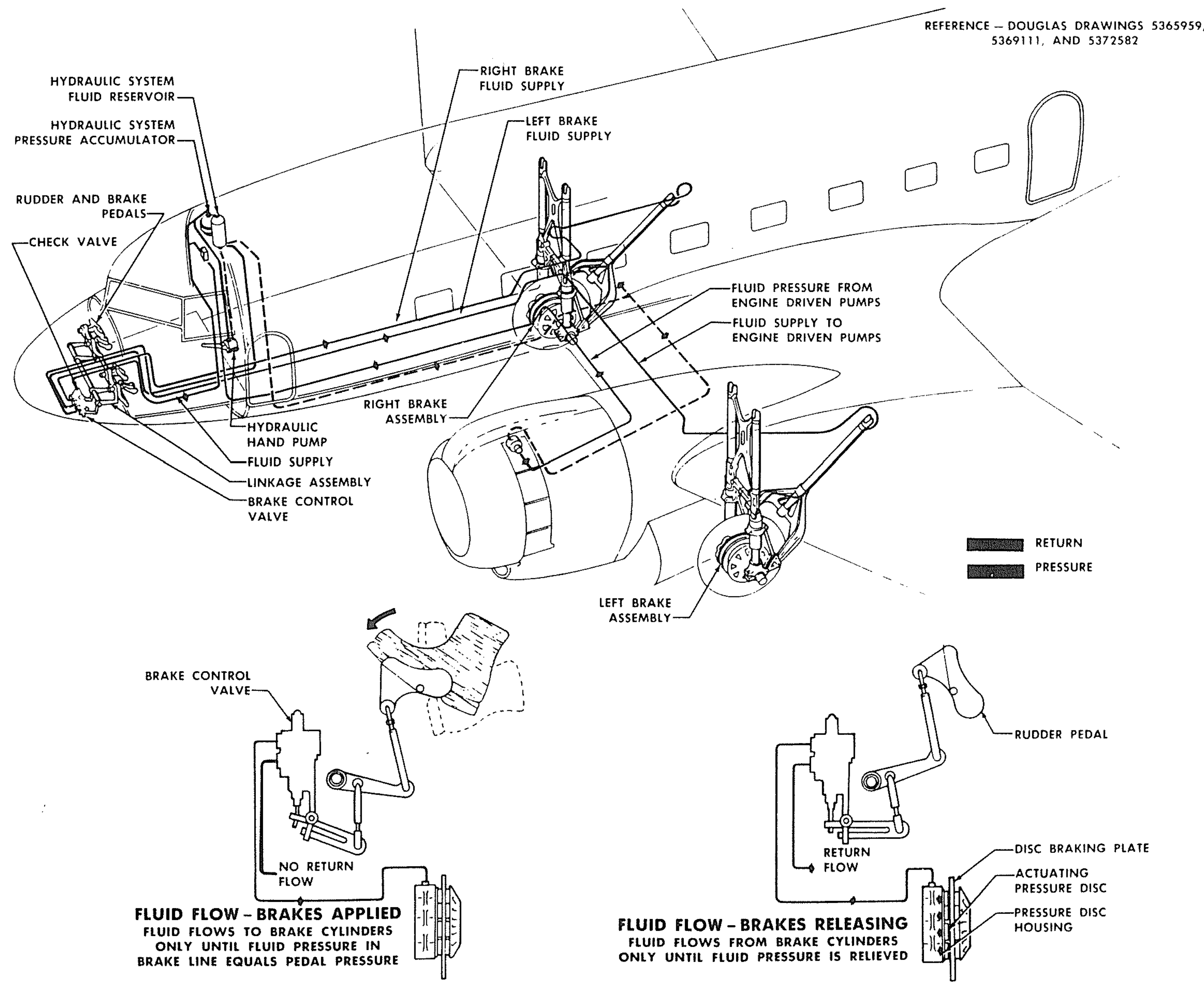

Figure 2-53. Brake System

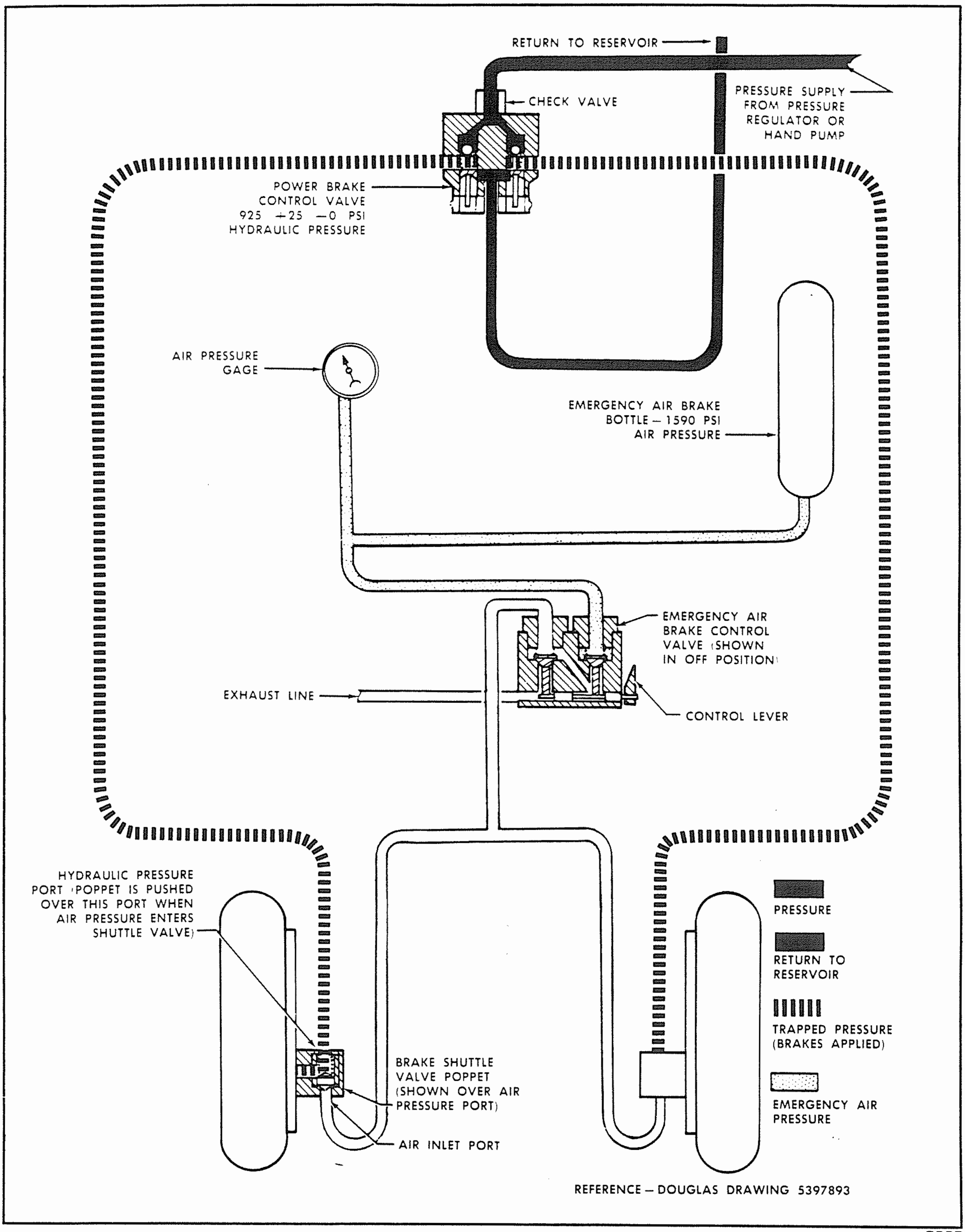

Figure 2-54. Main Alighting Gear Brakes

through linkage to the rudder pedals. The brake assembly incorporates four sections, each consisting of two flat, opposed sections of brake lining, separated by an annular disc, which is keyed to rotate with the wheel assembly. Braking action is accomplished by applying hydraulic pressure, which forces the two opposed spot linings against the disc. The linings are held within the recesses of an anchor bracket, which also contains a groove for the annular disc. The annular disc floats laterally on the wheel drive keys. When braking pressure is applied, this floating action compensates for the movement of the disc, so that only the outer brake lining moves toward the disc, and the inner lining remains stationary. As the outer lining presses against it, the floating disc compensates by moving until it contacts the inner lining. The disc then receives equal clamping action on both sides as pressure is increased. The brake requires no adjustment during the entire life of the lining, since the brake lining clearance remains constant and the brake pedal remains in the same position, regardless of wear on the brake lining. As the brake linings wear, the pressure applied to the brake will cause the return spring stationary plate to close against the return spring movable plate and pull the adjusting pin through the adjusting pin packing nut and packing. This causes the entire piston assembly to move toward the brake disc and compensate for any degree of lining wear. Upon releasing the brake, the piston will again return to the designed clearance. When the adjusting pin has receded to a depth of 1/4 inch, measured from the shoulder of the torque nut with brakes applied or parked, the brakes must be relined within a maximum of 25 more landings *(see figure 2-53)*.

2-419. REMOVAL OF MAIN ALIGHTING GEAR HYDRAULIC BRAKES.

a. Support the aircraft on wing jacks (see paragraph 1-65).

b. Reduce the hydraulic system pressure to zero by operating the wing flaps.

c. Disconnect and cap the hydraulic hoses and emergency air brake pipes at the shuttle valve.

d. Remove the axle caps.

e. Remove the wheel and axle assembly.

f. Remove the four bolts attaching the spot brake assembly to the axle.

g. Remove the brake assembly and the brake disc.

2-420. INSTALLATION OF MAIN ALIGHTING GEAR HYDRAULIC BRAKES. Reverse the removal procedure. Bleed the brake hoses before attaching them at the shuttle valve.

2-421. ADJUSTMENT OF MAIN ALIGHTING GEAR HYDRAULIC BRAKES. If, for any reason, it should become necessary to check the brake clearances, proceed as follows:

a. With the brakes OFF, insert a 0.015-inch shim between the lining and the disc at all spots. (It may be necessary to loosen the brake self-adjusting pin friction nut and pull the pin out.)

b. With the shim still in place, tighten the brake self-adjusting pin friction nut to 25 foot-pounds torque.

c. Remove the shim and apply full brake pressure.

d. Release the brakes and check for a clearance of 0.015 inch.

2-422. POWER BRAKE CONTROL VALVE. The power brake control valve is mounted on a bracket in the nose section of the fuselage *(see figure 2-55)*. The power brake control valve provides a means of supplying hydraulic fluid to the wheel brakes at a controlled pressure and quantity with the same "feel" of sensitivity as that obtained with manually operated brakes. The application of toe pressure to the rudder brake pedal is transmitted to the "tuning fork" arm of the brake valve, causing the brake valve piston and pin to move upwards, unseating the inlet valve ball. Hydraulic fluid, under pressure, then flows into the piston chamber and into the brake-operating pipe, actuating the brake pistons and brake shoes. The hydraulic fluid continues to flow to the brake operating pipe until the force exerted by fluid pressure in the brake operating pipe equals the force applied to the rudder-brake pedals. The fluid pressure then forces the piston down and the inlet valve ball is seated. Piston thrust is taken up by deflection of the "tuning fork" so that the fluid pressure is felt at the rudder-brake pedals. When pressure on the rudder pedal is released, the piston moves down and a fixed pin in the piston unseats the outlet valve. The excess fluid in the brake-operating pipe flows back into the return pipe to the reservoir.

2-423. REMOVAL OF POWER BRAKE CONTROL VALVE.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and drain the hydraulic fluid from the system.

b. Disconnect and cap the hydraulic pipes leading to the brake control valve.

c. Disconnect the rudder-brake pedal operating linkage from the "tuning fork" arms of the brake control valve.

d. Remove the bolts that attach the brake control valve to the bracket and remove the valve.

2-424. INSTALLATION OF POWER BRAKE CONTROL VALVE. Reverse the removal procedure.

2-425. TEST OF POWER BRAKE CONTROL VALVE. The power brake control valve is tested in conjunction with the brake system test.

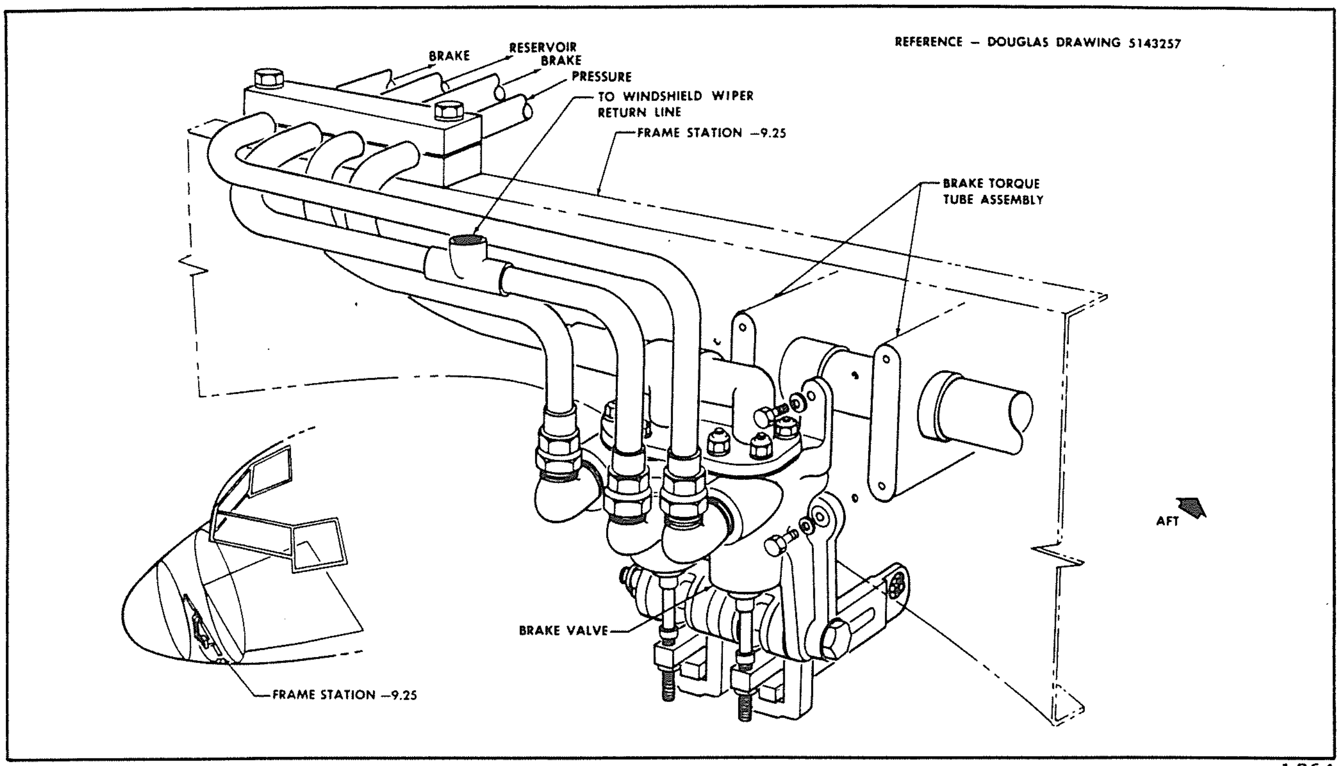

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Figure 2-55. Power Brake Control Valve

2-426. BRAKE LINE SHUTTLE VALVES. The brake line shuttle valves (Douglas part No. 4392008-502) are located on the brake assemblies of each of the main gear wheels *(see figure 2-53)*. The shuttle valve forms a junction point for the hydraulic and emergency air pressure pipes at the brake assembly on the main gear wheels and allows either hydraulic or air pressure to be conducted to the brake.

2-427. REMOVAL OF BRAKE LINE SHUTTLE VALVES.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and drain the hydraulic fluid from the system.

Note

Before removing the valve, make certain that the parking brakes are released and that the emergency air brake valve is closed.

b. Disconnect the hydraulic and emergency air pipes at the valve to be removed; cap the pipes and the open ports of the valve.

c. Remove the bolt attaching the valve to the brake housing and remove the valve.

2-428. INSTALLATION OF BRAKE LINE SHUTTLE VALVES. Reverse the removal procedure.

2-429. TEST OF BRAKE LINE SHUTTLE VALVES. The shuttle valves are tested in conjunction with the brake system test.

2-430. PARKING BRAKES. Setting the main wheel brakes for parking is accomplished by means of the manually operated plunger installed on the lower aft face of the control pedestal. The parking brake mechanism is a locking device that holds the rudder brake pedals in the depressed or operated position. To set the parking brake, depress the rudder-brake pedals fully and pull out the parking brake control knob. The hydraulic system pressure gage should indicate a minimum of 500 psi for satisfactory parking brake operation. To release the parking brake, apply toe pressure to the rudder-brake pedals *(see figures 2-53 and 2-56)*.

Note

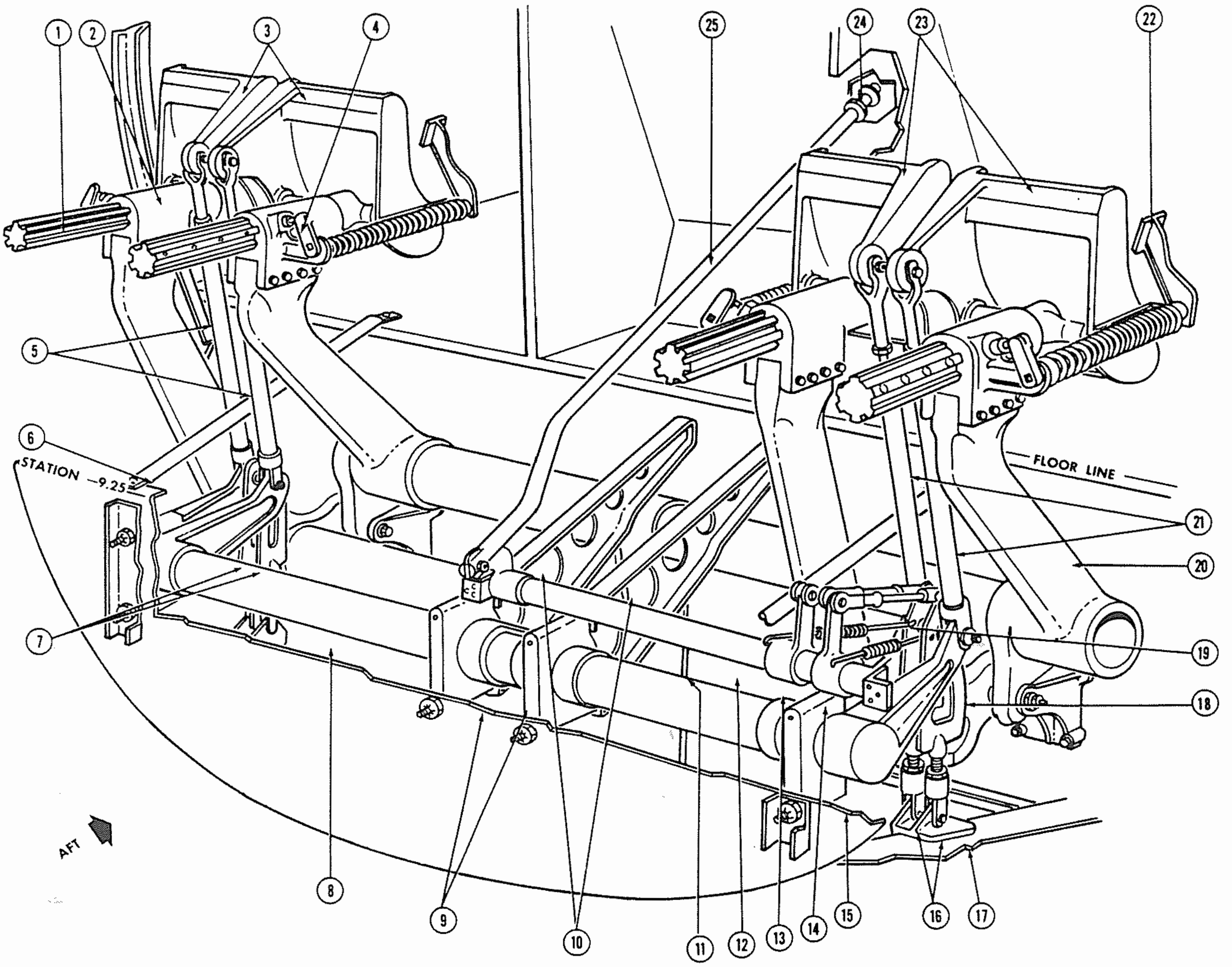
In releasing the parking brake, it is only necessary to depress the pedals fully. This will allow the spring to return the parking brake lock to the off position.

2-431. REMOVAL OF PARKING BRAKE MECHANISM.

a. Release the rudder-brake pedal springs.

b. Disconnect and remove the brake rod assemblies.

c. Remove the parking brake pedal stops.



1. Spline
2. Rudder Pedal Slide Tube Support
3. RH Rest Assembly
4. Pedal Adjusting Linkage
5. Brake Rods
6. Bulkhead Support
7. RH Stop Assembly
8. Brake Torque Tube Assembly
9. Center Bearings
10. Braces
11. Horn
12. Shaft Assembly
13. RH Arm
14. LH Arm
15. Outer Bearing
16. LH Stop Assembly
17. Support
18. Brake Torque Tube Lock Stop Assembly
19. Rod Assembly
20. Rudder Pedal Assembly
21. Brake Rods
22. Lever
23. LH Rest Assembly
24. Handle
25. Rod

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Figure 2-56. Parking Brake Mechanism

2-432. INSTALLATION OF PARKING BRAKE MECHANISM. Reverse the removal procedure.

2-433. ADJUSTMENT OF PARKING BRAKES. For a complete description of the power brake control valve, see paragraph 2-422.

2-434. AIR BRAKE SYSTEM. If the brake hydraulic system should become inoperative, the emergency air brake system substitutes air pressure for hydraulic pressure to permit operation of the brakes. The system consists of an air pressure tank and an air pressure control valve. The initial air pressure available from the pressure tank is 1590 psi *(see figure 2-54).*

2-435. BRAKE SYSTEM TESTS AND ADJUSTMENTS.

2-436. AIR BRAKE SYSTEM PRESSURE TEST.

a. With the air cylinder inflated to 1590 psi and the air brake valve off, the system must show no leakage in a 24-hour period (allow for temperature variation during the test period).

b. Disconnect and plug the air brake hoses at the shuttle valves. With the air cylinder inflated to 1590 psi, turn on the air brake valve. Check the pipes and fittings for leakage. The pressure may drop 5 psi in 15 minutes from the initial reading after the valve is opened.

2-437. BRAKE PRESSURE ADJUSTMENT.

a. Attach 1500-psi gages at the bleed ports of the brakes.

b. Adjust the off-setting screws in both brake levers to give the correct pressure at the gages.

c. Check "brakes off" by depressing both brake pedals slowly until the gages show approximately 200 psi. The wheels must not be free to rotate. Release the brake pedals slowly. The wheels should be free to rotate within approximately five seconds (the gages should drop to approximately 15 psi).

d. With 1100 psi in the hydraulic system, set the parking brakes. Relieve the pressure in the remainder of the hydraulic system by operating the wing flaps. An initial pressure drop indication of 50 psi to 150 psi at the gages is allowable. Subsequent pressure drops must not exceed 50 psi in 15 minutes. Release the brakes and remove the gages.

2-438. BRAKE CLEARANCE ADJUSTMENT.

a. With the brakes off, insert 0.015-inch shims between the lining and the disc at all spots. (It may be necessary to loosen the brake self-adjusting pin friction nut and pull the pin out.)

b. With the shims still in place, tighten the brake self-adjusting pin friction nut to 25 foot-pounds of torque.

c. Remove the shims and apply the brakes fully. Release the brakes and check for 0.015-inch clearance.

d. If the friction pins are more than 1/4 inch deep, the spot assemblies should be replaced.

2-439. AIR BRAKE CONTROL VALVE. The main alighting gear air brake control valve (Douglas Part No. 3369907) is installed to the left of the aircraft center line, just below the windshield Vee.

2-440. REMOVAL OF AIR BRAKE CONTROL VALVE.

a. Relieve the system air pressure.

b. Scribe a mark from the shaft to the valve control lever so that the lever can be replaced in the same splines on installation; remove the lever.

c. Detach the air pipes at the valve; cap the pipes and plug the ports in the valve.

d. Remove the valve attaching bolts and nuts, and remove the valve.

2-441. INSTALLATION OF AIR BRAKE CONTROL VALVE. Reverse the removal procedure. Charge the air cylinder to 1590-psi pressure (see paragraph 1-112).

2-442. AIR BRAKE CYLINDER. The main alighting gear air brake cylinder (Part No. 62238, Walter Kidde & Co. Inc., New York, N. Y.) is located directly behind the co-pilot's seat *(see figures 1-41 and 2-57).*

2-443. REMOVAL OF AIR BRAKE CYLINDER.

a. Relieve the system air pressure.

b. Disconnect and cap the pressure pipe.

c. Remove the bolts securing the cylinder mounting straps and remove the bottle.

2-444. INSTALLATION OF AIR BRAKE CYLINDER. Reverse the removal procedure. Charge the cylinder to 1590-psi pressure (see paragraph 1-112).

2-445. WHEELS, TIRES, AND TUBES.

2-446. MAIN ALIGHTING GEAR WHEELS. *(See figure 2-41.)* The interchangeable main alighting gear wheels are 17.00 x 16, Type III, magnesium-alloy wheels. Each unit contains a hydraulically operated disc-type brake.

2-447. REMOVAL OF MAIN ALIGHTING GEAR WHEELS. Support the aircraft on wing jacks (see paragraph 1-64), and proceed as follows:

a. Reduce the hydraulic system pressure to zero by operating the wing flaps.

b. Retract the alighting gear approximately six inches by operating the hydraulic hand pump.

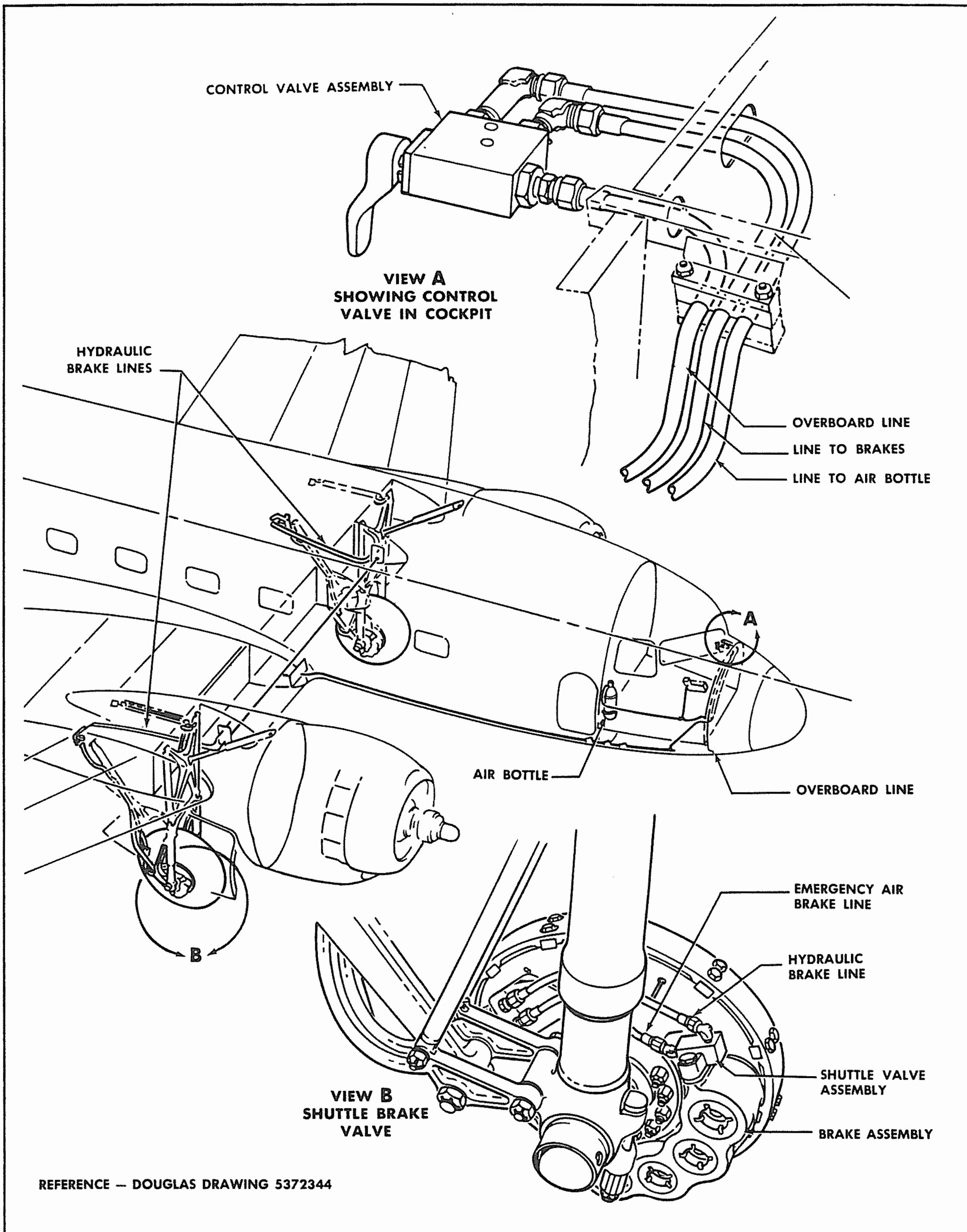

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Figure 2-57. Air Brake System

c. Disconnect and cap the hydraulic pipes and the emergency air brake pipes at the shuttle valve.

d. Remove the bolts and drop the axle cap.

e. Loosen one axle nut and remove the axle and wheel assembly.

2-448. MINOR REPAIR AND REPLACEMENT OF MAIN ALIGHTING GEAR WHEELS. Wheels should be inspected daily for bent or distorted rims. Remove burrs and indentations from the wheel castings, using a flat file. Check the condition of the paint on the wheels and the surface of the wheel for evidence of corrosion. If the paint has deteriorated and the wheel surface is exposed, repaint the wheel as follows:

a. Remove the old paint with acetone (Federal Specification No. O-A-51A), or paint remover.

b. Protect the bearing cups, drums, and other friction surfaces, and the tires by masking.

c. Apply one coat of zinc chromate primer (either Specification No. AN-E-3 or mixed in a ratio of from 1½ to 2 pounds of aluminum-bronze powder per gallon of varnish).

d. Clean any corroded area carefully and determine the extent of corrosion. If the corrosion extends deeper than the surface, or if the part is cracked or worn, replace the part.

2-449. INSTALLATION OF MAIN ALIGHTING GEAR WHEELS. Reverse the removal procedure, taking care to bleed the brake pipe before attaching the hoses at the brake shuttle valve. Set the wheel bearing up to 120 foot-pounds torque, then back off until the nut is free. Readjust the nut to the point of no play and no preload in the bearing, and then tighten to the next setscrew hole.

2-450. MAIN ALIGHTING GEAR TIRE AND TUBE. *(See figure 2-50.)* The main alighting gear wheels mount a 12-ply rating, 17.00 x 16 tire and plain tube on each wheel.

2-451. REMOVAL OF MAIN ALIGHTING GEAR TIRE AND TUBE. Remove the wheel from the axle, and proceed as follows:

a. Remove the valve core and allow the tire to deflate completely.

b. Break the adherence of the bead to the rim, on both sides of the tire, using either a rubber mallet or pressure applied with a hydraulic jack.

c. Set the tire on its tread and locate the valve stem. Push the stem into the casing behind one of the beads so that the stem will not catch on the wheel.

d. Work the casing off the wheel, using smooth tire irons.

2-452. INSTALLATION OF MAIN ALIGHTING GEAR TIRE AND TUBE. When remounting the tire, do not use powdered soapstone or talc on the tube or in the casing. To obtain a better balance of the wheel, the mark on the inner tube, which represents the heavy portion of the tube, should be lined up adjacent to the casing red balance marker, which indicates the lightest portion of the casing; then proceed as follows:

a. Place the tube in the casing and partially inflate it to eliminate the wrinkles.

b. With the valve stem placed in the proper position to be inserted through the hole in the wheel, place the casing over the wheel and double the valve stem into the casing behind the bead, so that the stem will not catch on the wheel.

c. Work the first bead on the wheel as far as possible, then work on the other one.

d. Pull the valve stem through the hole in the wheel with a suitable valve stem fishing tool. Install the valve stem washer and nut.

e. Without inserting the valve core, inflate the tire to 100 pounds, or enough to force the beads into place on the tapered bead seats. Deflate the tire, install the valve core, and inflate the tire to the proper pressure. Tighten the valve stem nut and install the valve cap.

2-453. AIRFRAME GROUP BOLT TORQUE VALUES.

2-454. If structural tension bolts are not properly tightened when installed, vibration and fatigue stresses will materially shorten their service life. It is equally dangerous, however, to tighten a bolt excessively, since over-tightening may result in stripped threads, a bent or sheared bolt, or eventual tension failure. It is essential, therefore, to observe the torque limits specified here when installing tension bolts. Unless otherwise noted on the installation drawing, bolts and nuts must be installed free of lubricants. Otherwise, the specified torque reading may produce an incorrect tension load. Torque readings should not be made on bolts that show painting or corrosion, as it is impossible to determine accurately the amount of tension being applied. Bolts and nuts that are to be cotter-pinned may be torqued to the special values shown in order to permit the alignment of their holes. However, these figures must never be exceeded, nor may the nuts be backed off to align the holes, unless they are backed off sufficiently to completely re-torque properly. When re-torquing bolts, the nut must be backed off part of a turn and then retightened to the correct value. This is necessary because more torque may be required to break the nut free than is actually on the bolt. To apply 700 inch-pounds and fail to move the nut does not imply that it is torqued to 700 inch-pounds. The nut may have a torque of only 650 inch-pounds, but 750 inch-pounds may be required to break it free. A value in excess of the required torque may also be needed to break the nut free in a counterclockwise direction. For this reason, the nut should be backed off and retightened, allowing no rotation of the bolt. Whenever possible, the torque reading must be taken on the nut and not on the bolt. Torque wrenches should be calibrated frequently to insure accurate readings. When adapters are used with torque wrenches, compensating torque values must be computed for each adapter and wrench involved. The following chart lists the standard torque values for structural tension bolts and also lists the bolts and nuts which should be installed with the torque values specified. Special torques are listed for all bolts to which the standard values are not applicable.

BOLT TORQUE VALUE TENSION CHART

Bolt Size	*AN Type Bolts*		*AN365 and AN310 Nuts*	*Torque Values*	*Cotter Pin Maximum Torque*
10-32	AN-3	AN-173	-1032	20-25 inch-pounds	40 inch-pounds
¼-28	AN-4	AN-174	-428	50-70 inch-pounds	100 inch-pounds
5/16-24	AN-5	AN-175	-524	100-140 inch-pounds	225 inch-pounds
⅜-24	AN-6	AN-176	-624	160-190 inch-pounds	390 inch-pounds
7/16-20	AN-7	AN-177	-720	450-500 inch-pounds	840 inch-pounds
½-20	AN-8	AN-178	-820	480-690 inch-pounds	1100 inch-pounds
9/16-18	AN-9	AN-179	-918	800-1000 inch-pounds	1600 inch-pounds
⅝-18	AN-10	AN-180	1018	90-105 foot-pounds	200 foot-pounds
¾-16	AN-12	AN-182	1216	190-210 foot-pounds	415 foot-pounds
⅞-14	AN-14	AN-184	1414	210-250 foot-pounds	590 foot-pounds
1-14	AN-16	AN-186	1614	310-460 foot-pounds	835 foot-pounds
1⅛-12	AN-18		1812	415-585 foot-pounds	1250 foot-pounds

Bolt Size	*NAS Heat-Treated Bolts*	*Douglas Heat-Treated Bolts*	*Elastic Stop Heat-Treated Nuts*	*Torque Values*
¼-28	NAS-144	S-2076904	12B-048	73-100 inch-pounds
5/16-24	NAS-145	S-2076905	12B-054	145-200 inch-pounds
⅜-24	NAS-146	S-2076906	12B-064	230-280 inch-pounds
7/16-20	NAS-147	S-2076907	12B-070	650-720 inch-pounds
½-20	NAS-148	S-2076908	12B-080	700-1000 inch-pounds
9/16-18	NAS-149	S-2076909	12B-098	100-120 foot-pounds
⅝-18	NAS-150	S-2076910	12B-108	135-155 foot-pounds
¾-16	NAS-152	S-2076912	12B-126	280-300 foot-pounds
⅞-14	NAS-154	S-2076914	12B-144	300-360 foot-pounds
1-14	NAS-156	S-2076916	12B-164	450-665 foot-pounds
1⅛-12	NAS-158	S-2076918	12B-182	605-845 foot-pounds

WING GROUP TORQUE VALUES

Attachment	*Bolt No.*	*Nut No.*	*No. Reqd.*	*Torque*
Inner wing to outer wing	NAS144	52-1610-048	654	135-150 inch-pounds
Front tank cover	NAS145	52-1610-054	160	135-150 inch-pounds
Rear tank cover	NAS145	52-1610-054	160	135-150 inch-pounds
Aileron Installation	5369009-22	AN365-720	4	450-500 inch-pounds
	2371804	AN365-720	2	450-500 inch-pounds
	2328991	AN365-720	2	450-500 inch-pounds
	2116354-3	AN365-524	8	100-140 inch-pounds

EMPENNAGE TORQUE VALUES

Attachment	Bolt No.	Nut No.	No. Required	Torque
Left Horizontal Stabilizer to Right Horizontal Stabilizer	AN-3	AN365-1032	148	20-25 inch-pounds
Elevator to Horizontal Stabilizer	2328991	AN365-720	4	450-500 inch-pounds
Rudder to Vertical Stabilizer	2328991	AN365-720	1	450-500 inch-pounds

SURFACE CONTROLS TORQUE VALUES

Attachment	Bolt No.	Nut No.	No. Required Per Airplane	Torque
Aileron to outer wing panel	5369609-22	AN365-720	4	450-500 inch-pounds
Elevator to horizontal stabilizer	2328991	AN365-720	6	450-500 inch-pounds
Rudder to vertical stabilizer	2328991	AN365-720	1	450-500 inch-pounds

FUSELAGE TORQUE VALUES

Attachment	Bolt No.	Nut No.	No. Required	Torque
Fuselage to wing	AN-6	AN310-624	24	160-190 inch-pounds
	AN-5	AN310-524	4	100-140 inch-pounds

ALIGHTING GEAR TORQUE VALUES

Location	Attachment	Installation Drawing	Bolt Number	Nut Number	Number Required Per Aircraft	Torque
Main Alighting Gear	Axle Clamp	5369471 or	Bendix 53926	Bendix 53927	4	270-320 foot-pounds
	Oleo Clamp	5369048	AN-10	AN310-104	8	90-200 foot-pounds
	Side Tube to Truss	5367272	NAS146	AN365-624	8	17 foot-pounds
			AN-8	AN365-820	16	39 foot-pounds
	Truss at Center		NAS148	12B-080	4	83 foot-pounds
			NAS150	12B-108	4	177 foot-pounds
	Mechanical Latch to Front Spar	5391088	1113780	AN310-624	4	160-190 inch-pounds
			1113781	AN310-720	4	450-500 inch-pounds
Tail Gear	Tail Wheel Spindle to Fork	5371349	2367425	AN310-14	1	135 ($\pm$ 7) foot-pounds

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z AIRCRAFT

SECTION III HYDRAULIC AND PNEUMATIC SYSTEM

THIS SECTION SUPERSEDES SECTION III OF AN 01-40NK-2
DATED 15 MAY 1952

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15 April 1953

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SECTION III

HYDRAULIC AND PNEUMATIC SYSTEMS

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SECTION III

HYDRAULIC AND PNEUMATIC SYSTEMS

3-1. HYDRAULIC SYSTEM.

3-2. DESCRIPTION. *(See figure 3-1.)* A pressure-accumulator-type hydraulic system is installed in the aircraft to operate the alighting gear, brakes, wing flaps, and windshield wipers. Two high-capacity, engine-driven, hydraulic pumps are installed, one on the accessory drive section of each engine, to furnish pressure for the hydraulic system. Only mineral oil hydraulic fluid, Specification MIL-O-5606 is to be used. The total capacity of the hydraulic system is approximately 6 US. (5.0 Imp.) gallons.

3-3. MINOR REPAIR AND REPLACEMENT OF HYDRAULIC SYSTEM COMPONENTS. Components of the hydraulic system, such as pumps, pressure regulator, pressure accumulator, etc., which are not functioning properly, should be removed and replaced with new or overhauled units. The pipes and fittings should be inspected for general condition, security of mounting, and evidence of leakage. Faulty pipes and fittings should be replaced if minor repair or adjustment is not sufficient to correct the unsatisfactory condition.

3-4. FILLING HYDRAULIC RESERVOIR. For detailed instructions on filling the hydraulic reservoir, see paragraph 1-96.

3-5. DRAINING HYDRAULIC RESERVOIR. For detailed instructions on draining the hydraulic reservoir, see paragraph 1-97.

3-6. TROUBLE SHOOTING OF HYDRAULIC SYSTEM. Hydraulic system failures may be traced to the three following sources: external leakage of fluid; internal leakage of fluid within the system; and mechanical failure of parts within the system. Before attempting any trouble shooting, make certain that both the air pressure in the accumulator (350 psi) and the fluid in the reservoir are sufficient. The hydraulic system *must* be free of air for proper operation. Bleed out a faulty system completely by operating its units through several cycles, with power supplied by a test stand. A system, in which a rework involving the breaking of a fitting or pipe has been performed, *must* be bled before proper operation can be restored. Hydraulic failures may be caused by leakage in any of the various valves in the system. Certain valves, however, are designed to permit a limited fluid loss without impairing proper operation.

Trouble	*Probable Cause*	*Remedy*
a. Dead system.	Lack of fluid in reservoir.	Check fluid quantity.
	External leaks.	Make visual check.
	System relief valve open.	Replace relief valve.
	Pressure regulator is stuck open.	Replace pressure regulator.
	Failure of both engine-driven pumps.	Apply test pressure with external power source. If pressure is registered on gage, pumps are faulty; remove and replace pumps.
b. Pressure drop to 950 psi is normal, but return to 1100 psi is slow.	Pressure regulator bypass valve partially stuck open.	Replace pressure regulator.
c. Pressure regulator operates too frequently (at intervals of less than 30 minutes), accompanied by unusual noise and fluctuation of pressure gage between 950 and 1100 psi. With pumps off, pressure may fall rapidly below 1100 psi.	Low air pressure in accumulator, leakage through one of several hydraulic sub-systems or leakage through system relief valve.	Add air to accumulator and check for leakage. Disconnect the system relief valve pipe from the hand pump shutoff valve fitting block and cap the fittings. Open hand pump shutoff valve and build up system pressure with test stand, reading pressure on test stand gage. If faulty condition persists, check sub-systems for leakage. If condition is remedied, system relief valve is at fault; remove and replace faulty relief valve.

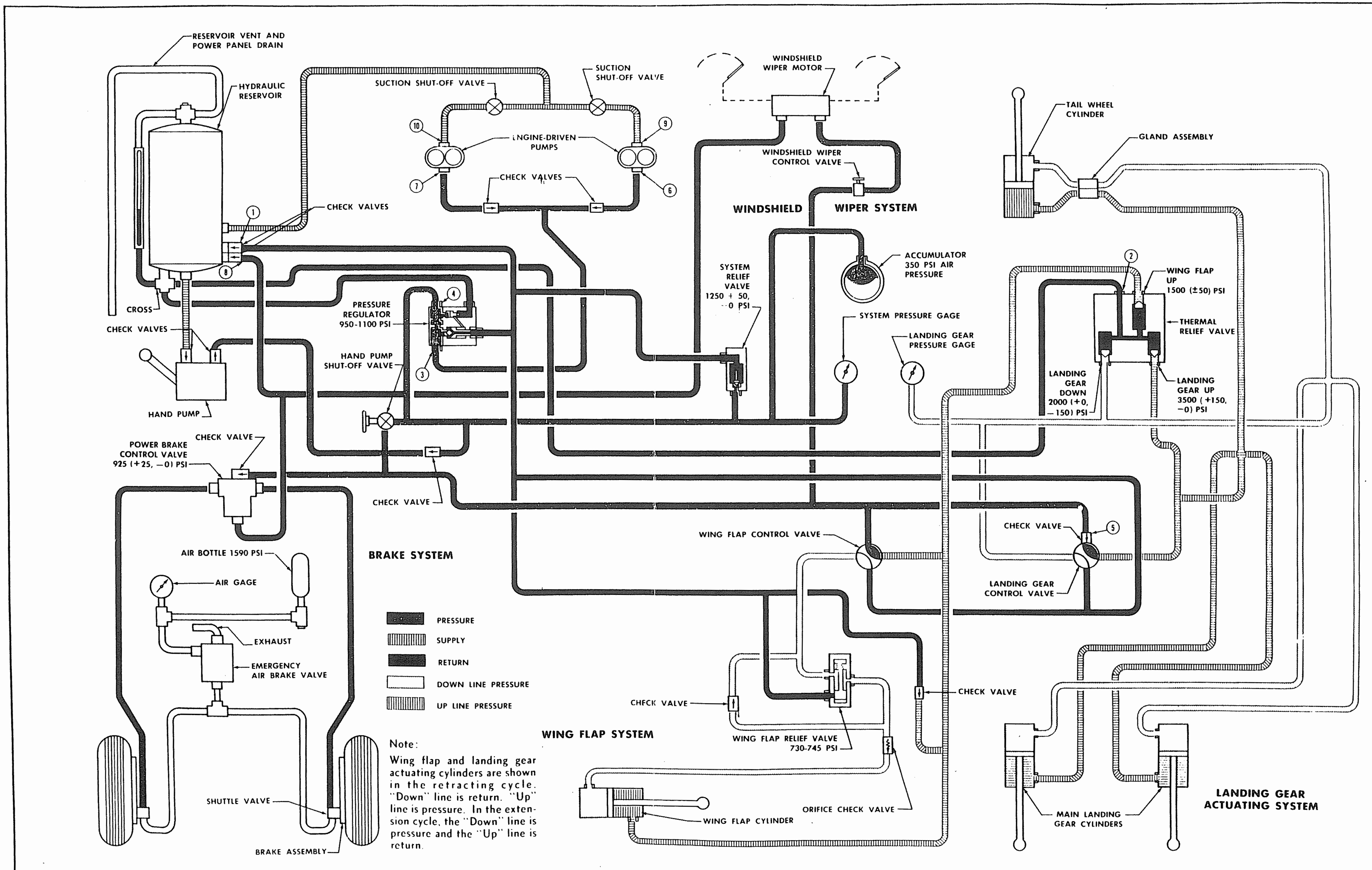


Figure 3-1. Hydraulic System — Schematic

Trouble	*Probable Cause*	*Remedy*
d. Overheating of hydraulic system, indicated by smoke emanating from hydraulic power panel compartment. Pressure reading on system gage is between 1100 and 1250 psi.	Pressure regulator is stuck in the closed position.	Replace the pressure regulator and check the system relief valve for damage.
e. Wing flaps droop.	Leakage through temperature relief valve.	Replace the faulty unit.
	Leakage through check valve between wing flap UP pipe and return pipe to reservoir.	Replace the faulty unit.
	Internal leakage in actuating strut.	Replace the faulty unit.

3-7. HYDRAULIC SYSTEM PIPES. If it becomes necessary to remove any of the pipes in the system, dummy plugs should be inserted in the openings to prevent the entrance of any foreign particles. The plugs should not be removed until just prior to the installation of the pipes.

3-8. REMOVAL OF HYDRAULIC SYSTEM PIPES. Before the removal of any of the hydraulic pipes, relieve the hydraulic system pressure by operating the wing flaps.

a. Disengage the fittings.

CAUTION

Do not attempt to loosen a flare nut with a standard open-end wrench. Use a tubing wrench, not over five inches long. This type of wrench can be made by cutting a standard box wrench so that it will just slip over the tubing.

b. Remove clips or clamps as necessary.

3-9. INSTALLATION OF HYDRAULIC SYSTEM PIPES. The following practices should be observed when installing the hydraulic pipes:

a. Check the tubing flares for out-of-round condition and evidence of cracks.

b. Clean out all pipes with a blast of clean, dry air.

c. Lubricate hydraulic pipe fittings (all straight threads of brass and steel) with hydraulic fluid only, when necessary.

d. Lubricate hydraulic pipe fittings (all straight threads of aluminum alloy fittings) with petrolatum.

e. Make certain that all clamps and blocks supporting the pipes fit snugly around the tubing. Loose pipes will set up static electric charges, which will interfere with the operation of the radio equipment.

f. Do not draw the flare to the fitting with the nut as this may spin off the flared end of the tubing. The tubing flare must meet the fitting squarely before the nut is started. It must be possible to start the nut by hand.

g. Install the clips or clamps.

3-10. HYDRAULIC SUPPLY, PRESSURE, AND RETURN SYSTEM. During normal operation, both engine-driven pumps supply fluid pressure to the hydraulic system. The hydraulic fluid, under pressure from the engine-driven pumps, flows through the hydraulic pressure regulator, to the hydraulic pressure accumulator, and then to the various systems. The hydraulic pressure regulator is set to maintain a system operating pressure of 1100 psi. If the hydraulic pressure regulator fails, the system relief valve opens when the hydraulic system pressure increases to 1250 psi and bypasses the excess fluid to the reservoir. The hydraulic hand pump on the hydraulic control panel may be utilized to operate the hydraulic system units when there is insufficient pressure in the hydraulic system or when the engine-driven hydraulic pumps fail. The hand pump may also be used to increase pressure in the hydraulic pressure accumulator for ground operation of the hydraulic system when the engines are not running. The hand pump-to-pressure accumulator shutoff valve, located on the hydraulic control panel, is installed to provide a bypass around the check valve in the pressure manifold when it is desired to increase pressure in the pressure accumulator by operating the hand pump. In flight, the hand pump shutoff valve must always remain closed. A firewall hydraulic shutoff valve is installed in the engine pump suction pipe aft of each engine section firewall to stop the flow of hydraulic fluid in the event of an engine section fire.

3-11. HYDRAULIC RESERVOIR. *(See figure 3-2.)* The hydraulic system fluid reservoir is a cylindrical container, constructed of aluminum sheets that are riveted and welded together. The reservoir incorporates a strainer assembly to filter the hydraulic fluid, and a baffle to separate the strained fluid from the fluid returned from the hydraulic system. The reservoir is installed to the right of the hydraulic control panel, with a filler neck that extends into the companionway. A sight gage is installed on the upper portion of the hydraulic control panel. The outlet to the engine-driven hydraulic pumps is located above the bottom of the reservoir so that an emergency supply of fluid, available only to the hydraulic hand pump, remains in the bottom of the reservoir. The outlet for the hydrau-

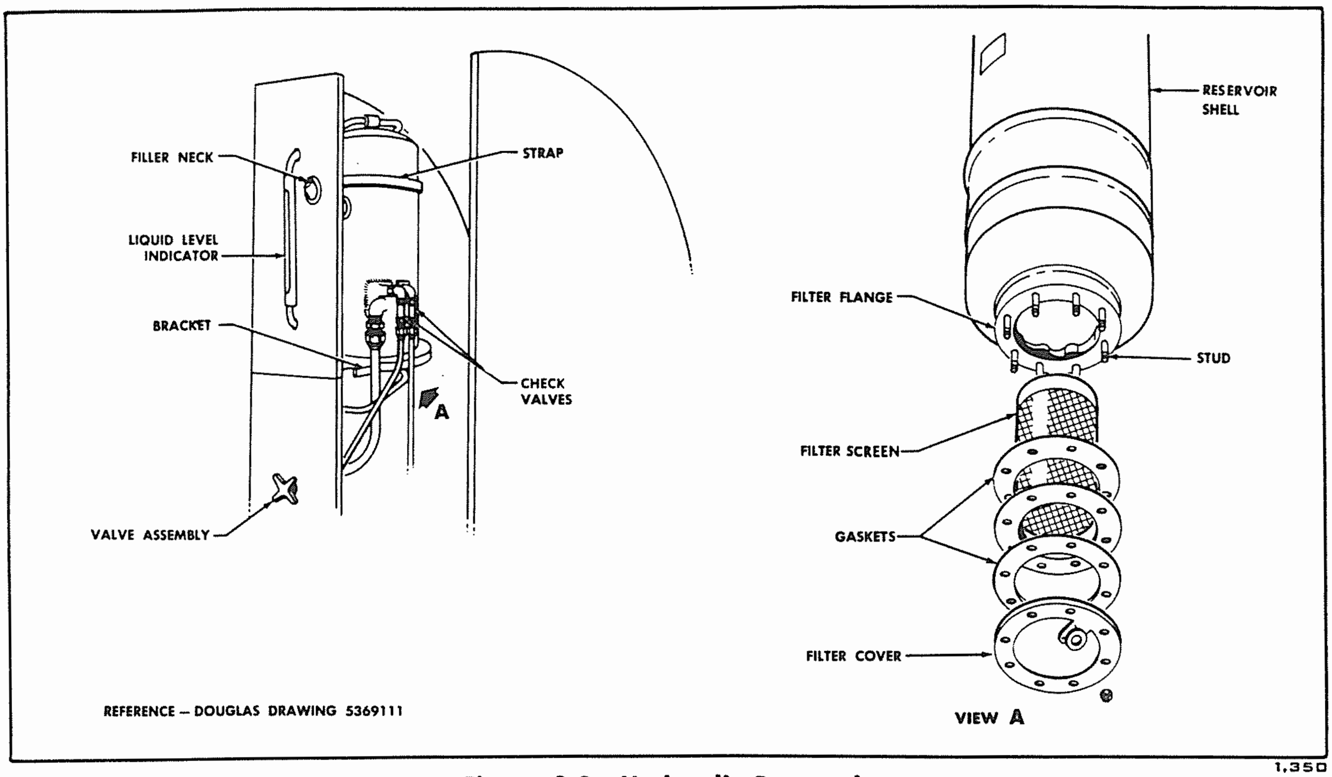


Figure 3-2. Hydraulic Reservoir

lic hand pump is located at the bottom of the reservoir sump. The total capacity of the hydraulic reservoir, when filled to the top arrow on the sight gage, is 3.1 US. (2.59 Imp.) gallons with a reserve capacity of 3.2 US. (2.66 Imp.) quarts, available only to the hydraulic hand pump. Sufficient space is provided at the top of the reservoir to allow for thermal expansion. A vent pipe is connected to the top of the reservoir.

Note

When the aircraft is in flight and the hydraulic units are being operated, the vortex caused within the reservoir may draw the fluid out of the fluid level sight gage. However, the vortex usually will subside when the hydraulic system is not being operated, permitting the fluid level to rise in the sight gage.

3-12. REMOVAL OF HYDRAULIC RESERVOIR.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

b. Disconnect one pipe at the cross at the bottom of the reservoir and drain the hydraulic fluid into a clean container.

c. Disconnect and cap all pipes leading to the reservoir.

d. Remove the strap assembly that secures the reservoir and remove the reservoir from the aircraft.

3-13. INSTALLATION OF HYDRAULIC RESERVOIR. Reverse the removal procedure.

3-14. TEST AFTER INSTALLATION OF HYDRAULIC RESERVOIR. For a complete description of the test procedure for the hydraulic reservoir, see paragraph 3-64.

3-15. ENGINE-DRIVEN HYDRAULIC PUMPS. *(See figure 3-3.)* Two engine-driven, positive-displacement hydraulic pumps with a rated capacity of 3 gallons per minute, at 1500 rpm (1500 psi), are used to provide fluid pressure for the hydraulic system. The engine-driven pumps are mounted on the right side of each engine accessory section.

3-16. REMOVAL OF ENGINE-DRIVEN HYDRAULIC PUMP.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

b. Disconnect the flexible hoses leading to the engine-driven pump at the bulkhead fittings on the firewall. Cap the exposed ends of the flexible hose. Disconnect and cap the pump drain pipe.

c. Remove the nuts that attach the pump to the mounting pad. Remove the pump and flexible hoses from the aircraft.

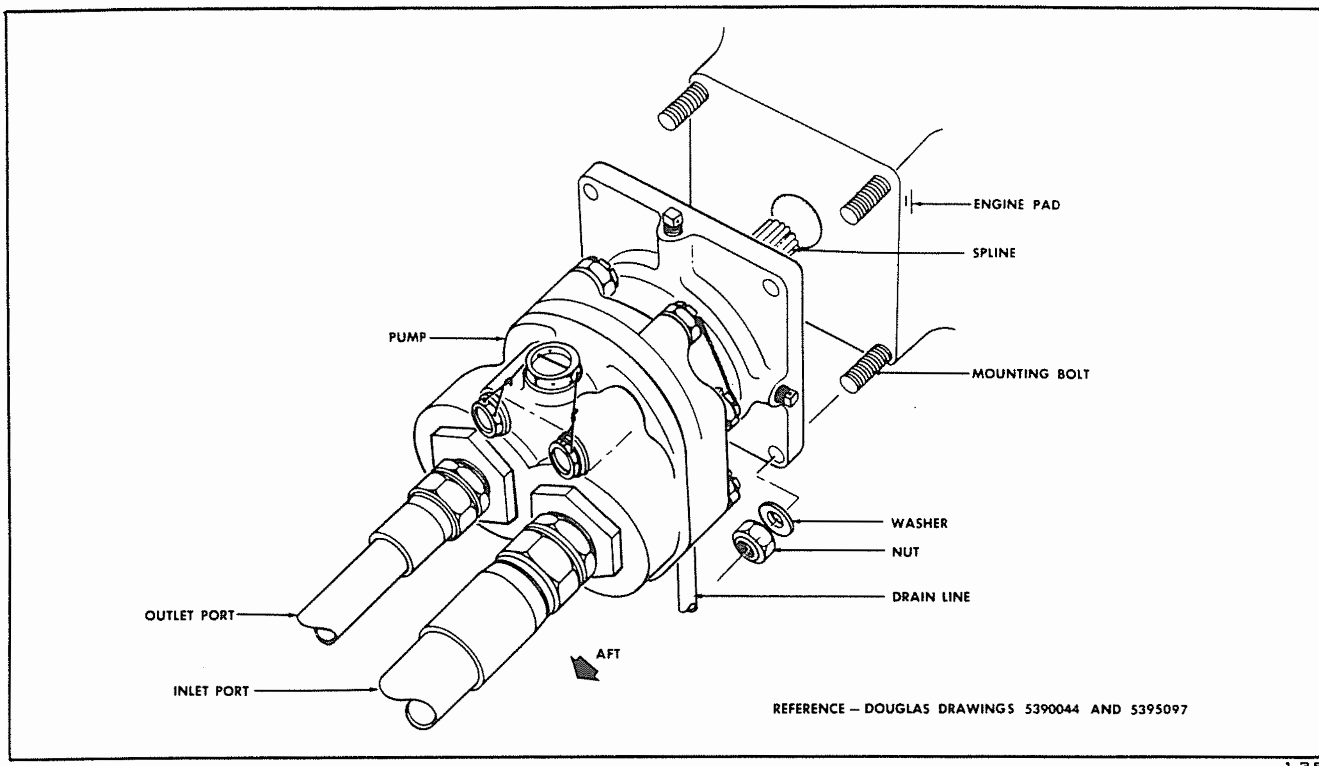


Figure 3-3. Engine-Driven Hydraulic Pump

3-17. INSTALLATION OF ENGINE-DRIVEN HYDRAULIC PUMP.

a. If a new engine-driven pump is installed, drain the oil from the pump and flush with mineral oil hydraulic fluid.

b. Wipe the engine mounting pad to remove any dirt or foreign particles. Turn the pump drive couplings with the fingers to test for freedom of operation. Note that certain tightness is inherent in the pump design.

c. Prior to installation, coat the drive coupling splines with anti-seize compound, Specification No. AN-C-53.

d. Note carefully the direction of rotation of the engine drive and make certain that it corresponds with the direction-of-rotation markings on the pump.

e. Place the seal on the mounting pad. Mount the pump on the studs. Tighten the mounting nuts evenly and lock with safety wire.

f. Install the fittings in the pump ports. Install the flexible hoses and connect the drain pipe.

3-18. ADUSTMENT AFTER INSTALLATION OF HYDRAULIC PUMP.

a. Check the fluid supply pipe from the hydraulic reservoir to the pump for evidence of leaks. Leaking connections may be tightened moderately to stop the fluid leakage. If tightening the connections will not correct the condition, the connector or pipe must be replaced with a new part.

b. After the installation is complete, operate the hydraulic hand pump to fill the pipes and hydraulic units with fluid. Operate all hydraulic units through several cycles to expel any trapped air.

3-19. TEST AFTER INSTALLATION OF HYDRAULIC PUMP. For a complete description of the test procedure for the hydraulic pumps, see paragraph 3-64.

3-20. HYDRAULIC SYSTEM POWER PANEL. The hydraulic system power panel is located on the right side of the companionway, aft of the co-pilot's seat *(see figures 3-4 and 3-5).* The control panel is divided into two parts. The lower panel assembly consists of the panel, the wing flap control valve, the alighting gear control valve, the hydraulic hand pump, the hydraulic pressure regulator, the system pressure relief valve, the wing flap relief valve, the hand pump-to-pressure accumulator shutoff valve, the wing flap and alighting gear relief valve, the windshield wiper control valve, and the necessary pipes and fittings. The upper panel includes the hydraulic reservoir fluid level sight gage and a filling instructions placard.

3-21. REMOVAL OF LOWER HYDRAULIC SYSTEM POWER PANEL.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

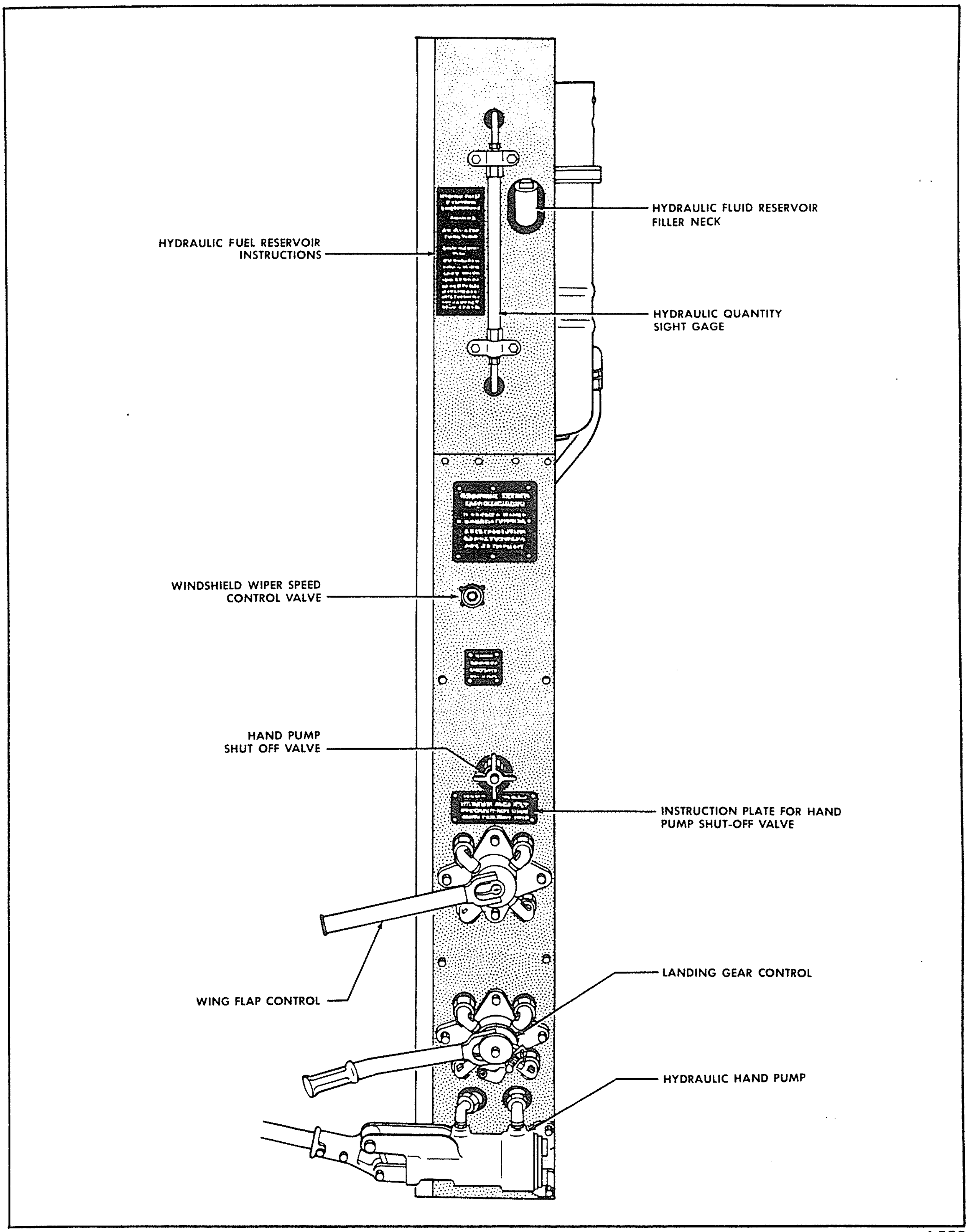
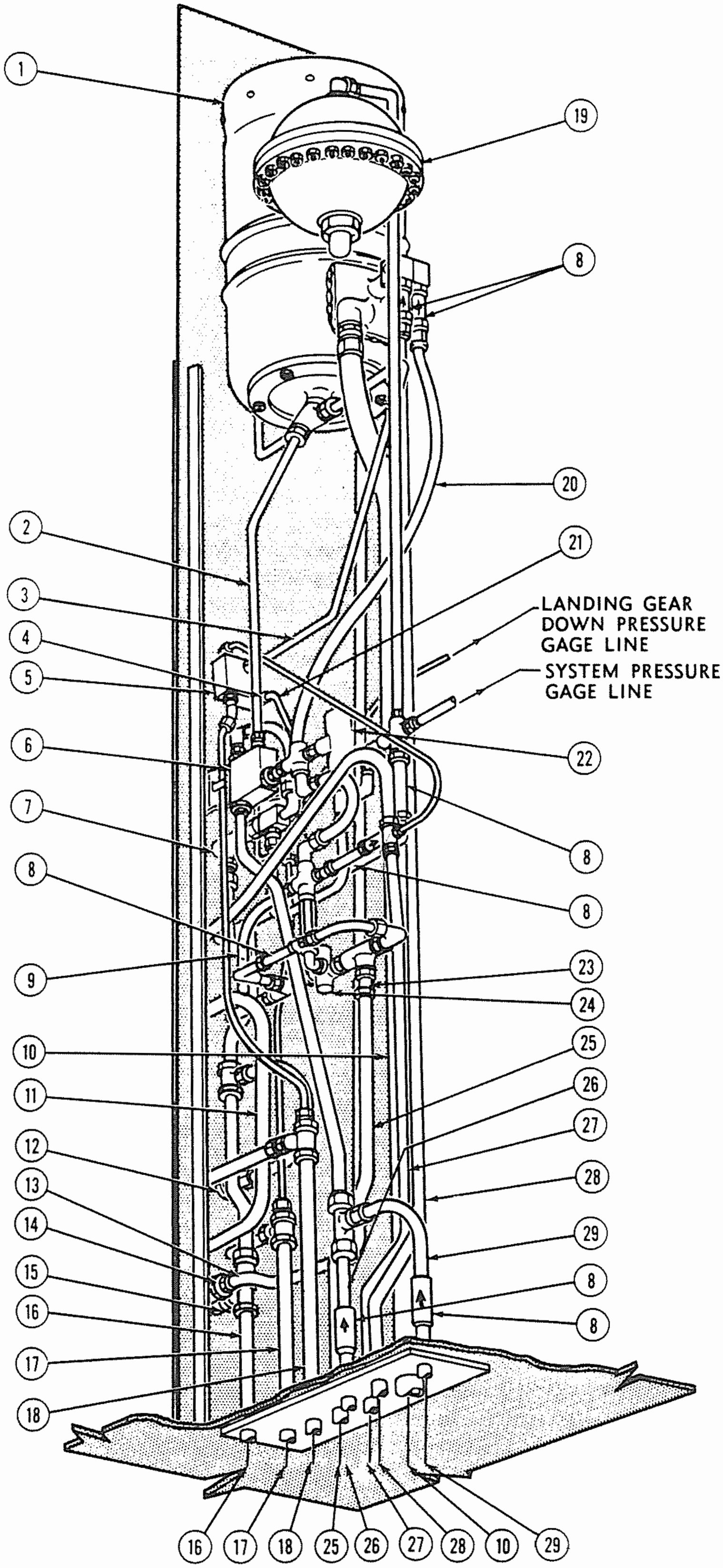


Figure 3-4. Hydraulic System Power Panel — Front View



1. Hydraulic Reservoir
2. Pressure Regulator Drain Line
3. Thermal Relief Return Line
4. Windshield Wiper Control Valve
5. Thermal Relief Valve Assembly
6. Pressure Regulator
7. Hand Pump Shut-Off Valve Assembly
8. Check Valve
9. Manifold Assembly Pressure Line
10. Engine Pump Suction Line
11. Manifold Assembly Return Line
12. Landing Gear Inlet Check Valve Assembly
13. Hand Pump Supply from Reservoir
14. Hand Pump Inlet Check Valve Assembly
15. Hand Pump Outlet Check Valve Assembly
16. Brake Supply Line
17. Landing Gear Down Line
18. Landing Gear Up Line
19. Hydraulic Pressure Accumulator
20. System Return Line
21. Windshield Wiper Line
22. System Relief Valve
23. Orifice Check Valve Assembly
24. Wing Flap Relief Valve Assembly
25. Wing Flap Down Line
26. LH Pump Pressure Line
27. Wing Flap Up Line
28. Brake Return Line
29. RH Pump Pressure Line

REFERENCE — DOUGLAS DRAWINGS 5369111 AND 5372096

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Figure 3-5. Hydraulic System Power Panel — Rear View

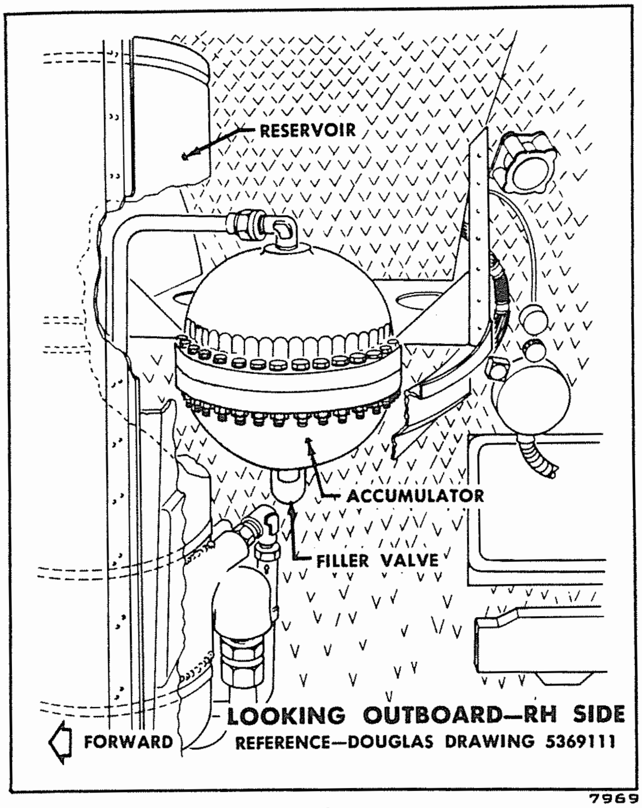

Figure 3-6. Hydraulic System Pressure Accumulator

b. Disconnect and cap all pipes leading to the panel assembly. Each disconnect point is indicated by a B-nut that is painted international orange.

c. Remove the bolts attaching the panel to the frames and remove the panel assembly.

3-22. INSTALLATION OF LOWER HYDRAULIC SYSTEM POWER PANEL. Reverse the removal procedure.

3-23. TEST AFTER INSTALLATION OF LOWER HYDRAULIC SYSTEM POWER PANEL. For a complete description of the test procedure for the hydraulic system power panel, see paragraph 3-64.

3-24. HYDRAULIC SYSTEM PRESSURE REGULATOR. The hydraulic system pressure regulator is installed on the outboard side of the hydraulic system power panel *(see figure 3-5)*. The pressure regulator maintains the pressure of the hydraulic system at between 950 and 1100 psi by the operation of a pilot valve assembly, which controls the closing and opening of a poppet valve to direct the hydraulic pump discharge either to the accumulator until the system pressure of 1100 psi has been reached or to the reservoir after the system pressure of 1100 psi has been reached.

3-25. REMOVAL OF HYDRAULIC SYSTEM PRESSURE REGULATOR.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and drain the hydraulic fluid from the system.

b. Caution must be observed when removing components from the hydraulic system power panel. Mark all pipes and fittings as removed, so that they can be replaced in their original positions on the control panel.

c. Disconnect and cap all pipes.

d. Remove the three bolts and nuts that attach the pressure regulator to the bracket on the hydraulic system power panel and remove the pressure regulator.

3-26. INSTALLATION OF HYDRAULIC SYSTEM PRESSURE REGULATOR. Reverse the removal procedure.

3-27. ADJUSTMENT AFTER INSTALLATION OF HYDRAULIC SYSTEM PRESSURE REGULATOR. If the pressure regulator does not maintain the proper maximum system operating pressure of 1100 psi (cut-out pressure), the pressure regulator may be adjusted by turning the adjusting screw above the return port clockwise to increase the cutout pressure, and counter-clockwise to decrease it. Then tighten the locknut to assure no further movement of the adjusting screw. The cutin pressure cannot be separately adjusted, and if it is below 950 psi, when the cutout pressure is 1100 psi, the unit should be replaced. The minimum permissible range is 80 psi.

Note

If the regulator cannot be adjusted to the proper operating pressure by means of the adjusting screw, remove and replace it with a new or serviceable regulator.

3-28. TEST AFTER INSTALLATION OF HYDRAULIC SYSTEM PRESSURE REGULATOR. For a complete description of the test procedure for the hydraulic system pressure regulator, see paragraph 3-64.

3-29. HYDRAULIC SYSTEM RELIEF VALVE. The hydraulic system pressure relief valve is located adjacent to the pressure regulator on the hydraulic system power panel assembly *(see figure 3-5)*. The system pressure relief valve functions to protect the hydraulic system from excessive fluid pressure when the hydraulic pressure regulator fails to operate. In such cases when the system pressure increases to 1250 (+ 50, — 0) psi, the system pressure relief valve is opened and the excess fluid flows through the system pressure relief valve to the hydraulic reservoir return pipe. The ball in the system pressure relief valve is reseated when the pressure decreases to approximately 1150 psi.

3-30. REMOVAL OF HYDRAULIC SYSTEM RELIEF VALVE.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and drain the hydraulic fluid from the system.

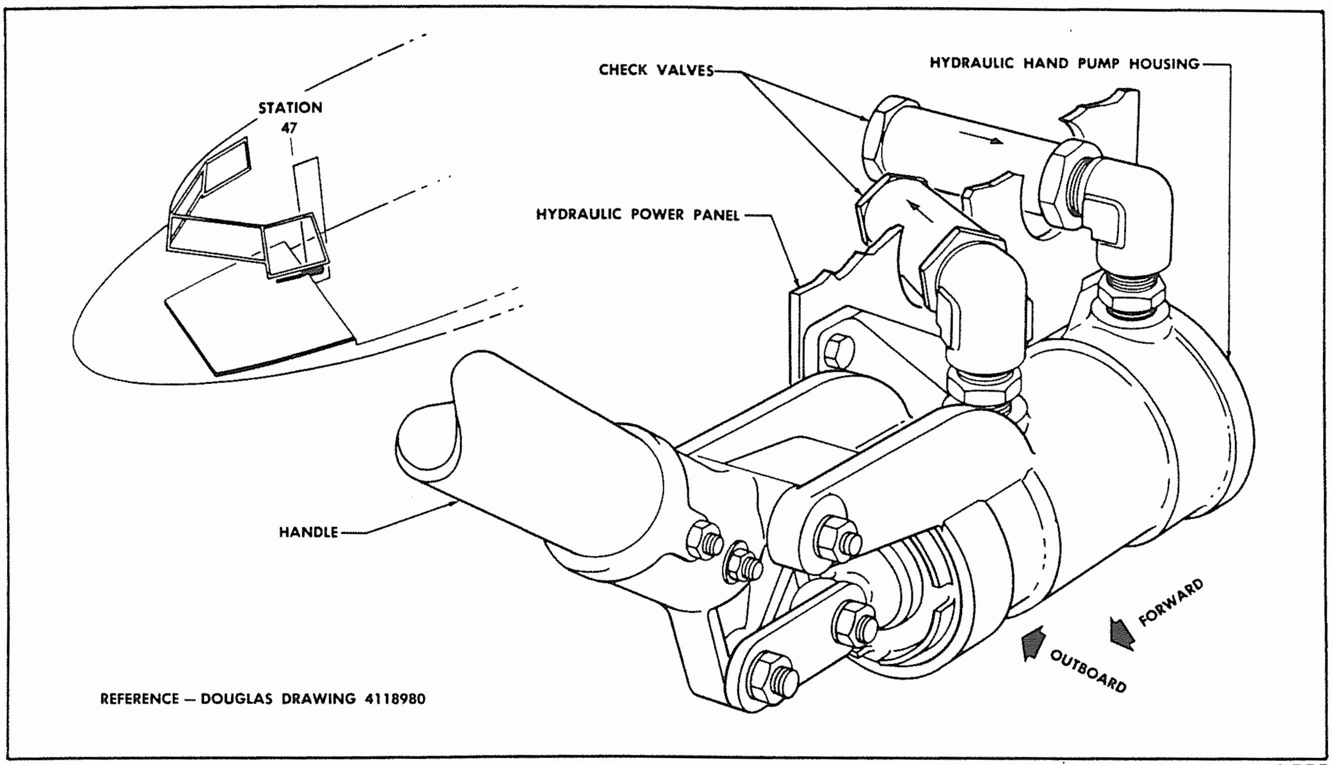

Figure 3-7. Hydraulic System Hand Pump

b. Disconnect and cap all pipes.

c. Remove the bolts and nuts attaching the pressure regulator to the control panel and remove the pressure regulator and fittings, including the system pressure relief valve.

d. Remove the pressure relief valve from the pressure regulator.

3-31. INSTALLATION OF HYDRAULIC SYSTEM RELIEF VALVE. Reverse the removal procedure.

3-32. ADJUSTMENT OF HYDRAULIC SYSTEM RELIEF VALVE. Adjust the spring release pressure so that the valve opens when the system pressure is greater than 1250 psi and closes when the system pressure decreases to 1250 psi. The spring tension may be increased or decreased by tightening or loosening the stem guide, after loosening the locknut. After the valve has been adjusted properly, tighten the locknut and replace the cap.

3-33. TEST AFTER INSTALLATION OF HYDRAULIC SYSTEM RELIEF VALVE. For a complete description of the test procedure for the hydraulic system relief valve, see paragraph 3-64.

3-34. HYDRAULIC SYSTEM PRESSURE ACCUMULATOR. The hydraulic system pressure accumulator is located aft of the bulkhead behind the co-pilot's seat and adjacent to the hydraulic fluid reservoir *(see figure 3-6).* The pressure accumulator functions to store energy so that a supply of hydraulic fluid under pressure is available instantly when either the engine-driven pumps are inoperative and/or when fluid is drawn from the pressure system faster than the engine-driven pumps can replenish the fluid supply. The pressure accumulator also functions as a shock absorber to dampen surges in the hydraulic system. Maximum operating pressure of this accumulator is 1100 psi.

3-35. REMOVAL OF HYDRAULIC SYSTEM PRESSURE ACCUMULATOR.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

b. Relieve the air pressure in the pressure accumulator by backing off the air valve body slightly, allowing the air pressure to escape.

c. Disconnect and cap the pipe leading to the pressure accumulator.

d. Remove the three nuts attaching the pressure accumulator to the supporting bracket and remove the pressure accumulator.

3-36. INSTALLATION OF HYDRAULIC SYSTEM PRESSURE ACCUMULATOR. Reverse the removal procedure.

3-37. INFLATING HYDRAULIC SYSTEM PRESSURE ACCUMULATOR. For a complete description of the pressure accumulator inflating procedure, see paragraph 1-98.

3-38. TEST OF HYDRAULIC SYSTEM PRESSURE ACCUMULATOR. For a complete description of the test procedure for the pressure accumulator, see paragraph 3-64.

3-39. HYDRAULIC SYSTEM HAND PUMP. The hydraulic system hand pump is located at the bottom of the hydraulic system power panel, adjacent to the companionway and aft of the co-pilot's seat *(see figure 3-7)*. The hydraulic hand pump serves two functions: it supplies hydraulic fluid pressure in flight in the event of an engine pump failure or it will supply hydraulic fluid pressure for the operation of hydraulic units when the aircraft is on the ground and it is not convenient to operate the engines. The hand pump may also be used to increase the pressure in the accumulator. Hydraulic fluid is supplied to the hand pump from the bottom of the hydraulic fluid reservoir. The reservoir is so designed that the engine-driven pump can not draw from the emergency supply of fluid reserved for the hand pump.

3-40. REMOVAL OF HYDRAULIC SYSTEM HAND PUMP.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and drain the hydraulic fluid from the system.

b. Disconnect the suction and pressure pipes from the external check valves.

c. Remove the bolts that attach the hand pump to the hydraulic system power panel and remove the pump.

d. Plug the suction and return pipes.

3-41. INSTALLATION OF HYDRAULIC SYSTEM HAND PUMP. Reverse the removal procedure.

3-42. TEST OF HYDRAULIC SYSTEM HAND PUMP. For a complete description of the test procedure for the hand pump, see paragraph 3-64.

3-43. HYDRAULIC SYSTEM HAND PUMP SHUTOFF VALVE. The hand pump shutoff valve is located on the hydraulic system power panel, directly above the wing flap control valve *(see figure 3-8)*.

3-44. REMOVAL OF HYDRAULIC SYSTEM HAND PUMP SHUTOFF VALVE.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

b. Remove the handle from the hand pump shutoff valve.

c. Disconnect and cap the pipes to the manifold block below the pressure regulator attach bracket.

d. Remove the bolts attaching the manifold to the pressure regulator bracket.

e. Remove the hand pump shutoff valve and the manifold block as a unit.

3-45. INSTALLATION OF HYDRAULIC SYSTEM HAND PUMP SHUTOFF VALVE. Reverse the removal procedure.

3-46. HYDRAULIC SYSTEM PRESSURE GAGES. The hydraulic system pressure gages are located on the right side of the pilot's compartment *(see figure 3-9)*. One gage indicates the fluid pressure in the hydraulic system, and the other indicates the fluid pressure in the alighting gear DOWN pipe.

3-47. REMOVAL OF HYDRAULIC SYSTEM PRESSURE GAGES.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gages indicate zero, and drain the hydraulic fluid from the system.

b. Remove the top and bottom bolts and nuts attaching the hydraulic system instrument panel assembly to the fuselage structure and remove the panel.

c. Disconnect and cap the hydraulic pipes leading to the gages.

d. Remove the screws attaching the pressure gages and collars to the bracket assembly, and remove the gages.

e. Perform the above steps in removing the emergency air brake pressure gage, after relieving air pressure in the system.

3-48. INSTALLATION OF HYDRAULIC SYSTEM PRESSURE GAGES. Reverse the removal procedure.

3-49. WING FLAP CONTROL VALVE. The wing flap control valve is located on the hydraulic system power panel, just below the hand pump shutoff valve *(see figure 3-4)*.

3-50. REMOVAL OF WING FLAP CONTROL VALVE.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

b. Disconnect and cap the four hydraulic pipes leading to the wing flap control valve elbows.

c. Remove the bolts attaching the valve to the panel and remove the valve.

3-51. INSTALLATION OF WING FLAP CONTROL VALVE. Reverse the removal procedure.

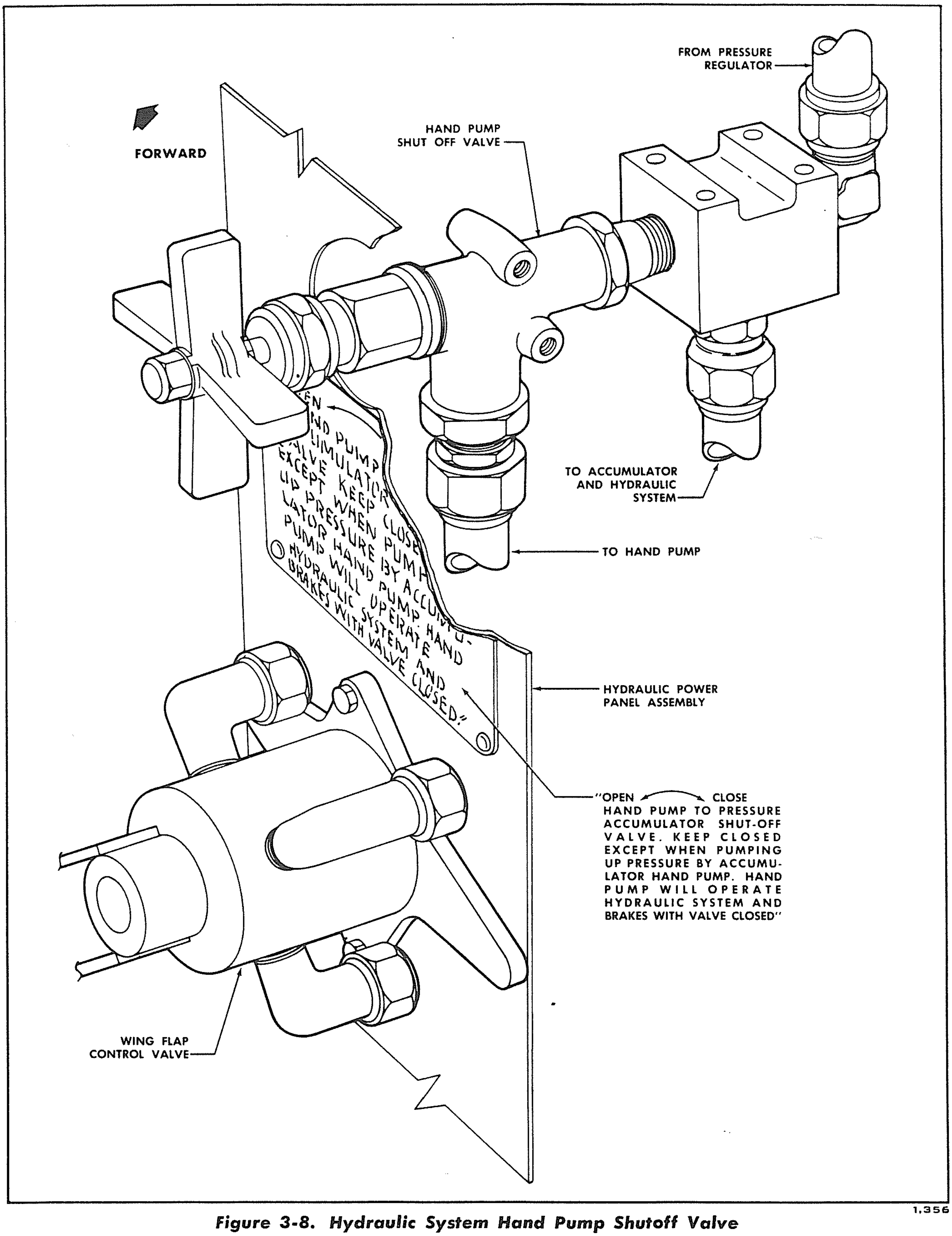

Figure 3-8. Hydraulic System Hand Pump Shutoff Valve

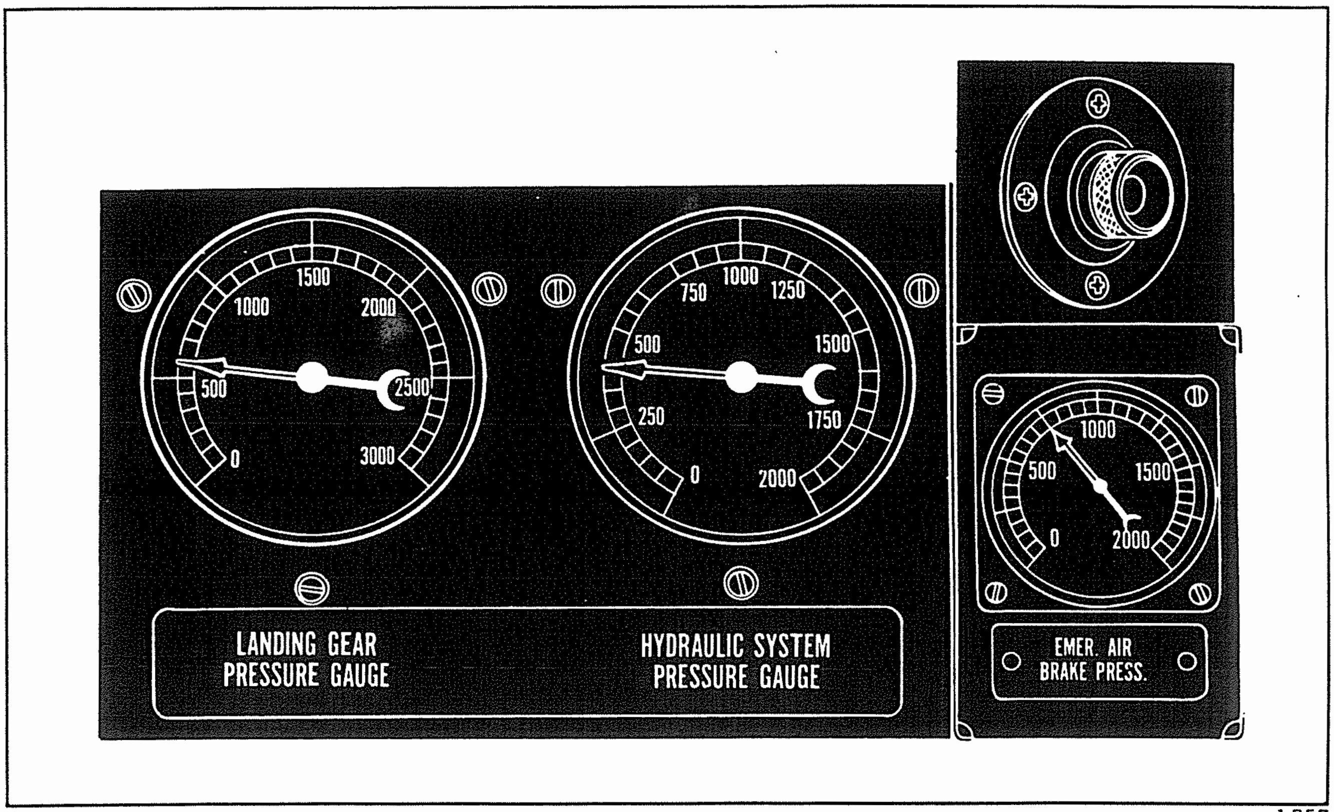

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Figure 3-9. Hydraulic System Pressure Gages

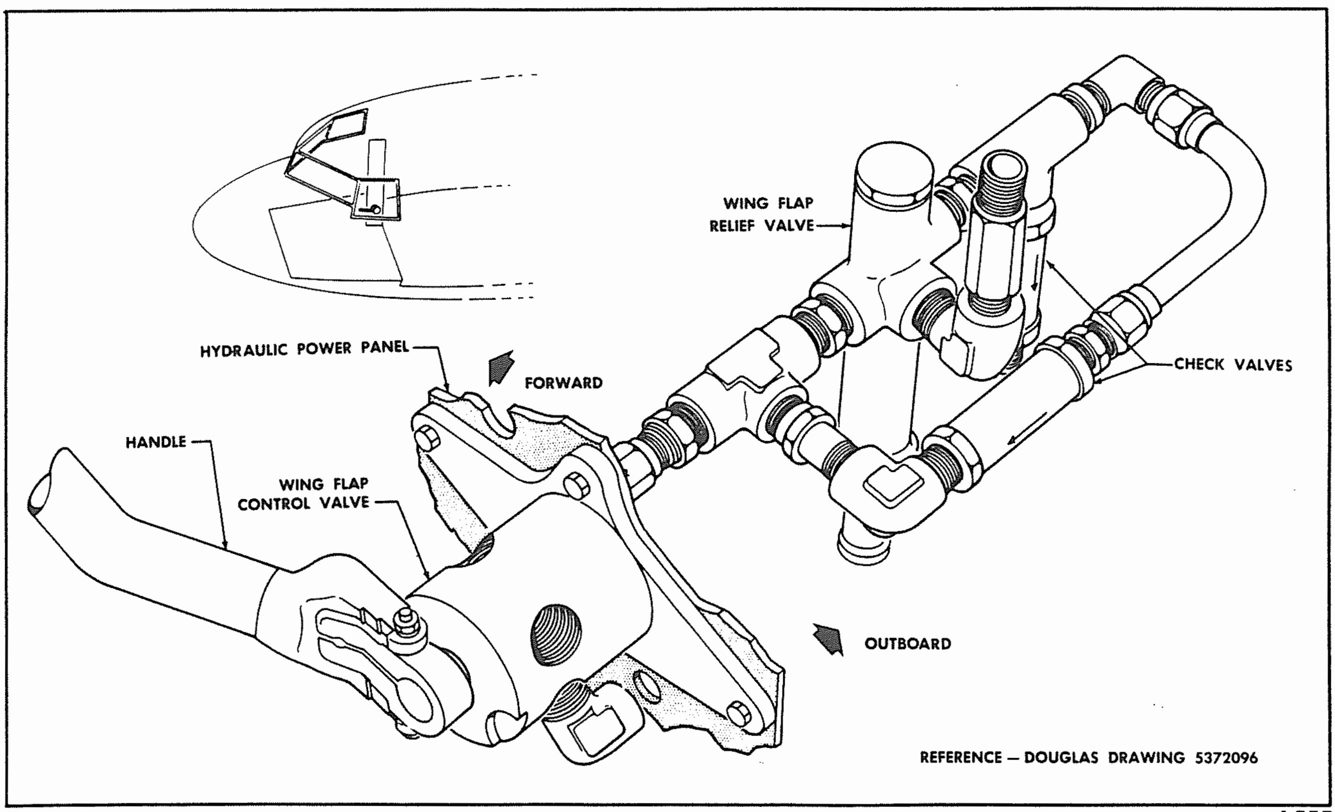

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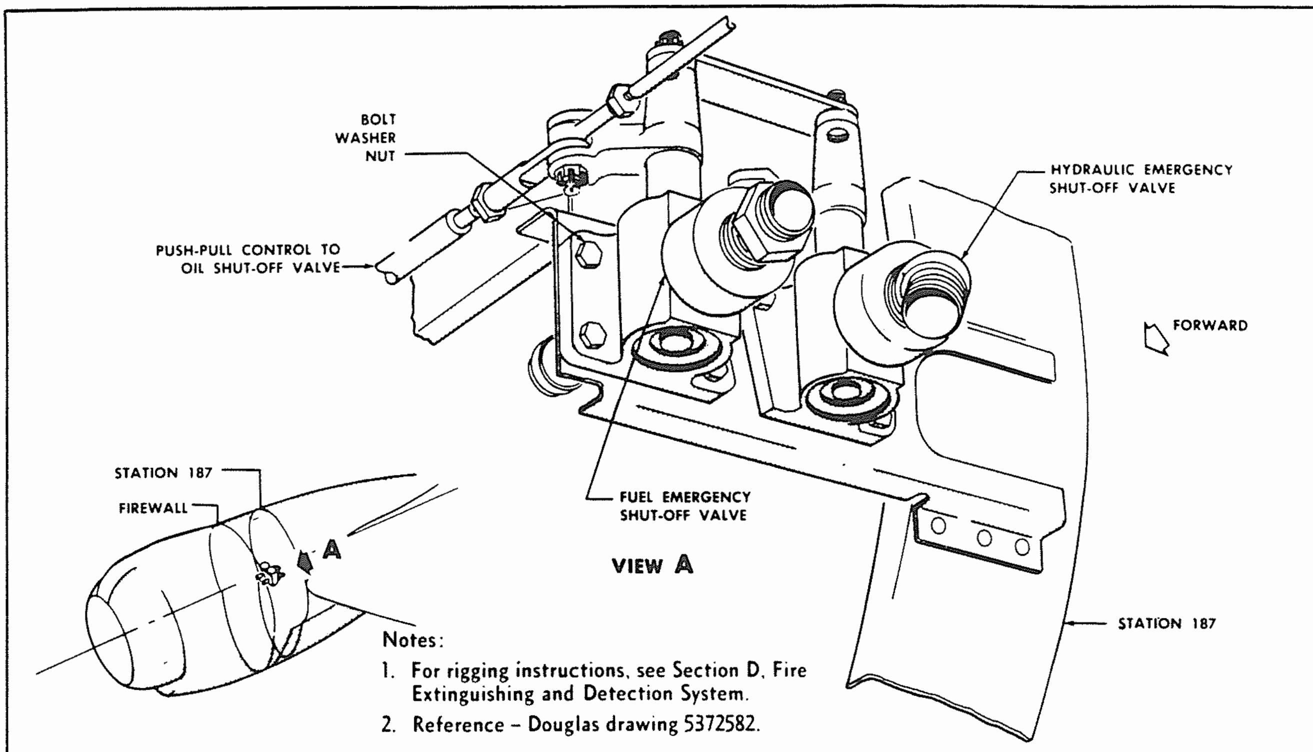
Figure 3-10. Wing Flap Relief Valve



Notes:
1. For rigging instructions, see Section D, Fire Extinguishing and Detection System.
2. Reference - Douglas drawing 5372582.

Figure 3-11. Hydraulic System Emergency Shutoff Valve

3-52. TEST AFTER INSTALLATION OF WING FLAP CONTROL VALVE. For a complete description of the test procedure for the wing flap control valve, see paragraph 3-64.

3-53. WING FLAP RELIEF VALVE. *(See figure 3-10.)* The wing flap relief valve is located behind the wing flap control valve, on the wing flap DOWN pipe. The valve limits the fluid pressure in the forward end of the wing flap actuating cylinder to between 730 and 745 psi to prevent possible damage to the wing flaps, caused by excessive air loads. The wing flap relief valve consists essentially of a spring-loaded piston that slides in the valve body and controls the flow of fluid through the three ports.

3-54. REMOVAL OF WING FLAP RELIEF VALVE.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and place the wing flap control valve in neutral.

b. Disconnect and cap the hydraulic pipes leading to the wing flap relief valve, and check valve assembly, and remove the assembly.

3-55. INSTALLATION OF WING FLAP RELIEF VALVE. Reverse the removal procedure.

3-56. WING FLAP AND ALIGHTING GEAR THERMAL RELIEF VALVE. The wing flap and alighting gear relief valve is located on the hydraulic system power panel, adjacent to the windshield wiper control valve *(see figure 3-5)*. The valve functions to prevent possible damage, caused by thermal expansion, in the flap and alighting gear systems when the aircraft is on the ground. When the aircraft is in flight and the alighting gear is retracted, the valve serves to protect the alighting gear hydraulic system from excessive loads due to gust conditions.

3-57. REMOVAL OF WING FLAP AND ALIGHTING GEAR THERMAL RELIEF VALVE.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero.

b. Disconnect and cap the hydraulic pipes leading to the thermal relief valve and remove the valve.

3-58. INSTALLATION OF WING FLAP AND ALIGHTING GEAR THERMAL RELIEF VALVE. Reverse the removal procedure.

3-59. ALIGHTING GEAR CONTROL VALVE. The alighting gear control valve is located on the hydraulic system power panel immediately above the hydraulic hand pump. For detailed information, see paragraph 2-379.

3-60. HYDRAULIC SYSTEM EMERGENCY SHUT-OFF VALVES. *(See figure 3-11.)* One hydraulic emergency shutoff valve is installed in the hydraulic fluid supply pipe on the aft side of the firewall of each nacelle.

3-61. REMOVAL OF HYDRAULIC SYSTEM EMERGENCY SHUTOFF VALVES.

a. Drain the reservoir.

b. Remove the nut and bolt attaching the valve control arm to the valve operating shaft.

c. Remove the pipe clamp located immediately forward of the valve mounting bracket.

d. Disconnect the pipe at the elbow immediately aft of the valve. Remove the nuts from the valve mounting bracket and remove the valve by pulling aft. Cap the pipes to prevent fluid leakage.

3-62. INSTALLATION OF HYDRAULIC SYSTEM EMERGENCY SHUTOFF VALVES. Reverse the removal procedure.

3-63. HYDRAULIC SYSTEM EMERGENCY SHUTOFF VALVE CONTROL CABLES. For a complete description of the removal, installation, and rigging procedures, see paragraph 4-213.

3-64. OPERATIONAL TEST AND ADJUSTMENT OF HYDRAULIC SYSTEM.

3-65. PRESSURE AND RETURN SYSTEM CHECK. *(See figure 3-1.)* .

a. Safety the gear to prevent retraction.

b. Jack up the aircraft (see paragraph 1-64).

c. Insert a 450-psi relief valve at (1) and a 300-psi relief valve at (2).

d. Disconnect and cap the pressure and system pipes at the pressure regulator and insert a jumper pipe between (3) and (4).

e. Disconnect and cap the pressure pipe at the alighting gear control valve inlet check valve (5).

f. Apply 1650-psi hydraulic pressure, with the test stand connected at the right engine pump flex hoses (6) and (9) and with the right pump suction shutoff valve open. Operate the wing flaps and windshield wiper controls (do not operate wipers on dry glass). Check the pipes for security of connections and evidence of leakage.

g. Apply 1650-psi pressure, with the test stand connected at the left engine pump flex hoses (7) and (10) and with the left pump suction shutoff valve open. Check the pump pressure pipe and connections for leakage.

h. Relieve the pressure in the system. Open the brake control valve and apply 1000-psi hydraulic pressure to the brake pipes. Check for evidence of leakage.

i. With the alighting gear control valve handle in the DOWN position, apply 2000-psi hydraulic pressure at the alighting gear control valve inlet check valve (5). Check the alighting gear and tail wheel DOWN pipes for evidence of leakage.

j. Release the alighting gear DOWN latches and remove the safety pins. With the alighting gear control valve handle in the UP position, apply 3500-psi hydraulic pressure at (5). Check the alighting gear and tail wheel UP pipes for evidence of leakage. Latch and safety the gear.

Note

The shock struts must be in the fully extended position before retracting the gear.

k. Disconnect the brake return pipe at the reservoir check valve (8) and apply 50-psi hydraulic pressure. Check the brake and windshield wiper return pipes for evidence of leakage. Reconnect the pipe to the check valve.

l. Remove the 450-psi relief valve at (1) and the 300-psi relief valve at (2) and connect the pipes.

m. Remove the jumper pipe at the pressure regulator and connect the pressure and system pipes at (3) and (4).

n. Reconnect the pressure pipe at the alighting gear control valve inlet check valve (5).

3-66. SUCTION SYSTEM CHECK. With both pump suction shutoff valves open, apply 25-psi air pressure at the reservoir drain pipe. Check all suction pipes for leakage, including the hand pump suction pipe and the thermal relief valve return pipe.

3-67. OPERATION CHECK.

a. Fill the reservoir and apply hydraulic pressure at the engine pump flex hose. The regulator is set to relieve at 1100 psi maximum and to reseat at 950 psi minimum (minimum allowable range is 80 psi). Check the regulator range by operating the wing flaps. Minimum time interval between consecutive reseatings must be 30 minutes.

b. Release the DOWN latches on the gear and remove the safety pins. Operate all units slowly at first to check clearances; then operate at full speed.

Note

The shock struts must be in the fully extended position before retracting the alighting gear.

3-68. HYDRAULIC SYSTEM TUBING TORQUE VALUES. The following charts list the standard torque values for tubing and fitting nuts.

HYDRAULIC SYSTEM TUBING TORQUE VALUE CHART

Fitting Nut Installation

Tube	*Steel — Black*		*Aluminum — Blue*	
OD (Inch)	*Nut (Inches)*	*Torque*	*Nut (Inches)*	*Torque*
3/16	5/8	200 inch-pounds	5/8	78 inch-pounds
1/4	11/16	300 inch-pounds	11/16	112 inch-pounds
5/16	3/4	300 inch-pounds	3/4	112 inch-pounds
3/8	13/16	400 inch-pounds	13/16	150 inch-pounds
1/2	1	700 inch-pounds	1	312 inch-pounds
5/8	1 1/4	1000 inch-pounds	1 1/4	450 inch-pounds

B-Nut Installation

Tube	** Steel — Black*		*† Aluminum — Blue*	
OD (Inch)	*Nut (Inches)*	*Torque*	*Nut (Inches)*	*Torque*
3/16	7/16	100 inch-pounds	—	28.5 inch-pounds
1/4	9/16	150 inch-pounds	9/16	50 inch-pounds
5/16	5/8	200 inch-pounds	5/8	78 inch-pounds
3/8	11/16	300 inch-pounds	11/16	112 inch-pounds
1/2	7/8	500 inch-pounds	7/8	200 inch-pounds
5/8	1	700 inch-pounds	1	312 inch-pounds
3/4	1 1/4	1000 inch-pounds	1 1/4	450 inch-pounds
1	1 1/2	1300 inch-pounds	1 1/2	600 inch-pounds

* Overtightening of 1/6-turn permissible to stop leakage after torquing steel nuts.

† No overtightening permissible on aluminum nuts.

3-69. HYDRAULIC SYSTEM BOLT TORQUE VALUES.

3-70. If structural tension bolts are not properly tightened when installed, vibration and fatigue stresses will materially shorten their service life. It is equally dangerous, however, to tighten a bolt excessively, since over-tightening may result in stripped threads, a bent or sheared bolt, or eventual tension failure. It is essential, therefore, to observe the torque limits specified here when installing tension bolts. Unless otherwise noted on the installation drawing, bolts and nuts must be installed free of lubricants. Otherwise, the specified torque reading may produce an incorrect tension load. Torque readings should not be made on bolts that show painting or corrosion, as it is impossible to determine accurately the amount of tension being applied. Bolts and nuts that are to be cotter-pinned may be torqued to the special values shown in order to permit the alignment of their holes. However, these figures must never be exceeded, nor may the nuts be backed off to align the holes, unless they are backed off sufficiently far to completely re-torque properly. When re-torquing bolts, the nut must be backed off part of a turn and then retightened to the correct value. This is necessary because more torque may be required to break the nut free than is actually on the bolt. To apply 700 inch-pounds and fail to move the nut does not imply that it torqued to 700 inch-pounds. The nut may have a torque of only 650 inch-pounds, but 750 inch-pounds may be required to break it free. A value in excess of the required torque may also be needed to break the nut free in a counterclockwise direction. For this reason, the nut should be backed off and retightened, allowing no rotation of the bolt. Whenever possible, the torque reading must be taken on the nut and not on the bolt. Torque wrenches should be calibrated frequently to insure accurate readings. When adapters are used with torque wrenches, compensating torque values must be computed for each adapter and wrench involved. The following chart lists the standard torque values for structural tension bolts and also lists the bolts and nuts which should be installed with the torque values specified. Special torques are listed for all bolts to which standard values are not applicable.

HYDRAULIC SYSTEM BOLT TORQUE VALUE TENSION CHART

Bolt Size	*AN Type Bolts*		*AN365 and AN310 Nuts*	*Torque Values*	*Cotter Pin Maximum Torque*
10-32	AN-3	AN-173	-1032	20-25 inch-pounds	40 inch-pounds
¼-28	AN-4	AN-174	-428	50-70 inch-pounds	100 inch-pounds
5/16-24	AN-5	AN-175	-524	100-140 inch-pounds	225 inch-pounds
⅜-24	AN-6	AN-176	-624	160-190 inch-pounds	390 inch-pounds
7/16-20	AN-7	AN-177	-720	450-500 inch-pounds	840 inch-pounds
½-20	AN-8	AN-178	-820	480-690 inch-pounds	1100 inch-pounds
9/16-18	AN-9	AN-179	-918	800-1000 inch-pounds	1600 inch-pounds
⅝-18	AN-10	AN-180	1018	90-105 foot-pounds	200 foot-pounds
¾-16	AN-12	AN-182	1216	190-210 foot-pounds	415 foot-pounds
⅞-14	AN-14	AN-184	1414	210-250 foot-pounds	590 foot-pounds
1-14	AN-16	AN-186	1614	310-460 foot-pounds	835 foot-pounds
1⅛-12	AN-18		1812	415-585 foot-pounds	1250 foot-pounds

Bolt Size	*NAS Heat-Treated Bolts*	*Douglas Heat-Treated Bolts*	*Elastic Stop Heat-Treated Nuts*	*Torque Values*
¼-28	NAS-144	S-2076904	12B-048	73-100 inch-pounds
5/16-24	NAS-145	S-2076905	12B-054	145-200 inch-pounds
⅜-24	NAS-146	S-2076906	12B-064	230-280 inch-pounds
7/16-20	NAS-147	S-2076907	12B-070	650-720 inch-pounds
½-20	NAS-148	S-2076908	12B-080	700-1000 inch-pounds
9/16-18	NAS-149	S-2076909	12B-098	100-120 foot-pounds
⅝-18	NAS-150	S-2076910	12B-108	135-155 foot-pounds
¾-16	NAS-152	S-2076912	12B-126	280-300 foot-pounds
⅞-14	NAS-154	S-2076914	12B-144	300-360 foot-pounds
1-14	NAS-156	S-2076916	12B-164	450-665 foot-pounds
1⅛-12	NAS-158	S-2076918	12B-182	605-845 foot-pounds

3-71. PNEUMATIC SYSTEM.

3-72. DESCRIPTION. The pneumatic system does not apply to this aircraft. For information on the air brakes, see paragraph 2-434.

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z
AIRCRAFT

SECTION IV
UTILITY SYSTEMS

THIS SECTION SUPERSEDES SECTION IV OF AN 01-40NK-2
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SECTION IV

UTILITY SYSTEMS

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SECTION IV

UTILITY SYSTEMS

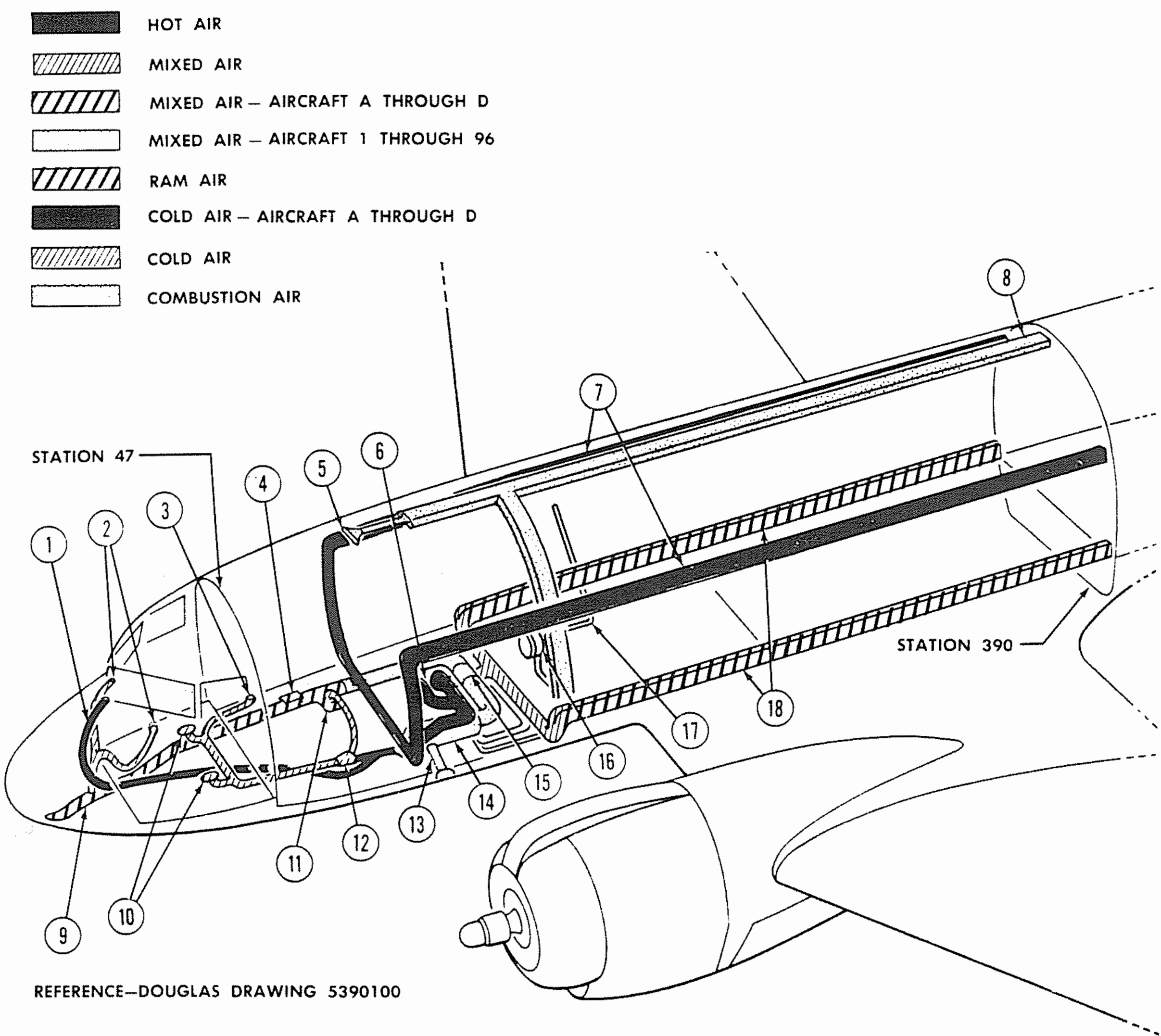
4-1. HEATING AND VENTILATING SYSTEM.

4-2. DESCRIPTION. *(See figure 4-1.)* The heating and ventilating system provides heating and ventilating air to the main cabin and to the flight compartment both during flight and while the aircraft is on the ground. It also provides heated air for anti-icing of the windshield. Hot air for the system is supplied by one 150,000-BTU combustion heater located under the fuselage floor in the heater compartment. Fuel for the operation of the heater is supplied from the right forward tank of the main aircraft fuel system. Ram air for ventilating the aircraft and combustion air for operation of the heater is obtained from a duct installed in the nose of the aircraft. An air intake check valve is installed in the ram air duct to prevent the escape of air when the ground blower is operating and also to provide a means of manually shutting off the supply of ram air to the main cabin. Heating and ventilating the aircraft is manually controlled by four controls: the master heater ON-OFF switch, the cabin temperature manual control, the flight compartment temperature manual control, and the windshield temperature manual control *(see figure 4-2)*. These controls are connected by mechanical linkage to valves in the ducting system of the aircraft whereby the desired proportions of hot or cold air can be mixed to heat or cool the aircraft. A temperature indicator and heater-inoperative light are installed in the flight compartment on the main instrument panel. Fire indicators for the heater include dual warning lights in the flight compartment and fire detector switches in the heater ducting and in the heater compartment. Fire protection for the heater is furnished by the main CO_2 system for the aircraft. Air for heating or ventilating is supplied for ground operation by a ground blower installation.

4-3. MINOR REPAIR AND REPLACEMENT OF HEATING AND VENTILATING SYSTEM COMPONENTS. No minor repair of the heating and ventilating system components is recommended. If, upon careful inspection, a component of the system is malfunctioning, the unit should be removed from the aircraft and replaced with a new or overhauled unit.

4-4. TROUBLE SHOOTING OF HEATING AND VENTILATING SYSTEM.

Trouble	*Probable Cause*	*Remedy*
a. Heater inoperative (heater-inoperative warning light ON).	Intake duct check valve manual override control not fully open.	Move control to open position.
	Overheat drop-out fuse blown.	Check for cause and replace fuse. Use three-ampere fuse only.
	Heater cycling switch stuck closed.	Replace switch.
	Drop-out switch stuck closed.	Replace switch.
	One or both switches changed temperature setting.	Replace switch.
	Fuel cycling solenoids in heater fuel control container stuck OPEN.	Replace heater fuel control assembly.
b. Heater inoperative (heater-inoperative warning light OFF).	Heater cycling switch stuck OPEN.	Replace switch.
	Heater cycling solenoid valve in heater fuel control assembly stuck CLOSED.	Replace heater fuel control assembly.
	Defective vibrator or coil in ignition unit.	To check vibrator, pull switch on ignition unit out to switch to alternate contacts. If coil is defective, replace unit.
	Defective spark plug in heater.	Replace spark plug.
	Main fuel shutoff solenoid valve stuck CLOSED.	Replace valve.
	Fuel pump in heater fuel control assembly inoperative.	Replace heater fuel control assembly.



1. Windshield De-Icing Installation
2. Cockpit Cold Air Outlet
3. Radioman's Mixed Air Duct
4. Intake Air Duct Check Valve
5. Navigator's Window Mixed Air Outlet
6. Hot Air Dump Valve
7. Cabin Cold Air Duct
8. Cabin Mixed Air Duct Installation
9. Airscoop
10. Cockpit Mixed Air Outlet
11. Ground Blower
12. Cockpit Air Mixing Valve
13. Combustion Air Exhaust Duct
14. Combustion Heater
15. Cabin Air Mixing Valve
16. Combustion Heater Fuel Control Assembly
17. Heater Fuel Supply Line
18. Cabin Mixed Air Duct

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Figure 4-1. Heating and Ventilating System

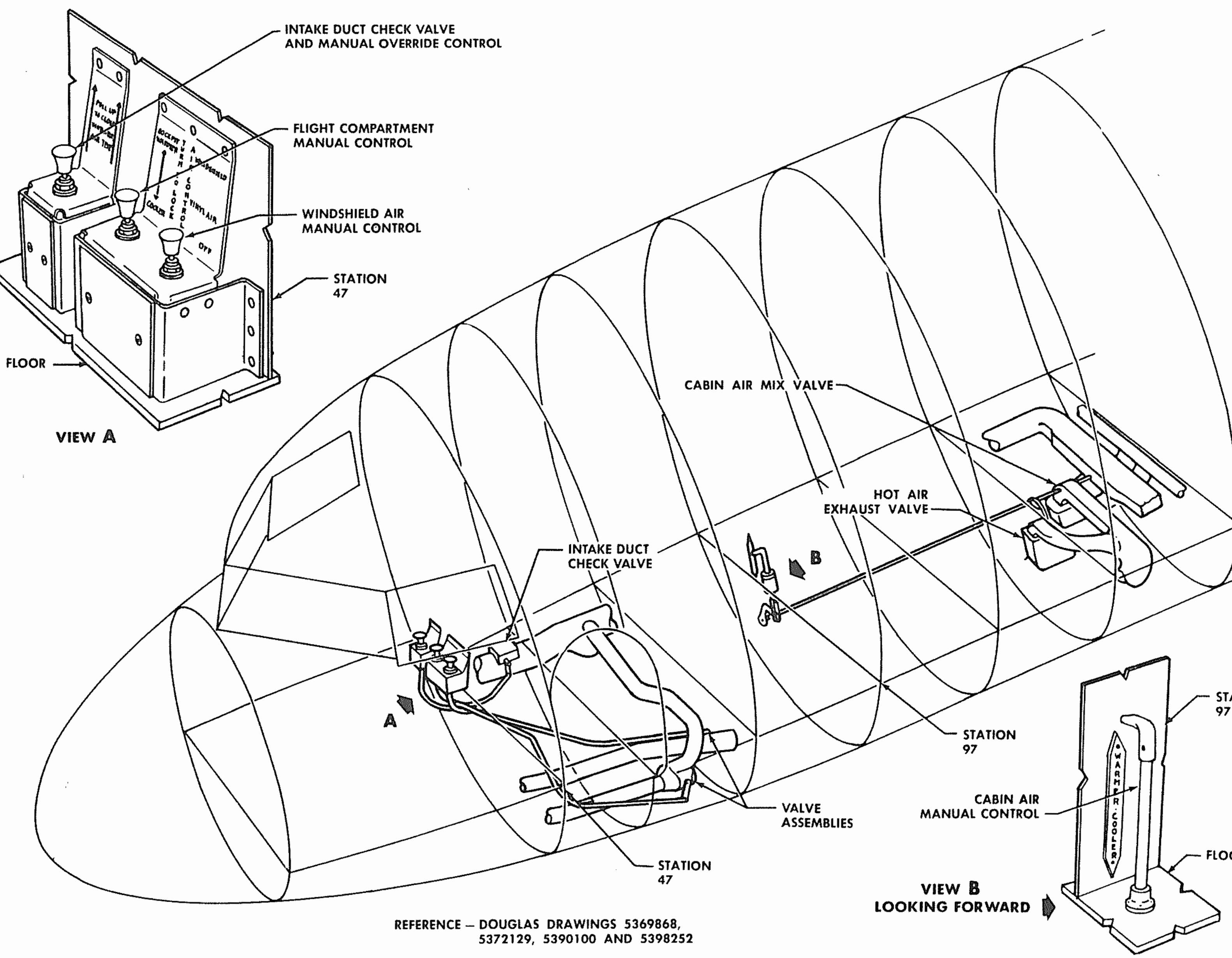


Figure 4-2. Heating and Ventilating Control System

Trouble	Probable Cause	Remedy
b. (*Continued*)	Ruptured diaphragm in fuel pressure regulator.	Replace heater fuel control assembly.
	Heater control circuit breaker tripped.	Reset circuit breaker.
	Defective heater slave relay.	Replace relay.
	Defective master heater switch.	Replace switch.
	Heater fuel supply depleted.	Replenish fuel supply.
c. Heater backfires.	Heater fuel nozzle plugged.	Remove and clean nozzle.
	Ruptured diaphragm in fuel pressure regulator.	Replace heater fuel control assembly.
d. Ground blower inoperative. Aircraft engines running above generator cut-in speed.	Ground power relay defective.	Replace relay.
	Generator circuit relays in heater circuit defective.	Replace defective relay.
	Ground-flight transfer relay defective.	Replace defective relay.
	Right main alighting gear oleo switch defective.	Replace switch.
e. Ground blower inoperative. Ground power source connected to aircraft.	Selector switch not positioned to GRD. PWR.	Position switch to GRD. PWR.
	Ground-flight transfer relay defective.	Replace relay.
	Ground power transfer relay defective.	Replace relay.
	Ground blower relay defective.	Replace relay.
	Right main alighting gear oleo switch defective.	Replace switch.
	Ground blower circuit breaker tripped.	Reset circuit breaker.
	Ground control circuit breaker tripped.	Reset circuit breaker.
	Ground blower motor defective.	Replace ground blower.

4-5. TEST OF HEATING AND VENTILATING SYSTEM.

Note

In all tests, do not permit the heater temperature to go above 180°C (356°F) as indicated on the heater temperature indicator on the pilot's main instrument panel, or damage to the heater or ducts may result.

a. The following steps should be taken to prepare the aircraft for test of the heating and ventilating system:

All four heater system circuit breakers (ground control, ground blower, heater control and inverter warning lights) on the main circuit breaker panel must be OPEN.

The fuse must be removed from the drop-out fuse clips.

The ground power supply should be plugged in.

The master battery switch must be set to aircraft's battery.

The gear oleo switches must be set in normal position for the aircraft while on the ground.

Remove the plug from the receptacle on the heater main fuel valve.

Remove the plug from the receptacle on the heater can.

The ventilating air intake valve must be closed manually.

b. After the above operations are performed, the entire heater system including the ground blower, should be inoperative.

c. Close the ground blower circuit breaker (70 ampere). Nothing should come on. Operating the heater inoperative indicator light to the test position should not light the indicator.

d. Close the heater control circuit breaker (10 ampere); the heater inoperative indicator light should now illuminate in the test position. Nothing else should change.

e. Close the inverter warning light circuit breaker and set the master battery switch to ground power supply position. The ground blower should operate.

f. Operate the right gear oleo switch to simulate the setting in flight. The ground blower should stop. Now close the master heater switch and check to be sure that the heater inoperative indicator light illuminates when the oleo switches are in the flight position, and the light is off when the oleo switches are in the ground position. Return the oleo switches to the normal ground position.

g. Set the master battery switch to aircraft's battery and the ground blower will stop. Return to ground power, start the engines and bring up to 1200 rpm. Turn the master battery switch to AIRCRAFT BATTERY. Turn on either generator and check to be sure that the ground blower will not come on with just one generator on. Turn on both generators and make certain that the ground blower operates. Shut down the engines.

h. Put the master battery switch in the ground power position and close the ground control circuit breaker (10 ampere) and heater master switch. The ground blower should be on and the heater inoperative light should be illuminated.

i. Open the master heater switch and install the heater drop-out fuse. Close the master heater switch; the heater inoperative light should illuminate. Open the ventilating air valve control fully; the heater inoperative light should go out.

j. If the oleo switches are set to flight condition so that the ground blower stops, the heater ignition vibrator and the spark in the heater should be audible. Set the oleo switches back to ground position.

k. Check for 28 volts on the heater main fuel valve plug and on pins A, B, and C of the heater can plug.

l. Set the heater control system as follows:

The master heater switch must be OFF.

All four heater circuit breakers must be CLOSED.

Reconnect all heater system electrical plugs in their receptacles.

Return the gear oleo switches to their normal ground condition.

Place the master battery switch in the ground power position.

Install the drop-out fuse.

The ventilating air intake valve control must be fully open.

The heater is now ready for operation.

m. After the above conditions have been complied with, ground operation check of the heaters may be accomplished by proceeding as follows:

n. Set the flight compartment heat control in the cold position; the windshield heat control in the off position; and the cabin heat control in the full cold position. This will cause all the heated air to be dumped overboard.

Note

In all the tests that follow, do not permit the heater temperature to go above 180°C (350°F) as indicated on the heater temperature indicator on the pilot's main instrument panel, or damage to the heater or ducts may result.

o. Have a man stationed where he safely may watch the heater exhaust for indications of combustion or backfire. Turn the master heater switch on and watch the heater temperature indicator. Continue heater operation and watch for stabilization, which should occur around 120°C to 130°C (248°F to 266°F). Note the stabilization temperature, which is the operating point of the heater cycling switches. Shut off the heater and make certain that the ground blower continues to operate in order to cool the heater and the control switches.

p. Connect a jumper across the terminals of the heater cycling switch, thus shortcircuiting it. Providing that the jumper is connected in such a manner as to shortcircuit the cycling switch, any alternative point of attaching it may be used.

q. After the ground blower has been cooling the heater and temperature control switches for five minutes, proceed with the test. Turn on the master heater switch and watch the heater temperature indicator and the heater inoperative indicator light. Note the temperature at which the indicator light illuminates. The indicator light illuminating indicates that the heater is off and that the drop-out switch has blown the drop-out fuse. The temperature should be less than 176°C (350°F).

r. Restore the system to normal by removing the jumper installed in step p, preceding, and replace the blown drop-out fuse, using only a three-ampere fuse.

4-6. COMBUSTION AND VENTILATING AIR INTAKE SCOOP. *(See figure 4-1.)* Ram air for ventilating the aircraft is supplied by a scoop located in the nose of the aircraft. A portion of this air also is used as combustion air when the heater is operating. The ram air enters the scoop and is ducted aft to interconnecting ducts through which it can be used to ventilate the flight compartment, the main cabin, or can be used as ventilating air and combustion air for operation of the combustion heater. The scoop is constructed of aluminum and, on aircraft A, B, and 1 through 62, incorporates an electric Calrod-type heating element. The airscoop heating element, which can be energized only during flight, is put in operation by turning the scoop heater switch to the ON position.

4-7. REMOVAL OF COMBUSTION AND VENTILATING AIR INTAKE SCOOP.

a. Remove the clamps connecting the scoop to the air intake duct and to the flight compartment cold air ducts.

b. Disconnect the wires attached to the heating element if installed.

c. Remove the screws attaching the scoop to the fuselage skin and structure.

d. Pull the scoop out through the hole in the fuselage skin.

4-8. INSTALLATION OF COMBUSTION AND VENTILATING AIR INTAKE SCOOP. Reverse the removal procedure.

4-9. FLIGHT COMPARTMENT COLD AIR DUCT AND OUTLETS. On aircraft factory serial numbers 43301 through 43400, individual cold air valves (outlets) are installed in the flight compartment for the delivery of cold air to the pilot and co-pilot. On aircraft A, B, C, and D, similar outlets also are located adjacent to passenger seats. The cold air outlets in the flight compartment are eyeball-type valves with a 90-degree on-off, thumb-and-finger control on the unit. Air is supplied to the cold air outlets when the aircraft is in flight by ram air through the intake scoop.

4-10. COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE AND HEATER SHUTOFF SWITCH. *(See figure 4-2.)* The flapper-type combustion and ventilating air intake duct check valve is installed immediately forward of station 64. The valve is open whenever ram air is entering the intake scoop. The valve will remain closed when there is no ram air passing through the intake duct, thus preventing the reverse flow of air from the ground blower out through the intake duct. This valve can be manually closed in flight, by the manual override control located in the flight compartment, to shut off the flow of ram air from the heating and ventilating system of the aircraft. Operation of the manual control to close the check valve will automatically shut off the heater by means of a switch installed on the valve. The switch is actuated by the mechanical linkage of the valve control.

4-11. REMOVAL OF COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE AND HEATER SHUTOFF SWITCH.

a. Disconnect the wire leads to the heater shutoff switch installed on the inboard side of the valve.

b. Remove the screws attaching the switch to the valve.

c. Disconnect the mechanical linkage from the valve manual control at the valve.

d. Remove the clamps connecting the valve to the air duct.

e. Disconnect the drain pipe from the bottom of the valve.

f. Remove the screws and nuts securing the valve to the structure.

4-12 TEST OF COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE BEFORE INSTALLATION. With the valve in normal position, apply air pressure at six inches H_2O to the outlet port of the valve. The allowable leakage is 10 cfm.

4-13. INSTALLATION OF COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE AND HEATER SHUTOFF SWITCH. Reverse the removal procedure, except as follows:

a. Before connecting the mechanical linkage between the valve and the manual override control in the flight compartment, be sure that the flapper in the valve is held fully closed by the lever and that the control in the flight compartment is in the closed position.

b. When the heater shutoff switch is installed, care should be exercised to insure that the switch will be actuated to shut off the heater when the override manual control in the flight compartment is moved slightly away from the full OPEN position.

4-14. COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE MANUAL OVERRIDE CONTROLS. *(See figure 4-3.)* The intake duct check valve manual override control is a push-pull-type and is located on the lower forward side of the bulkhead at station 47, aft of the co-pilot's seat. The control is mechanically connected to the flapper-type check valve installed in the ram air intake duct and is used to override manually the action of the check valve and reduce the amount of air entering the heating and ventilating system. This control normally is left in the full OPEN position. When the control is moved out of the full OPEN position, a micro switch is actuated by the mechanical linkage between the control and the valve, and will electrically shut off the heater.

4-15. COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE MANUAL OVERRIDE CONTROL CABLE. *(See figure 4-3.)* The combustion and ventilating air intake duct check valve manual override control cable extends from the push-pull unit, located aft of the co-pilot's seat, through a conduit, downward to the ram air intake duct check valve.

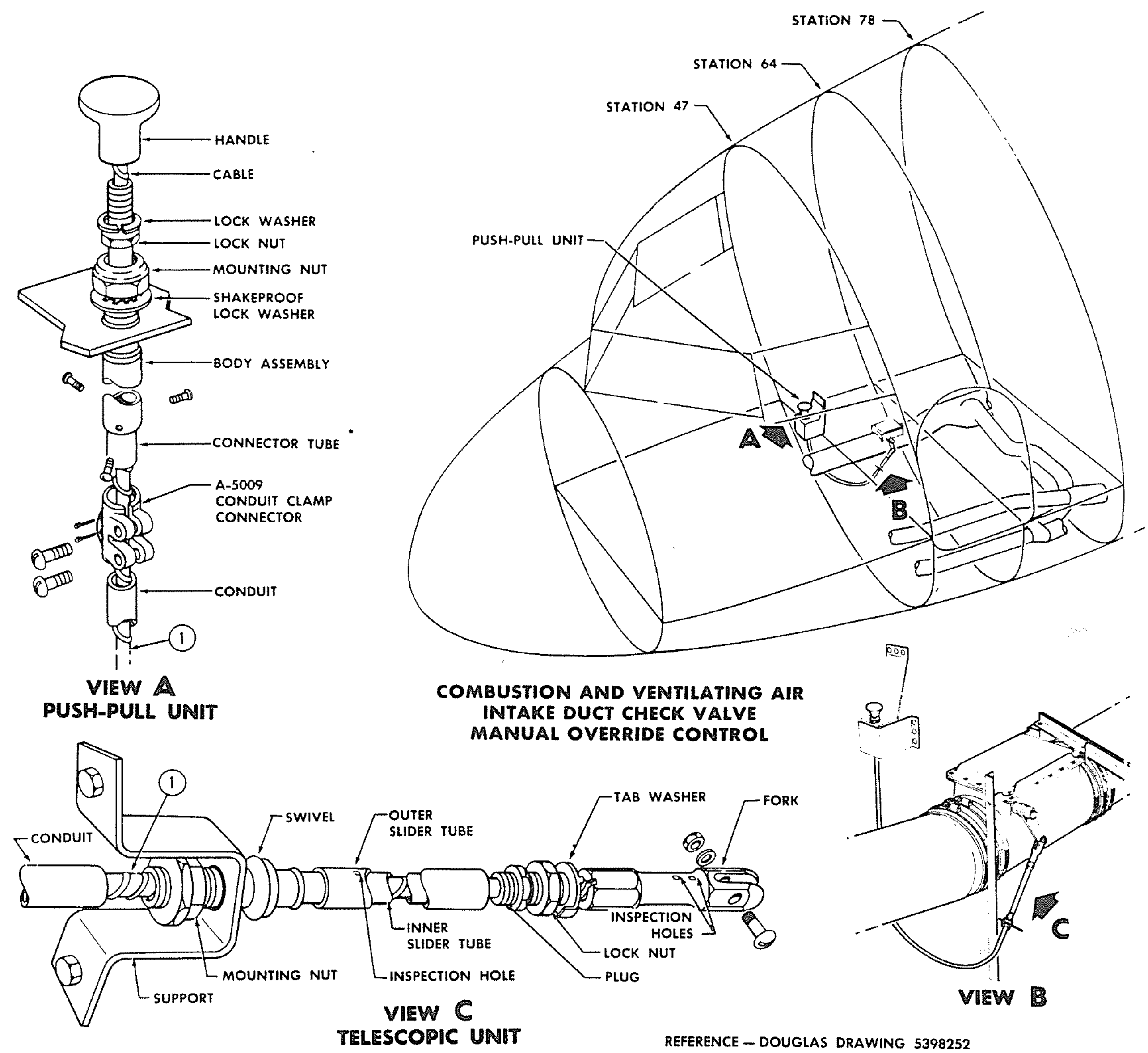
4-16. REMOVAL OF COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE MANUAL OVERRIDE CONTROL CABLE.

a. Remove the floor panel in the aisle adjacent to the hydraulic control panel.

b. Loosen the locknut at the mixing valve end of the control cable and unscrew the telescoping barrel from the clevis end.

c. Remove the spring from the end of the spiral control cable. The telescoping barrel then will be free and can be removed. Thread the cable end for removal.

d. Loosen the locknut at the handle end of the control and screw the handle off the barrel. The locknut and handle both have left threads.



INTAKE DUCT CHECK VALVE MANUAL OVERRIDE CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEM. DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH	CABLE SIZE	FITTINGS 1	2	3	4	5
1	5398252-21 (A5222 Teleflex)	1		52	.187 dia.	XA346A-14	5372512-19 (A-5023-340 Teleflex)	SA167A-30		

CABLE ASSEMBLY

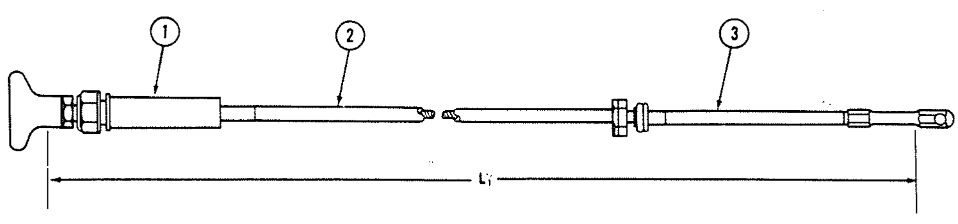


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Figure 4-3. Combustion and Ventilating Air Intake Duct Check Valve Manual Override Control System

e. Remove the spring from the handle end of the spiral control cable and then pull the control cable out of the housing.

4-17. INSTALLATION OF COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE MANUAL OVERRIDE CONTROL CABLE. Before reinstalling the cable, it should be thoroughly lubricated with Douglas Lubricant B (Union Oil Co.—Strona LT-1; Standard of California—RPM Aviation Grease No. 1; Shell Oil Co.—Aeroshell No. 6, or Esso Aviation—All Purpose Grease No. 6).

a. Pull the spiral control cable through the telescoping barrel between the check valve and the control handle.

b. Install the spring on the valve end of the control cable. Screw the telescoping barrel onto the clevis end and tighten the locknut.

c. Install the spring on the handle end of the control cable and screw the handle onto the barrel.

d. Adjust the combustion and ventilating air intake duct check valve manual override controls (see paragraph 4-18)..

e. Replace the floor panel.

4-18. ADJUSTMENT OF COMBUSTION AND VENTILATING AIR INTAKE DUCT CHECK VALVE MANUAL OVERRIDE CONTROLS. Adjust the combustion and ventilating air intake duct check valve manual override control cable so that the travel is stopped by the closing of the valve rather than by the bottoming of the handle. Tighten the locknut. The locknut and handle both have left threads.

4-19. COMBUSTION HEATER ASSEMBLY. *(See figure 4-4.)* The combustion heater installed in the heating and ventilating system is located beneath the fuselage floor between stations 97 and 138. It has a rated output of 150,000 BTU per hour. The heater supplies hot air for heating the main cabin and the flight compartment, and for anti-icing the windshield. Ventilation and combustion air for the heater is supplied from the ram air intake duct. The heater is controlled by a master heater ON-OFF switch located on the pilot's overhead electrical panel. In operation, a cycling switch will cycle the heater ON and OFF automatically at 130°C (265°F) to prevent overheating. The heater is further protected from overheating by a drop-out switch that will close at a set temperature and blow a fuse, which in turn shuts off all power and fuel to the heater. The fuel supply for the heater is taken from the right forward tank of the aircraft fuel system. Exhaust gases from the heater combustion chamber are ducted overboard through the lower part of the fuselage.

4-20. REMOVAL OF COMBUSTION HEATER ASSEMBLY.

a. Disconnect the antenna.

b. Open the access door.

c. Remove the lead to the heater spark plug.

d. Disconnect the heater fuel pipe and joint drain pipe at the heater.

e. Disconnect the heater overboard drain pipe at the heater.

f. Remove the clamps that connect the heater intake and discharge ventilating air ducts, the heater combustion air duct, and the heater exhaust shroud and duct.

g. Disconnect the metal straps that secure the heater assembly in position. When the straps are removed, care should be exercised to prevent dropping or jolting the heater.

4-21. INSTALLATION OF COMBUSTION HEATER ASSEMBLY. Reverse the removal procedure, replacing seals if new ones are available. For the test of the heater assembly after installation, see paragraph 4-5, preceding.

4-22. CABIN AIR MIXING VALVE ASSEMBLY. *(See figure 4-5.)* A cabin air mixing valve, installed in the heating and ventilating system, is located below the fuselage floor at station 138.5. The purpose of the mixing valve is to mix the proper proportions of hot air from the combustion heater with cold air from the cold air intake duct, so as to provide the desired temperature in the cabin. The mixing valve assembly incorporates two ports, with a butterfly valve in each port. The butterfly valves are connected by mechanical linkage to the cabin air manual control located in the cabin. Operation of the control positions the butterfly valves in the ports of the mixing valve assembly so that one port closes as the other is opened. When the cabin air manual control is positioned to full COLD, port A *(see figure 4-5)* of the mixing valve will be full open and port B will be completely closed. If the heater is operating, the hot air from the heater then will be exhausted overboard through a hot-air exhaust valve. If the cabin air manual control is positioned to the opposite extreme, port A of the mixing valve will be fully closed and port B will be full open, admitting only hot air to the cabin. Intermediate positioning of the control will permit various mixtures of hot and cold air to flow through the mixing valve to provide the desired cabin temperature.

4-23. REMOVAL OF CABIN AIR MIXING VALVE ASSEMBLY.

a. Remove the second floor panel aft of the forward cabin bulkhead on the right side of the aircraft.

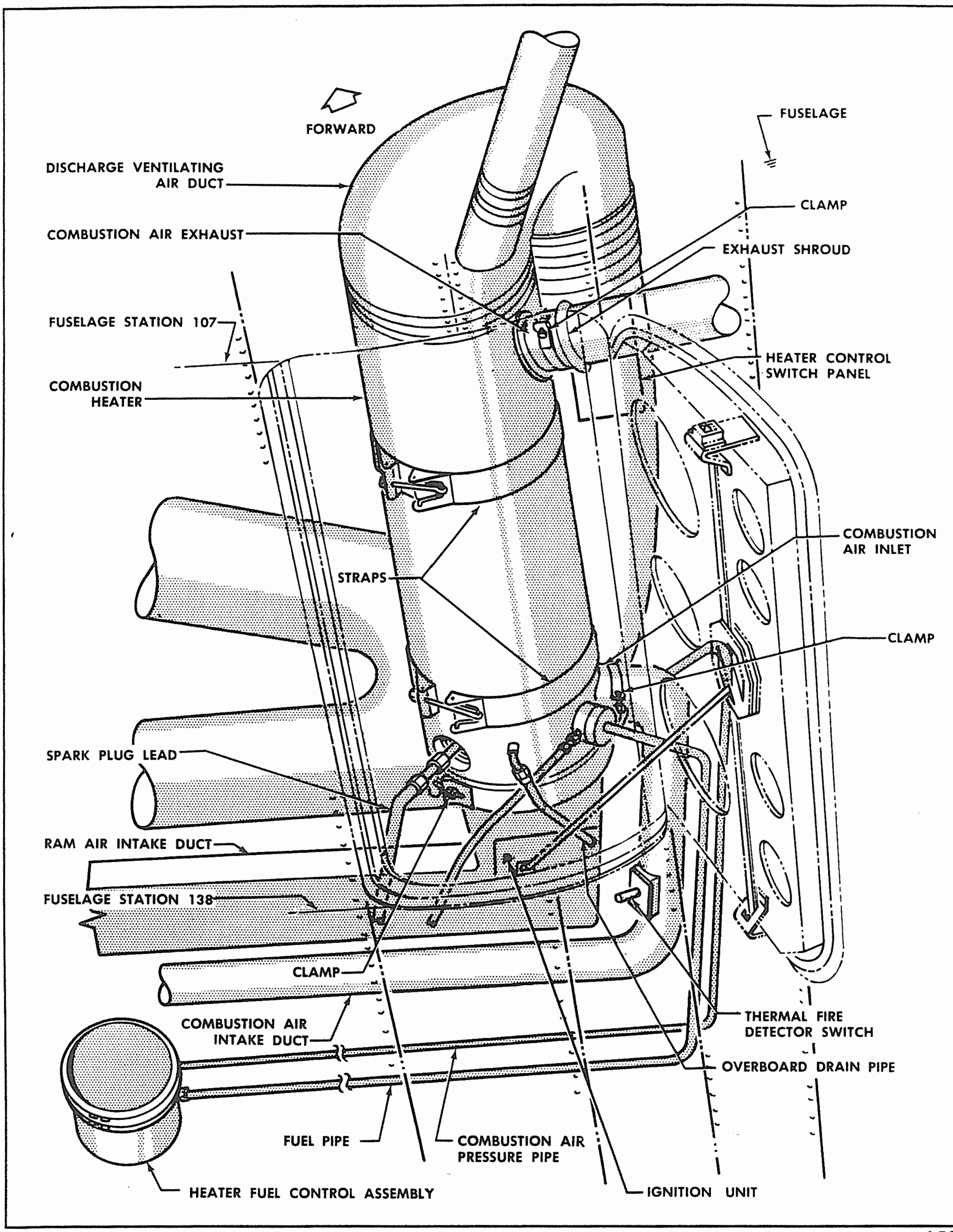


Figure 4-4. Combustion Heater Assembly

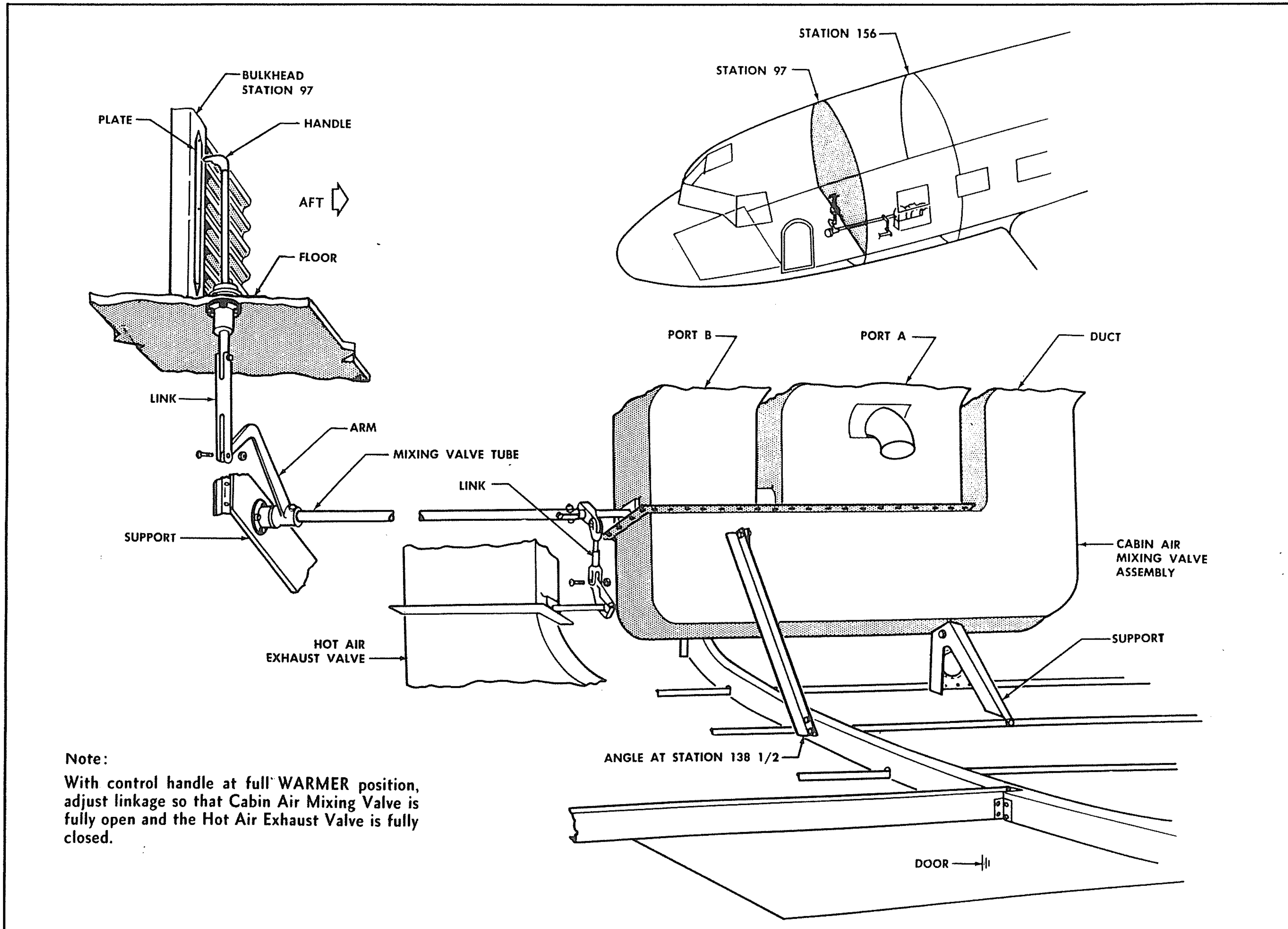


Figure 4-5. Cabin Air Mixing Valve Assembly and Manual Control

b. Place the cabin air manual control in the cold position, and then remove the tube from the serrated shaft of the mixing valve.

c. Disconnect the linkage between the mixing valve and the hot air exhaust valve assembly.

d. Remove the clamps from the mixing valve outlet duct (three places), and remove the duct tee.

e. Remove the clamps from the hot and cold air ducts that connect to the top of the mixing valve.

f. Remove the bolts and nuts that connect the mixing valve to the supporting brackets. The mixing valve can then be removed through the fuselage floor.

4-24. INSTALLATION OF CABIN AIR MIXING VALVE ASSEMBLY. Reverse the removal procedure, with the following additional step: After connecting the linkage between the mixing valve and the hot air exhaust valve, check the installation to be sure that the hot air exhaust valve is open when the hot air damper of the mixing valve assembly is closed.

4-25. CABIN AIR MANUAL CONTROL. *(See figure 4-5.)* The cabin air manual control is a push-pull type control and is located on the floor of the main cabin aft of the bulkhead at station 97. Operation of this control will actuate the cabin air mixing valve and hot air exhaust valve to vary the proportions of hot and cold air to be admitted to the main cabin. When the cabin air manual control is in the full WARMER position, the maximum amount of hot air will be delivered to the cabin. When the cabin air manual control is in the full COOLER position, the maximum amount of cold air will be directed to the cabin for ventilating. Intermediate positioning of the control handle will mix hot and cold air to provide the desired temperature in the cabin.

4-26. REMOVAL OF CABIN AIR MANUAL CONTROL.

a. Pull the control handle to the full WARMER position.

b. Remove the bolt and nut that secure the control handle to the linkage beneath the floor, and remove the handle.

c. Remove the floor panel immediately aft of the forward cabin bulkhead, through which the control handle projects.

d. Remove the bolts connecting the arm to the tube. Loosen the clamp at the aft end of the tube. The tube will then slide forward and can be disengaged from the serrated mixing valve shaft at the bracket aft of station 97 and from the serrated shaft of the mixing valve.

e. Do not alter the link connecting the mixing valve and the hot air exhaust valve.

4-27. INSTALLATION OF CABIN AIR MANUAL CONTROL.

a. Set the mixing valve to the full WARMER position.

b. Connect the mixing valve tube to the serrated shaft of the mixing valve and the serrated mixing valve shaft at the bracket aft of station 97.

c. Tighten the clamp at the aft end of the tube and install the bolts connecting the linkage arm to the tube.

d. Replace the floor panel aft of the forward cabin bulkhead.

e. Connect the control handle to the linkage with the bolt and nut.

4-28. ADJUSTMENT OF CABIN AIR MANUAL CONTROL AFTER INSTALLATION. Adjust the linkage so that the hot port of the cabin air mixing valve is fully open when the hot air exhaust valve is fully closed and the control indicates WARMER position.

4-29. FLIGHT COMPARTMENT AIR MIXING VALVE. *(See figure 4-6.)* A flight compartment air mixing valve assembly is installed in the ducting below the fuselage floor at approximately station 64. A hot air duct from the heater is connected to one port of the mixing valve, and a cold air duct from the cold air intake duct is connected into the second port of the mixing valve. A butterfly-type valve is installed in each port of the mixing valve assembly, and they are interconnected. The butterfly valves are connected by mechanical linkage to the flight compartment air manual control. The operation of the flight compartment air mixing valve assembly is the same as the cabin air mixing valve assembly, as described in paragraph 4-22 preceding, except that there is no interconnected heat exhaust valve.

4-30. REMOVAL OF FLIGHT COMPARTMENT AIR MIXING VALVE.

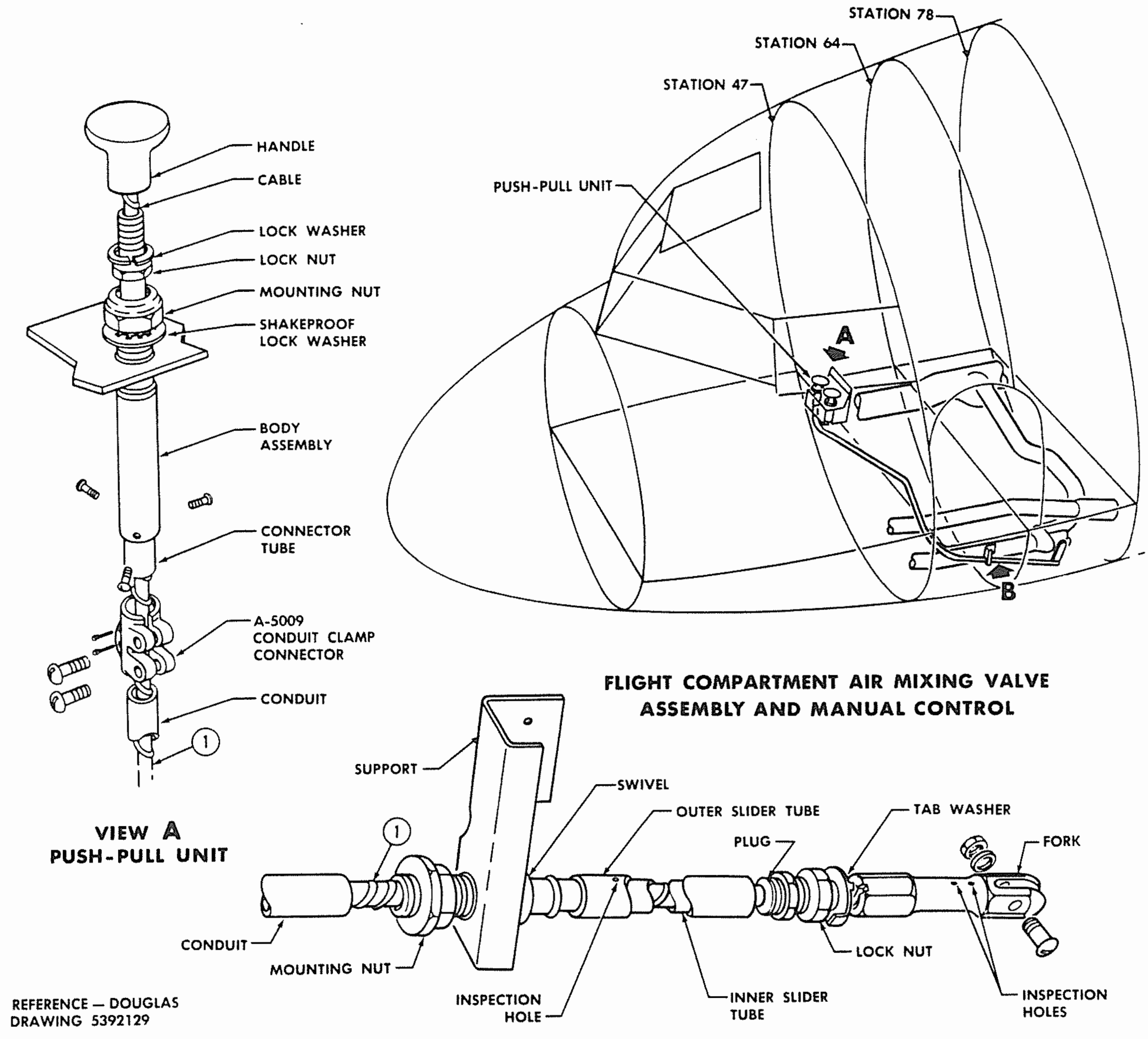
a. Disconnect the mechanical control linkage from the mixing valve.

b. Remove the clamps that secure the hot air duct, the cold air duct, and the mixing valve discharge duct to the mixing valve assembly.

c. Remove the screws that secure the mixing valve assembly to the floor beams, and remove the mixing valve assembly.

4-31. INSTALLATION OF FLIGHT COMPARTMENT AIR MIXING VALVE. Reverse the removal procedure.

4-32. FLIGHT COMPARTMENT AIR MANUAL CONTROLS. *(See figure 4-6.)* The flight compartment air manual control is a push-pull type and is installed in a box assembly located on the flight compartment floor behind and to the left of the co-pilot's seat. The control is connected by mechanical linkage to a mixing



FLIGHT COMPARTMENT AIR MIXING VALVE MANUAL CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQD.	TYPE	CABLE LENGTH L_1	CABLE LENGTH L_2	CABLE SIZE	FITTINGS 1	FITTINGS 2	FITTINGS 3	FITTINGS 4	FITTINGS 5
1	5372129-9 (A5222 Teleflex)	1		93		.187 dia.	XA346A-15	5372129-13 (A5023-780 Teleflex)	SA167A-32		

CABLE ASSEMBLY

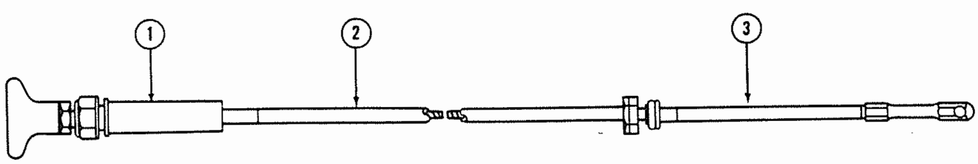


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Figure 4-6. Flight Compartment Air Mixing Valve Manual Control System

valve installed in the ducting for the flight compartment air. Operation of the control will vary the proportion of hot and cold air to be admitted to the flight compartment through the mixing valve.

4-33. FLIGHT COMPARTMENT AIR MANUAL CONTROL CABLE. *(See figure 4-6.)* The flight compartment air manual control cable extends from the push-pull control aft of the co-pilot's seat, through a conduit, downward to the flight compartment air mixing valve.

4-34. REMOVAL OF FLIGHT COMPARTMENT AIR MANUAL CONTROL CABLE.

a. Remove the floor panel at the flight compartment main entrance door.

b. Loosen the locknut at the mixing valve end of the control cable and unscrew the telescoping barrel from the clevis end.

c. Remove the spring from the end of the spiral control cable. The telescoping barrel will then be free and can be removed. Thread the cable for removal.

d. Loosen the locknut at the handle end of the control and screw the handle off the barrel. The locknut and handle both have left threads.

e. Remove the spring from the handle end of the spiral control cable and then pull the control cable out of the housing.

4-35. INSTALLATION OF FLIGHT COMPARTMENT AIR MANUAL CONTROL CABLE. Before installing the cable, it should be thoroughly lubricated with Douglas Lubricant B (Union Oil Co.—Strona LT-1; Standard of California—RPM Aviation Grease No. 1; Shell Oil Co.—Aeroshell No. 6; or Esso Aviation Co.—All Purpose Grease No. 6).

a. Pull the spiral control cable through the telescoping barrel between the mixing valve and the control handle.

b. Install the spring on the valve end of the cable, screw the telescoping barrel onto the clevis end, and tighten the locknut.

c. Install the spring on the handle end of the control cable and screw the handle onto the barrel.

d. Adjust the flight compartment air manual controls (see paragraph 4-36).

e. Replace the floor panel.

4-36. ADJUSTMENT OF FLIGHT COMPARTMENT AIR MANUAL CONTROLS. Adjust the flight compartment air manual control cable so that the travel is stopped by the closing of the valve rather than by the bottoming of the handle. Tighten the locknut. The locknut and handle both have left threads.

4-37. WINDSHIELD VOLUME CONTROL VALVE ASSEMBLY. *(See figure 4-7.)* A butterfly-type windshield volume control valve installed in the hot-air duct to the windshield is located under the fuselage floor at approximately station 78. The valve is connected by mechanical linkage to the windshield volume control in the flight compartment. Operation of the control varies the amount of hot air to be directed to the windshield for vinyl and anti-icing warming.

4-38. REMOVAL OF WINDSHIELD VOLUME CONTROL VALVE ASSEMBLY.

a. Remove the floor panel at the flight compartment main entrance door.

b. Disconnect the valve control linkage from the valve lever.

c. Disconnect the clamps securing the duct to each end of the valve assembly.

d. Remove the valve assembly.

4-39. INSTALLATION OF WINDSHIELD VOLUME CONTROL VALVE ASSEMBLY. Reverse the removal procedure.

4-40. WINDSHIELD AIR MANUAL CONTROLS. *(See figure 4-7.)* The windshield air manual control is a push-pull-type control located in the same box assembly as the flight compartment air manual control on the floor to the left and rear of the co-pilot's seat. This control is connected by mechanical linkage to a butterfly-type valve installed in the windshield air duct. Operation of the control regulates the volume of hot air that is admitted to the windshield for vinyl and anti-icing warming.

4-41. WINDSHIELD AIR MANUAL CONTROL CABLE. *(See figure 4-7.)* The windshield air manual control cable extends from the push-pull control aft of the co-pilot's seat, downward through a conduit, to the windshield volume control valve.

4-42. REMOVAL OF WINDSHIELD AIR MANUAL CONTROL CABLE.

a. Remove the floor panel at the flight compartment main entrance door.

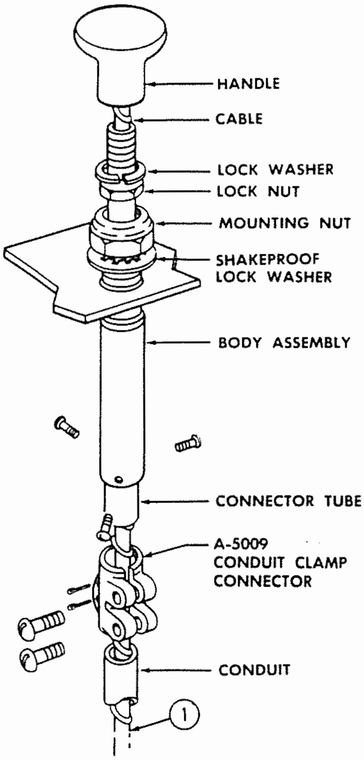
b. Loosen the locknut at the mixing valve end of the control cable and unscrew the telescoping barrel from the clevis end.

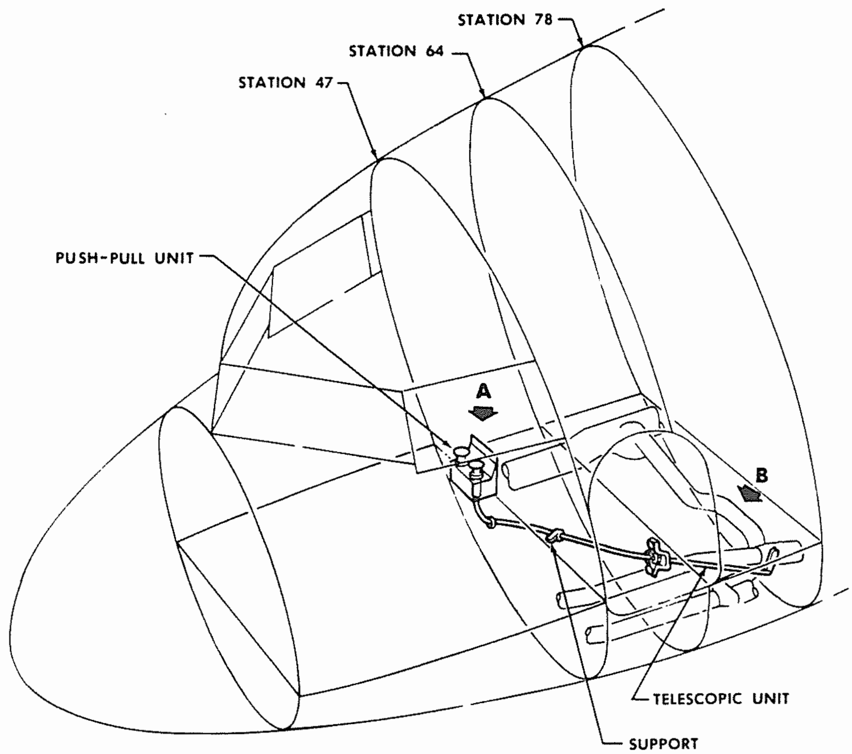
c. Remove the spring from the end of the spiral control cable. The telescoping barrel will then be free and can be removed.

d. Loosen the locknut at the handle end of the control and unscrew the handle from the barrel. The locknut and handle both have left threads.

e. Remove the spring from the handle end of the spiral control cable and then pull the control cable out of the housing.

4-43. INSTALLATION OF WINDSHIELD AIR MANUAL CONTROL CABLE. Before installing the cable, it should be thoroughly lubricated with Douglas Lubricant B (Union Oil Co.—Strona LT-1; Standard

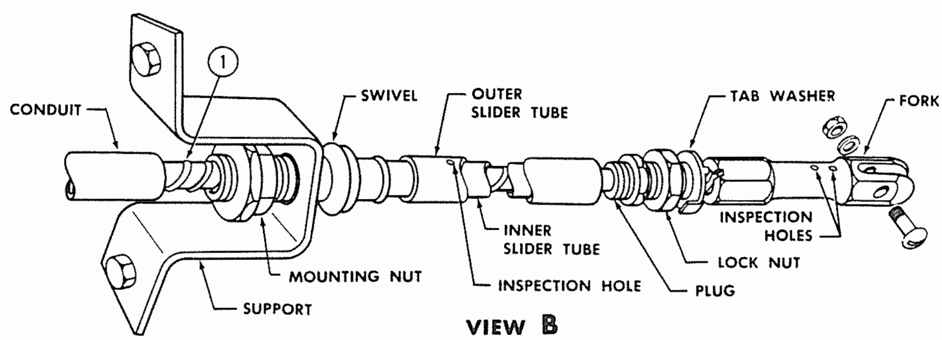




WINDSHIELD VOLUME CONTROL VALVE ASSEMBLY AND MANUAL CONTROL

VIEW A
PUSH-PULL UNIT

REFERENCE — DOUGLAS DRAWING 5372129



VIEW B
TELESCOPIC UNIT

WINDSHIELD VOLUME CONTROL VALVE MANUAL CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQD.	TYPE	CABLE LENGTH L_1	L_2	CABLE SIZE	FITTINGS 1	2	3	4	5
1	5372129-7 (A-5222 Teleflex)	1		74.3		.187 dia.	XA346A-15	5372129-11 (A-5023-580 Teleflex)	SA167A-32		

CABLE ASSEMBLY

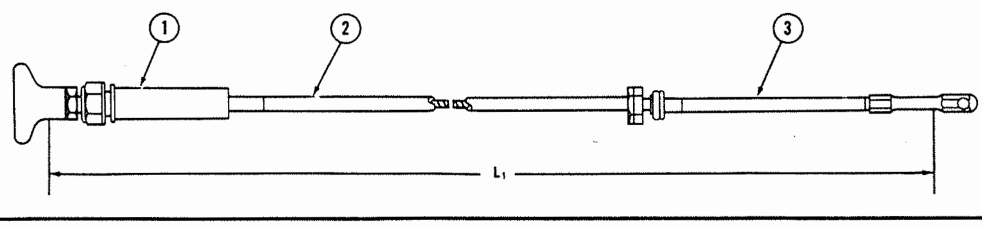


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Figure 4-7. Windshield Volume Control Valve Manual Control System

of California—RPM Aviation Grease No. 1; Shell Oil Co.—Aeroshell No. 6; or Esso Aviation Co.—All Purpose Grease No. 6).

a. Pull the spiral control cable through the telescoping barrel between the mixing valve and the control handle.

b. Install the spring on the valve end of the cable, screw the telescoping barrel onto the clevis end, and tighten the locknut.

c. Install the spring on the handle end of the control cable and screw the handle onto the barrel.

d. Adjust the windshield air manual controls (see paragraph 4-44).

e. Replace the floor panel.

4-44. ADJUSTMENT OF WINDSHIELD AIR MANUAL CONTROLS. Adjust the windshield air manual control cable so that the travel is stopped by the closing of the valve rather than by the bottoming of the handle. Tighten the locknut. The locknut and handle both have left threads.

4-45. MASTER HEATER SWITCH. A two-position master heater switch, located on the pilot's electrical switch panel, is used to turn the cabin heater ON and OFF electrically. When the switch is in the OFF position, the electrical circuit to the heater ignition is open, the solenoid fuel shutoff valve is de-energized to close the valve, and the heater fuel pump and ignition unit are inoperative.

4-46. REMOVAL OF MASTER HEATER SWITCH.

a. Remove the screws that secure the switch panel in position.

b. Disconnect the wiring from the switch and remove the screws that secure it to the switch panel.

4-47. INSTALLATION OF MASTER HEATER SWITCH. Reverse the removal procedure.

4-48. HEATER IGNITION UNIT. *(See figure 4-4.)* A combustion heater ignition unit, located aft and below the heater, supplies high-tension current to the spark plug in the heater. The unit consists of a coil, vibrator, resistor, condenser, radio noise filter, and a single-pole, double-throw switch. The unit operates on the 28-volt d-c system of the aircraft, and delivers up to 20,000 volts for the heater spark plug. The vibrator for the ignition unit has an alternate set of contacts which can be used if the first set fails. The vibrator should be operated with the vibrator contact selector switch, located on the ignition box, in the IN position for a maximum period of 500 hours, or until the first set of contacts fails. Then, by pulling the switch to the OUT position, the auxiliary vibrator contacts can be used. The vibrator contact switch is not accessible while the aircraft is in flight.

4-49. REMOVAL OF HEATER IGNITION UNIT.

a. Disconnect the lead from the ignition unit to the heater spark plug at the spark plug.

b. Break the electrical connection to the ignition unit at the terminal.

c. Remove the bolts which secure the ignition unit to the mounting bracket.

4-50. TEST OF HEATER IGNITION UNIT BEFORE INSTALLATION.

a. Connect the ignition unit through an ammeter to 28 volts direct current. Connect the high-tension wire to a heater spark plug, or a similar spark plug with a $\frac{5}{16}$-inch spark gap.

b. A fat spark should be produced with an input current between 1.6 and 1.9 amperes.

c. Repeat the test with the vibrator switch in the alternate position.

d. Return switch to primary position.

4-51. INSTALLATION OF HEATER IGNITION UNIT. Reverse the removal procedure.

4-52. HEATING AND VENTILATING SYSTEM INDICATORS. The indicators of the heating and ventilating system include a heater temperature indicator, a heater-inoperative light, and a dual fire warning light.

4-53. The heater temperature indicator is installed on the left section of the main instrument panel. The purpose of the indicator is to provide the flight crew with visual indication of the heater discharge air temperature. The indicator is connected electrically to a thermocouple installed in the heater discharge air duct.

4-54. The heater-inoperative indicator light is installed on the left section of the main instrument panel adjacent to the heater temperature indicator. The heater-inoperative light will only be ON when the drop-out fuse is blown or when the scoop air inlet valve is manually closed.

4-55. The dual fire warning light and a warning light test switch are installed on the left overhead electrical panel. The warning lights are actuated by the fire detector switches installed in the heater ducting and in the heater compartment. A warning light test switch is installed on the upper left electrical panel immediately adjacent to the warning lights. The warning light test switch is used to check the continuity of the lights and wiring to the fire detector switches. When the test switch is depressed, the warning lights should come ON.

4-56. REMOVAL OF HEATING AND VENTILATING SYSTEM INDICATORS.

a. To remove the heater temperature indicator, remove the screws securing the instrument to the panel, pull the instrument out, and remove the wire leads to the indicator.

b. To remove the heater-inoperative warning light, remove the nut securing the light to the panel, pull the light out, and disconnect the electrical leads to the light.

4-57. INSTALLATION OF HEATING AND VENTILATING SYSTEM INDICATORS.

a. To install the heater temperature indicator, reverse the removal procedure.

b. To install the heater-inoperative warning light, reverse the removal procedure.

4-58. HEATER CONTROL SWITCHES. *(See figure 4-4.)* The heater control switches, installed in the heating and ventilating system for protection of the heater, consist of a heater fuel cycling switch, a heater overheat drop-out switch, and two heater warning switches.

4-59. The Fenwal-type heater fuel cycling switch, installed in the heater discharge duct, prevents the heater from exceeding a normal operating temperature of 130°C (265°F). Whenever this temperature is attained by the heater, the cycling switch will open and break the electrical circuit to the solenoid fuel shutoff valves in the heater fuel control assembly. The balance of the heater electrical circuit is not affected by the operation of the fuel cycling switch.

4-60. The Fenwal-type overheat drop-out switch, installed in the heater discharge duct, prevents overheating of the heater. The drop-out switch is set to close at a temperature of 176°C (350°F). In the event of excessive heat, the drop-out switch will close and blow out the drop-out fuse, which will shut off the fuel and electricity to the heater and the ignition unit and illuminate the heater-inoperative light.

4-61. Two Fenwal switch-type thermal fire detectors, installed in the heater ducting, provide a warning to the flight crew in the event of a heater fire. One detector is installed in the combustion air intake duct of the heater and the other is installed forward of the wye of the heater ventilating air duct. In the event that one or both of the switches are actuated, dual warning lights on the left overhead electrical panel will illuminate. The lights also will illuminate when the fire detector switches in the heater compartment are actuated.

4-62. REMOVAL OF HEATER CONTROL SWITCHES. Each of the heater control switches are removed in the same manner, as follows:

a. Remove the bolts securing the switch plate to the heater ventilating duct.

b. Disconnect the wire leads connected to the switch.

c. Remove the screws connecting the switch to the switch plate.

4-63. INSTALLATION OF HEATER CONTROL SWITCHES. Reverse the removal procedure.

4-64. HEATER FUEL CONTROL ASSEMBLY. *(See figure 4-8.)* The heater fuel control assembly is located under the fuselage floor on the right side of the aircraft, just forward of station 177.5. The units contained in the control assembly are a fuel filter, a fuel pressure regulator, a fuel pump, and two solenoid shutoff valves that are connected hydraulically in series and electrically in parallel. A drain pipe, connected into the bottom of the control assembly container, drains any accumulated fuel overboard. Another pipe connected to the bottom of the control assembly acts as a source of static air for the fuel pressure regulator.

4-65. The heater fuel filter, located within the heater fuel control assembly, is designed to prevent dirt or any foreign matter from entering or clogging any unit of the heater fuel system.

4-66. An air-loaded fuel pressure regulator varies the rate of fuel flow into the heater unit in response to changes in combustion air pressure differential. Control is accomplished by the action of a large diaphragm in the regulator. Combustion air inlet pressure is applied to the top of the diaphragm and a pressure equivalent to heater exhaust pressure is applied to the bottom of the diaphragm. The net loading is the combustion air pressure differential. As the combustion air pressure differential varies in accordance with changing rates of combustion air flow, the diaphragm moves to regulate fuel flow to maintain a correct fuel-air ratio.

4-67. An electrically driven fuel pump is installed in the heater fuel control assembly between the filter and the fuel pressure regulator. The pump will supply a constant source of fuel under pressure to the regulator, which in turn controls the amount of fuel that is permitted to enter the combustion chamber of the heater. The fuel pump is automatically shut off under any one of the following conditions: whenever the master heater control switch is OFF; when the airscoop inlet valve is manually closed; and/or when the heater overheat drop-out switch closes and blows out the heater electrical system fuse.

4-68. Two fuel cycling solenoid-operated shutoff valves, installed in the heater fuel control assembly, are connected in series. The solenoid valves open and close on the demand of the heater temperature control cycling switch. Both solenoids will close if the heater overheat drop-out switch closes and blows out the fuse of the electrical system.

4-69. REMOVAL OF HEATER FUEL CONTROL ASSEMBLY. If one or more of the units contained in the control assembly should require replacement, the complete assembly must be replaced. The control as-

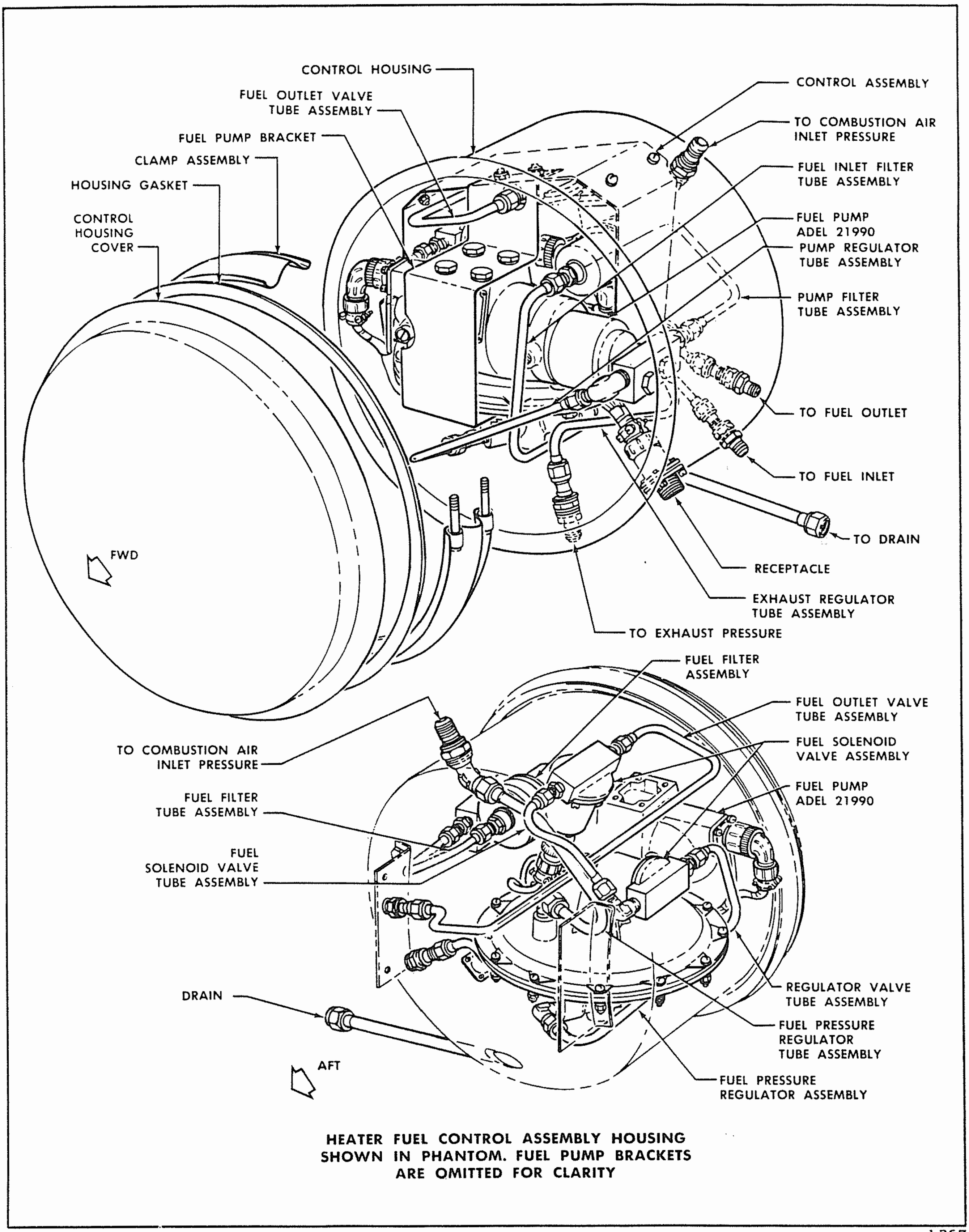


Figure 4-8. Heater Fuel Control Assembly

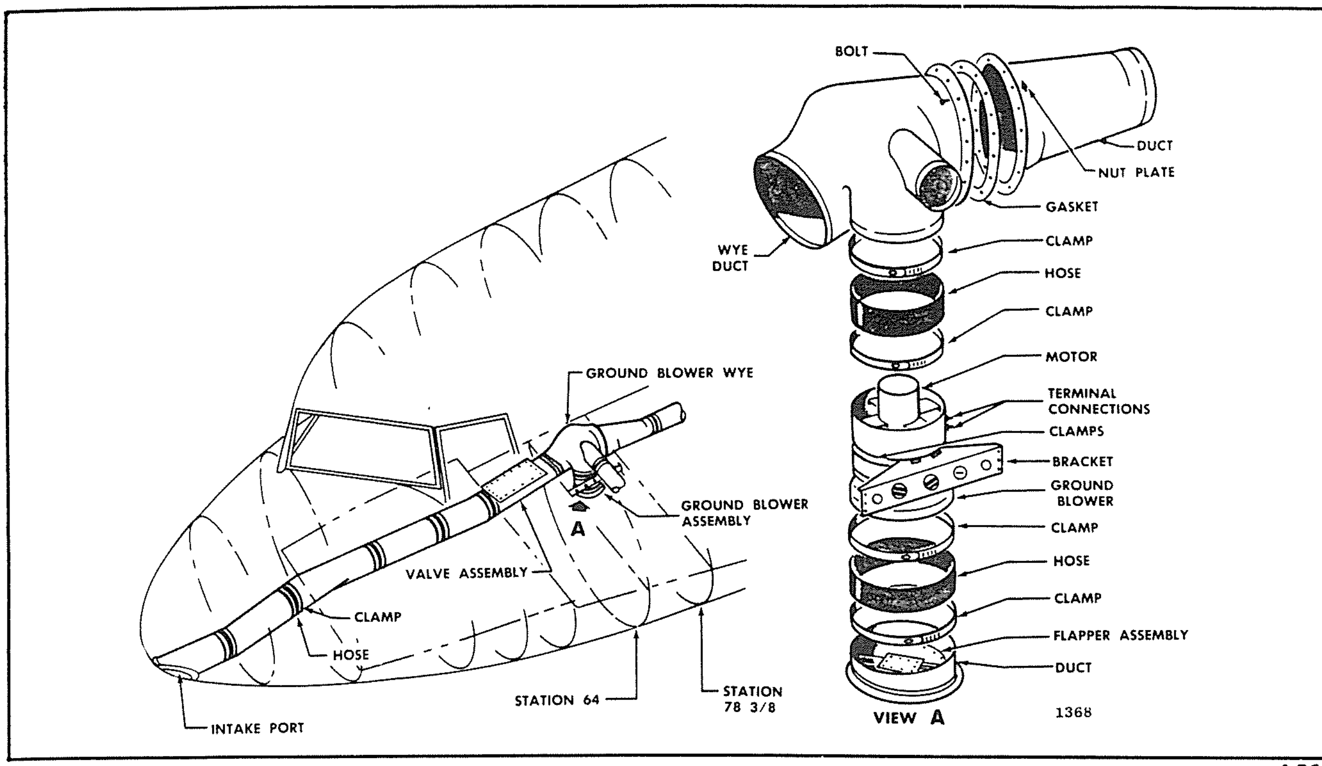


Figure 4-9. Ground Blower Assembly

sembly is accessible through an access door in the bottom of the fuselage.

a. Open the access door in the bottom of the fuselage.

b. Disconnect the fuel inlet and outlet pipes at the control assembly.

c. Disconnect the control assembly overboard drain pipe and static air pipe.

d. Break the electrical connection to the fuel control assembly by disconnecting the plug on the bottom of the control assembly container.

e. Remove the bolts securing the control assembly to the mounting brackets and remove the control assembly.

4-70. INSTALLATION OF HEATER FUEL CONTROL ASSEMBLY. Reverse the removal procedure.

4-71. HEATER FUEL SOLENOID SHUTOFF VALVE. *(See figure 4-8.)* A solenoid-operated fuel shutoff valve is installed in the fuel pipe to the heater between the fuel source and the heater fuel control assembly. The valve, located at station 195 under the fuselage floor, is accessible through the wing access door. The valve will close whenever the heater control switch is OFF, or whenever the drop-out switch closes and blows the drop-out fuse to prevent the flow of fuel from the tank into the heater fuel control assembly.

4-72. REMOVAL OF HEATER FUEL SOLENOID SHUTOFF VALVE.

a. Disconnect the fuel pipe at each end of the valve. The fuel supply pipe should be capped when the valve is removed.

b. Remove the electrical plug from the valve.

c. Loosen the clamp that secures the valve to the aircraft structure.

4-73. INSTALLATION OF HEATER FUEL SOLENOID SHUTOFF VALVE. Reverse the removal procedure.

4-74. GROUND BLOWER ASSEMBLY. *(See figure 4-9.)* The ground blower assembly, installed in the heating and ventilating system, is a 1.2 horsepower, continuous-duty, axial-flow fan capable of supplying the air necessary to heat or ventilate the aircraft while it is on the ground. The ground blower is installed under the floor between stations 64 and 78 on the right side of the aircraft. Air from the ground blower is ducted directly into the main air intake duct of the heating and ventilating system. The ground blower will operate only when the aircraft is on the ground, and then only under either of two conditions: on the ground the blower will be energized when both engines are operating above the generator cut-in speed; or when a ground power unit is connected to the aircraft's electrical system with the BATT-GRD PWR se-

lector switch positioned to GRD PWR. The ground blower is accessible through an access door in the skin of the fuselage.

Note

If a ground air conditioning unit is installed to the ground blower inlet duct connection, the ground unit must not supply air in excess of 1200 cfm. The ground blower in the aircraft also must be operating. This prevents excessive loads on the blower, which may result in the burning out of the ground blower motor.

4-75. REMOVAL OF GROUND BLOWER ASSEMBLY.

a. Unlock the fasteners that secure the intake duct collar of the ground blower installation to the ground blower access door.

b. Open the ground blower access door and remove the clamps that secure the intake duct collar to the ground blower.

c. Remove the clamps that secure the ground blower to the ram air intake duct.

d. Disconnect the electrical leads to the ground blower.

e. Remove the clamps that secure the ground blower to the mounting bracket.

4-76. INSTALLATION OF GROUND BLOWER ASSEMBLY. Reverse the removal procedure.

4-77. HEATER GROUND-FLIGHT TRANSFER RELAY. A ground-flight transfer relay, installed in the heater electrical system, is actuated by an oleo switch on the right strut of the main alighting gear. When the aircraft is on the ground, the ground-flight transfer relay is de-energized to prevent operation of the heater whenever the ground blower is not operating.

4-78. HEATER GROUND-POWER TRANSFER RELAY. A ground-power transfer relay, installed in the heater electrical system, permits operation of the ground blower by connecting an outside source of power into the aircraft electrical system.

4-79. AFT CABIN VENT OUTLET (AIRCRAFT A THROUGH D). *(See figure 4-10.)* The aft cabin vent outlet is installed in the fuselage on station 623. The ventilator is spring-loaded in the closed position and is operated manually by a control located in the coat closet. The purpose of the outlet is to control the normal flow of ventilating air and thus provide a means for the evacuation of smoke from the cabin.

4-80. REMOVAL OF AFT CABIN VENT OUTLET (AIRCRAFT A THROUGH D). To remove the aft cabin vent outlet, proceed as follows:

a. Disconnect the control cable.

b. Disconnect the spring.

c. Remove the slide from the tracks.

4-81. INSTALLATION OF AFT CABIN VENT OUTLET (AIRCRAFT A THROUGH D). Reverse the removal procedure and safety the turnbuckle.

4-82. AFT CABIN VENT OUTLET CONTROLS (AIRCRAFT A THROUGH D). *(See figure 4-10.)* A single-cable, spring-return system extends from the control lever in the coat closet, aft through the fuselage, and connects to the slide at fuselage station 623. Operation of the cable system will move the slide to either the OPEN or CLOSED position.

4-83. AFT CABIN VENT OUTLET CONTROL CABLES (AIRCRAFT A THROUGH D). *(See figure 4-10.)* The aft cabin vent outlet control cables extend from the control handle in the coat closet aft to the vent outlet slide at station 624.

4-84. REMOVAL OF AFT CABIN VENT OUTLET CONTROL CABLES (AIRCRAFT A THROUGH D).

a. Disconnect the vent outlet control cables at the turnbuckle at approximately station 479. Thread the cable ends for removal. Remove the guard pins and rubstrips as necessary.

b. Disconnect the forward control cable at the control lever and draw it forward into the coat closet.

c. Disconnect the aft control cable from the vent slide at station 624 and pull it aft into the aft cargo compartment.

4-85. INSTALLATION OF AFT CABIN VENT OUTLET CONTROL CABLES (AIRCRAFT A THROUGH D).

a. Connect the aft vent outlet control cable to the vent slide at station 624 and route it forward through the aircraft, installing guard pins and rubstrips as necessary.

b. Connect the forward vent outlet control cable to the control lever in the coat closet and route it aft through the fuselage, installing necessary guard pins and rubstrips.

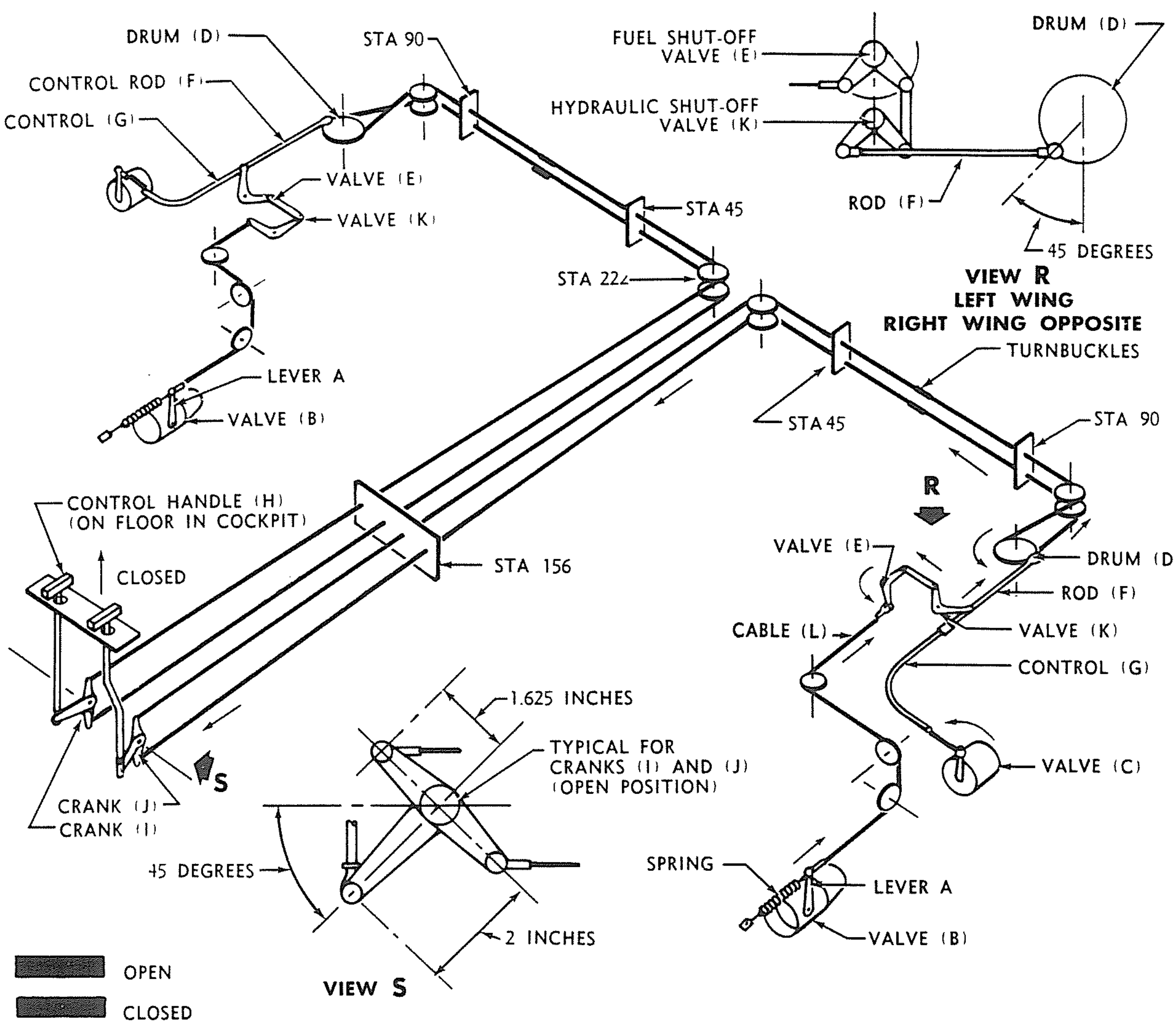
c. Connect the cables with a turnbuckle at approximately station 479.

d. Adjust the aft cabin vent outlet controls (see paragraph 4-86).

e. Safety the turnbuckle.

4-86. ADJUSTMENT OF AFT CABIN VENT OUTLET CONTROLS (AIRCRAFT A THROUGH D). Adjust the aft cabin vent outlet control cable so that the ventilator is fully OPEN when the control handle is in the OPEN position. Safety the turnbuckle.

4-87. HEATING AND VENTILATING SYSTEM BOLT TORQUE VALUES. Unless otherwise noted, see paragraphs 4-259 and 4-260 for bolt torque values.



RIGGING INSTRUCTIONS

ADJUSTMENT PROCEDURE

1. Check lever (A) to see that spring holds generator blast tube valve (B) in OPEN position, full forward against integral stop.
2. Set oil valve (C) in OPEN position (rotated outboard to integral stop).
3. Set drum (D) and fuel valve (E) in position shown in View R and adjust control rod (F) as necessary for installation.
4. Adjust control (G) as necessary for installation and check to assure that when linkage shown in View R is in CLOSED position, drum (D) rotated counterclockwise until valves (E) and (K) reach integral stop, oil valve (C) is in CLOSED position (rotated inboard to integral stop).
5. Rotate linkage back to OPEN position and adjust cable (L) to five pounds tension. Check to assure that spring holds fuel valve (E) open. Check to assure that when linkage is moved to CLOSED position, generator blast tube valve (B) also closes.
6. Reset linkage to OPEN position, check to see that cockpit control handles (H) are 1/16 inch above support panel when cranks (I) and (J) are in position shown in View S, and adjust turnbuckles between wing stations 45 and 90 to tension cables in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
7. Check the system to assure that generator blast tube valves (B), oil valves (C), fuel valves (E) and hydraulic valves (K) all close when cockpit control handle (H) is pulled up to full limit.
8. Safety the turnbuckles.

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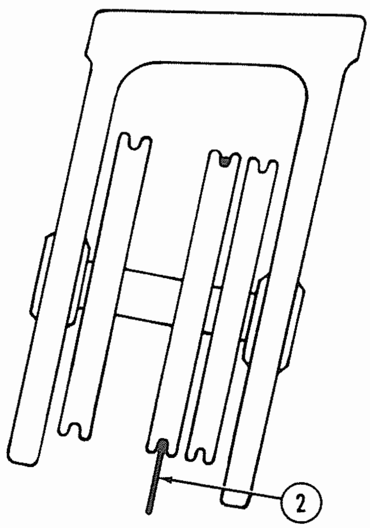
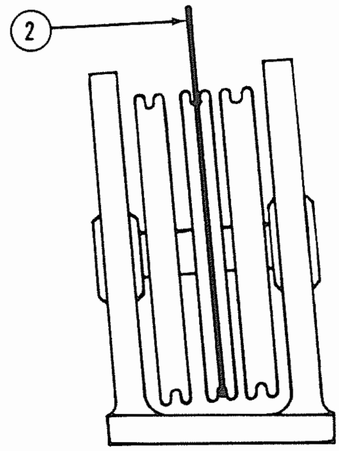
Figure 4-10 (Sheet 1 of 2 Sheets). Aft Cabin Vent Outlet Control System (Aircraft A through D) — Key Drawing, Control Lever



VIEW C—PULLEY BRACKET AT STATION 646
(LOOKING AFT)



VIEW D—PULLEY BRACKET AT STATION 623
(LOOKING FORWARD)

AFT CABIN VENT OUTLET CONTROL CABLES

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE LENGTH (L_2)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)	FITTINGS (5)
1	5403089-7	1	A	50		1/16 dia 7x7 flex	AN667-2	AN669-S2 RH			
2	5403089-5	1	B	170 1/2		1/16 dia 7x7 flex	AN669-S2 LH	AN667-2			

CABLE ASSEMBLIES

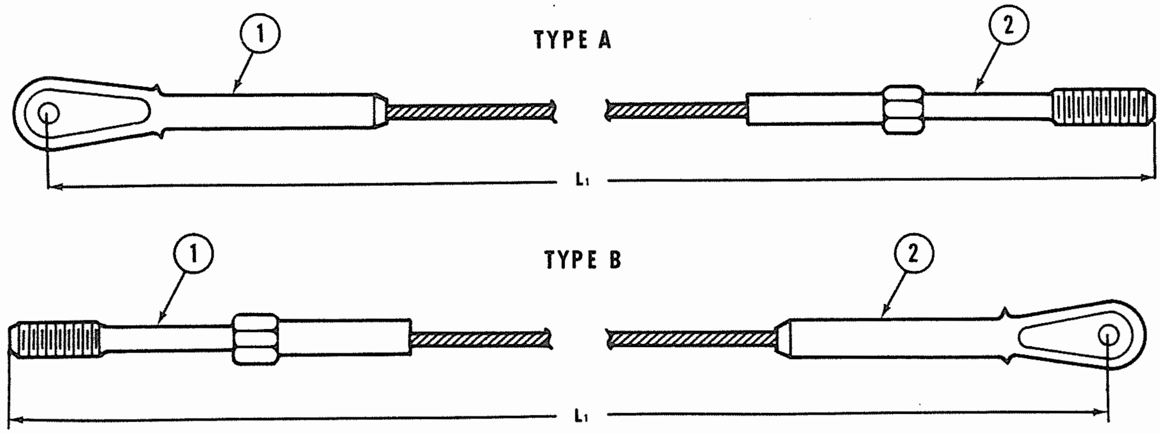


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Figure 4-10 (Sheet 2 of 2 Sheets). Aft Cabin Vent Outlet Control System (Aircraft A through D) — Pulley Brackets, Stations 624 and 646; Cable Chart and Cable Assemblies

4-88. ACCESSORY ANTI-ICING SYSTEMS.

4-89. DESCRIPTION. *(See figures 4-11, 4-12, and 4-13.)* The anti-icing systems consist of an alcohol, an electrical, and a heated-air-type system. The alcohol anti-icing system is installed to prevent the formation of ice on the propeller blades, in the carburetor throat, and on the windshield. The electrical anti-icing system prevents and/or eliminates ice in the pitot tubes and, on aircraft A, B, and 1 through 62, in the nose air-scoop area by the incorporation of electrical resistance-type heating elements in these components. For additional information concerning the electrical anti-icing system, see paragraph 7-129. The heated-air anti-icing system provides hot air to the windshield both for vinyl warming and to prevent the formation of ice on the outside of the windshield. In the propeller anti-icing system, the anti-icing fluid is pumped from a supply tank to the slinger rings installed on each propeller hub. Distribution pipes from the rings direct the alcohol to each blade. The rate of fluid flow may be regulated from the flight compartment. In the carburetor anti-icing system, the anti-icing fluid is pumped from the same alcohol supply tank to the spray nozzles in the carburetor throat. The fluid flow, but not the fluid flow rate, may be controlled from the flight compartment. Windshield anti-icing, accomplished through the use of alcohol spray and heated air from the combustion heater, is regulated by manual controls located in the flight compartment. For additional information concerning the heated air manual control, see paragraph 4-40.

4-90. TROUBLE SHOOTING OF ACCESSORY ANTI-ICING SYSTEM COMPONENTS.

Trouble	*Probable Cause*	*Remedy*
a. Alcohol pump motor inoperative.	Circuit breaker tripped.	Reset circuit breaker.
	Low input voltage.	Check power supply.
	Wiring not properly connected.	Check against wiring diagram.
	Loose or high-resistance connections.	Tighten or clean connections.
	Brushes binding in brush boxes.	Free brushes.
	Worn brushes.	Replace pump.
	Dirty commutator.	Remove dirt and grease from commutator with solvent-moistened cloth; smooth commutator with 000 sandpaper.
	Shorted or grounded armature.	Replace motor.
	Grounded or shorted field coils.	Replace motor.
b. No fluid flow at carburetor.	Tank empty (check drain valve).	Refill tank.
b. *(Continued)*	Pump motor inoperative.	See step a, preceding.
	Leaking connections; broken or clogged pipes.	Repair connections.
	Clogged filter.	Clean filter.
	Clogged spray nozzles.	Clean nozzles.
c. Low or fluctuating fluid output.	Fluid level low.	Fill supply tank.
	Pump motor not functioning at correct or constant rpm.	Check motor ground connection, input voltage, armature, fields, brushes, and for free action of motor shaft.
	Drive shaft on pump sheared.	Replace pump.

PITOT HEAD ANTI-ICER SYSTEM

Note

In order to check the heating element, the oleo switch must be in flight position.

d. Pitot head heaters inoperative.	Circuit breakers tripped.	Reset circuit breakers.
	Loose connections at terminals.	Check wiring.
	Defective switch.	If switch is defective, replace it.
	Defective element.	Check continuity of heating element. The element in each pitot head should indicate three to five amperes. Replace defective pitot heads.
	Faulty ammeter selector switch.	Replace switch.
	Defective isolation fuses.	Check fuse holder and replace fuse, if defective.

AIRSCOOP ANTI-ICER SYSTEM (AIRCRAFT A, B, AND 1 THROUGH 62)

WARNING

Do not keep the airscoop anti-icing circuit closed for more than one minute at a time.

e. Airscoop heater inoperative.	Circuit breakers tripped.	Reset circuit breakers.
	Loose connections at terminals.	Check wiring.
	Defective switch.	Replace switch.

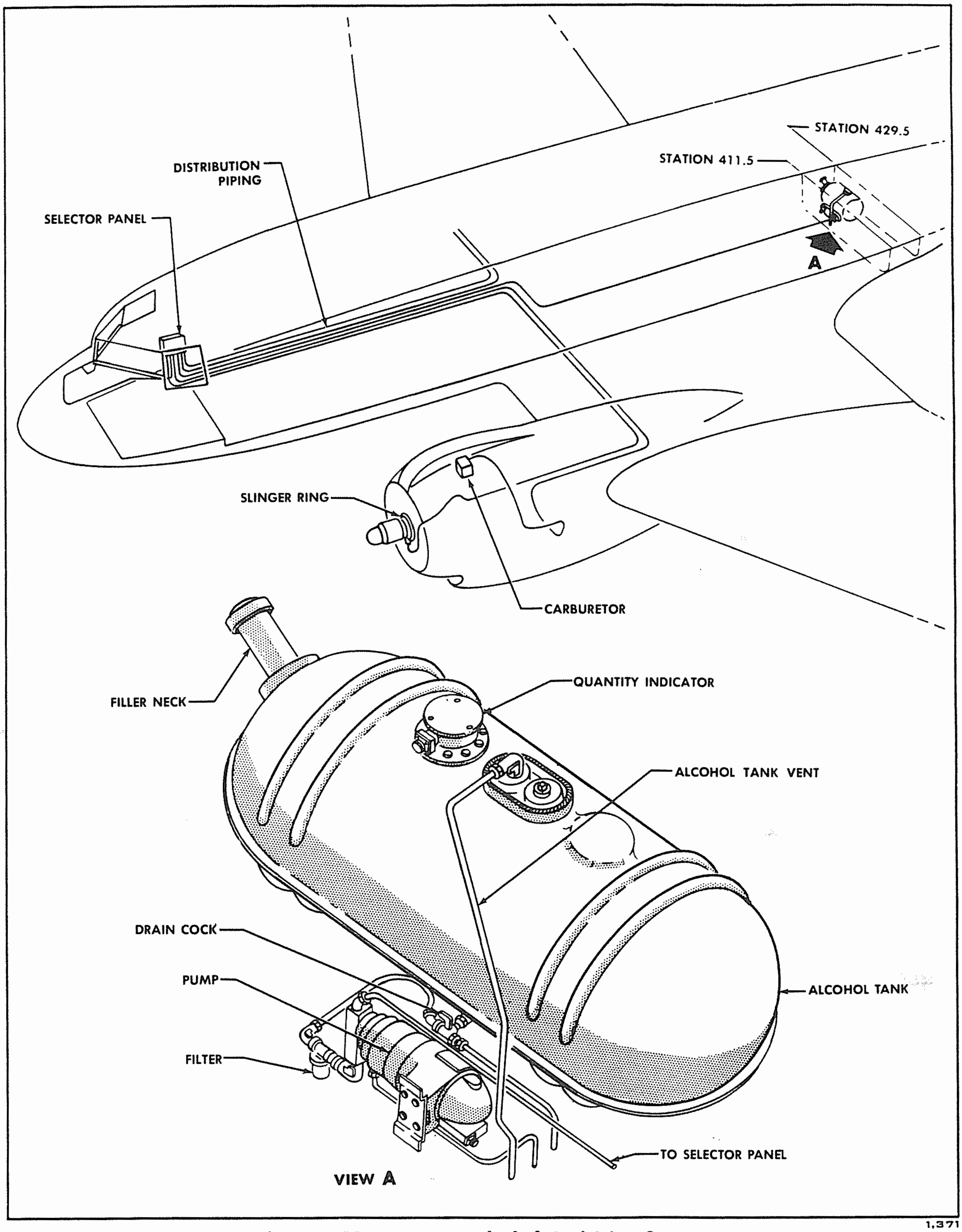

Figure 4-11. Accessory Alcohol Anti-Icing System

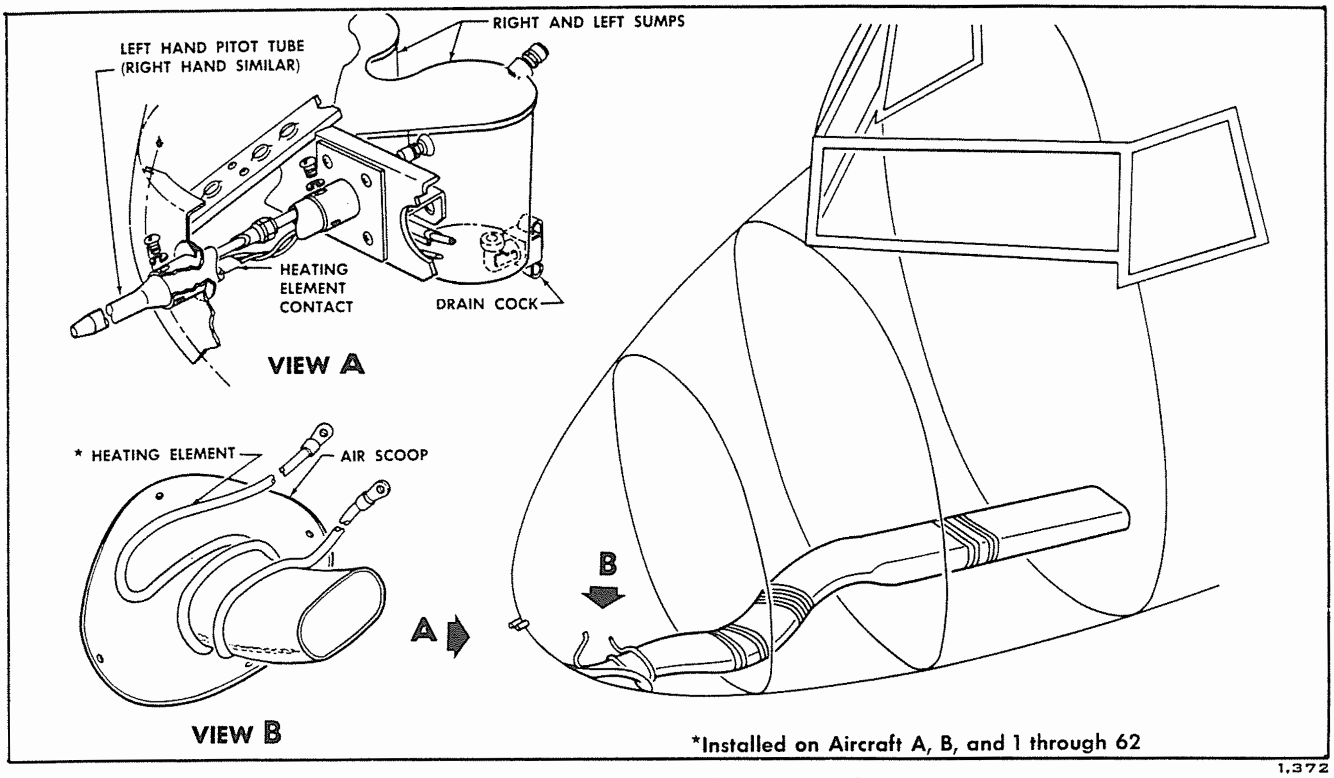

Figure 4-12. Accessory Electrical Anti-Icing System

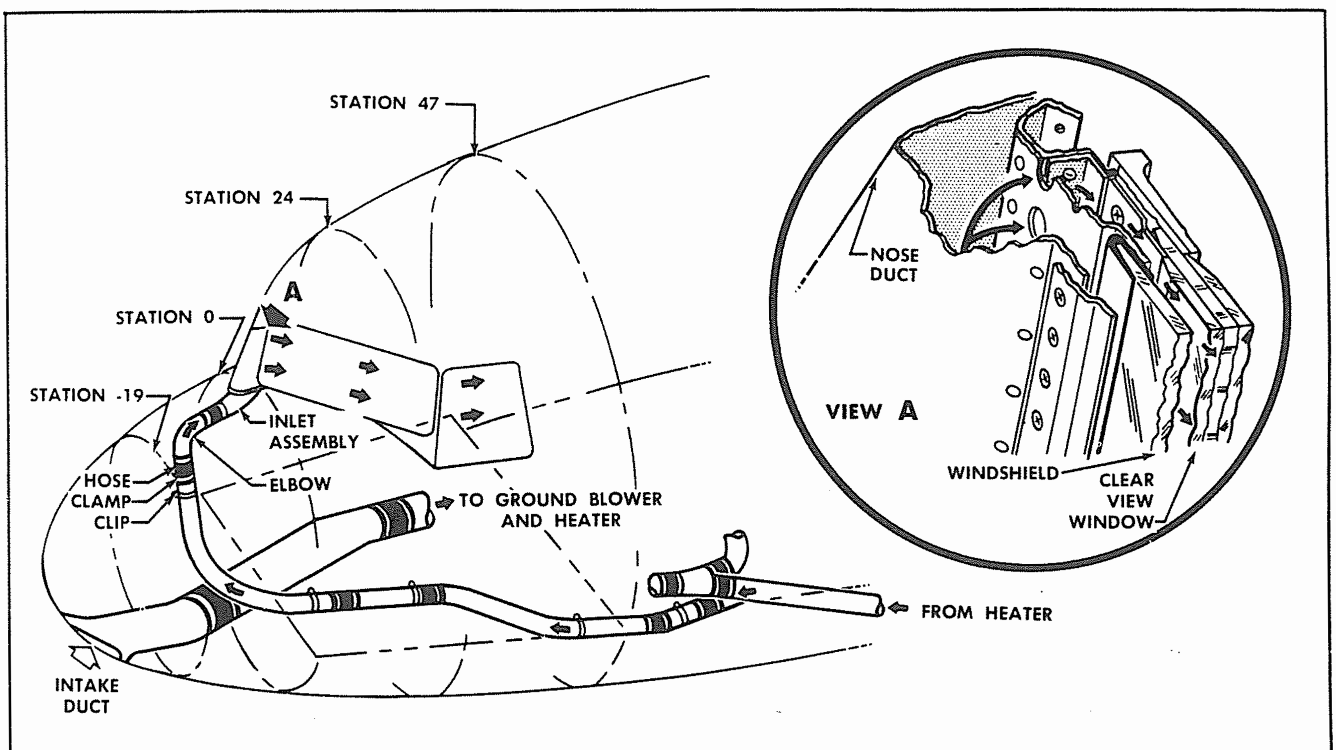

Figure 4-13. Accessory Heated Air Anti-Icing System

Trouble	*Probable Cause*	*Remedy*
e. *(Continued)*	Defective element.	Check continuity of heating element. The element in the scoop should indicate approximately 50 amperes. Replace scoop if defective.
	Faulty ammeter selector switch.	Replace switch.
	Defective isolation fuses.	Check fuse holder and replace fuse if defective.

4-91. TEST OF ALCOHOL ANTI-ICING DISTRIBUTION SYSTEM. To test the alcohol anti-icing distribution system, turn on the alcohol pump and check the slinger rings, the windshield, and the carburetor spray nozzles to make certain that fluid is being supplied to these areas.

Note

Do not run the alcohol system with the pump dry (without pumping alcohol) longer than is necessary to check its operation.

4-92. TEST OF ALCOHOL QUANTITY INDICATOR SYSTEM.

a. Replace the regulator seal on the Liquidometer unit on the alcohol tank with a dummy seal or spacer so that the tank float may be moved up and down.

b. Attach a string to the float arm so that it may be passed between the transmitter flange and the tank.

c. With power OFF, move the float from the lowest position to the highest position. The indicator needle must move from directly over the empty position to directly over the full position. If this condition is not met, adjust the transmitter at the empty and full positions.

d. To adjust the transmitter indicating system at the FULL position, remove the top cover of the transmitter unit and turn the adjustment screw on the contact shoe.

e. To adjust the transmitter at the EMPTY position, turn the adjustment screw on the opposite shoe.

4-93. TEST OF ELECTRICAL ANTI-ICING SYSTEMS.

a. On aircraft A, B, and 1 through 62, to test the airscoop heater, the aircraft must be on jacks or the oleo switch must be manually changed to the position it assumes when the aircraft is off the ground.

b. Turn ON the pitot and airscoop heaters.

c. Connect the ammeter to each of the circuits by means of the pitot and, on aircraft A, B, and 1 through 62, the airscoop heater selector switch in the cockpit. The current reading should be between 2.4 and 4.2 amperes for the pitot heaters at normal room temperature, and between 47 and 51 amperes for the scoop heater at a surface temperature of 38°C (100°F) and at an applied voltage of 27 volts.

CAUTION

After this test is completed, the oleo switch must be returned to its normal position.

4-94. TEST OF WINDSHIELD ANTI-ICING SYSTEM. To test the windshield heat supply, see paragraph 4-5.

4-95. ALCOHOL ANTI-ICING FLUID SUPPLY TANK. *(See figure 4-14.)* The alcohol tank, mounted under the fuselage floor between stations 411 and 429, supplies anti-icing fluid to the propellers, the windshield, and the carburetors. The capacity of the tank is 20 US. (16.65 Imp.) gallons. The tank is held in position with two strap assemblies, each with a clevis and pin adjustment in the center. The filler neck cap is located just below an access door located on the lower right side of the fuselage directly opposite the main entrance door. Both propeller and carburetor anti-icing systems have been designed for the use of AN-F-13 isopropyl alcohol.

4-96. REMOVAL OF ALCOHOL ANTI-ICING FLUID SUPPLY TANK

a. Remove the toilet (aircraft A and B only).

b. Remove the right fuselage floor panel between stations 411 and 429.

c. Remove the clamp attaching the filler neck boot to the tank drain pan.

d. Disconnect the fluid pipes.

e. Disconnect the electrical connection.

f. Loosen the tank hold-down clamps.

g. Remove the supply tank.

4-97. INSTALLATION OF ALCOHOL ANTI-ICING FLUID SUPPLY TANK. Reverse the removal procedure.

4-98. ALCOHOL ANTI-ICING SYSTEM FLUID QUANTITY INDICATING SYSTEM AND TRANSMITTER UNIT. *(See figure 4-14.)* The anti-icing alcohol quantity indicator system consists of a float-type transmitter mounted on the alcohol tank and connected electrically to an indicator on the main instrument panel. Bus voltage is applied to the system between one indicator terminal and ground connections on both units. Movement of the float, caused by a change in alcohol level, moves a contact arm across a resistance strip in the transmitter. This varies the current distribution in the indicator, causing the needle to deflect according to the change in the alcohol level.

4-99. REMOVAL OF ALCOHOL ANTI-ICING SYSTEM FLUID QUANTITY TRANSMITTER UNIT.

a. Disconnect the electrical connection.

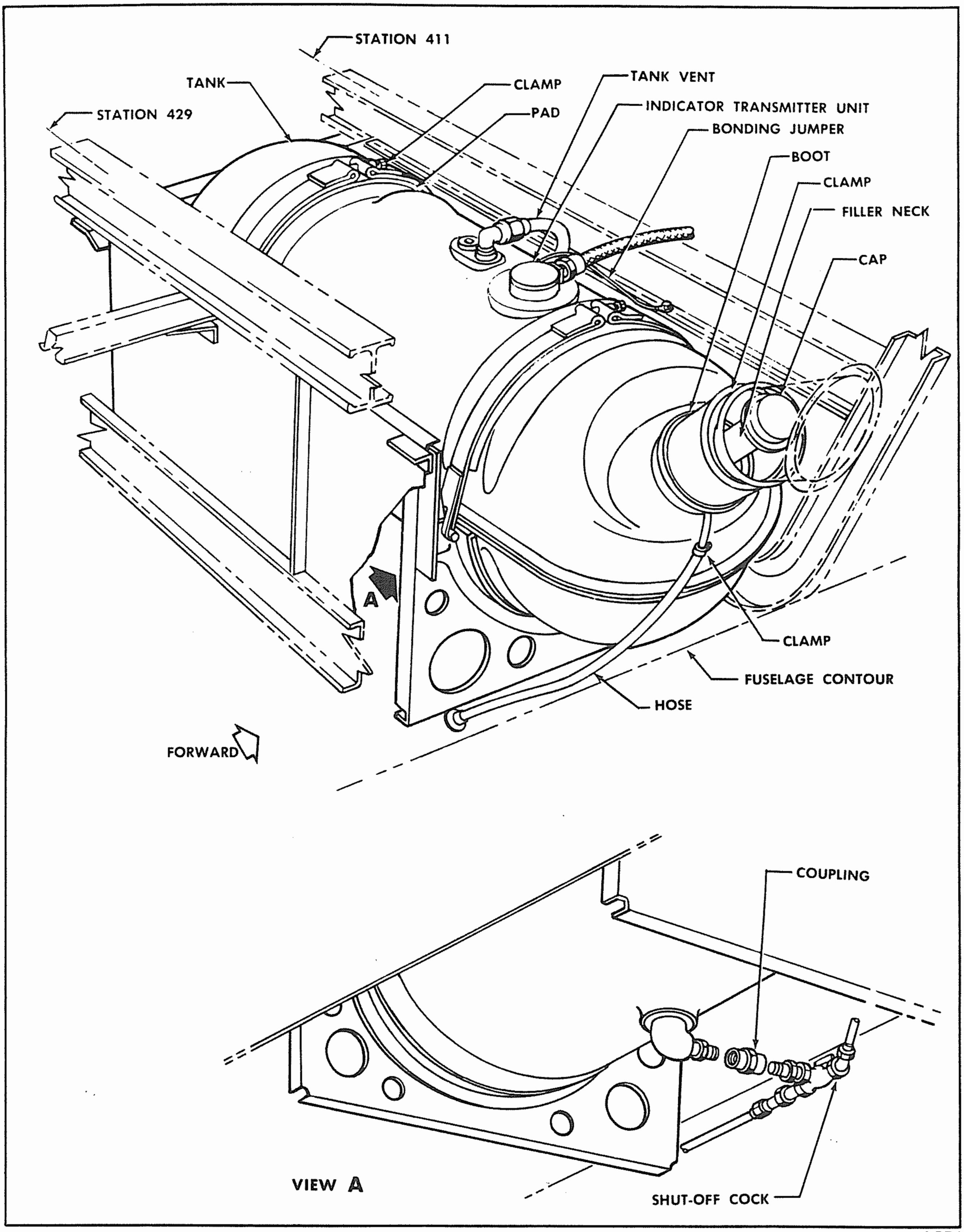

Figure 4-14. Alcohol Tank

b. Remove the eight screws securing the tank transmitter unit housing to the supply tank.

c. Carefully remove the bonding jumper.

d. Remove the tank transmitter unit assembly from the tank.

4-100. INSTALLATION OF ALCOHOL ANTI-ICING SYSTEM FLUID QUANTITY TRANSMITTER UNIT. Reverse the removal procedure, making certain that the bonding jumper is in position.

4-101. TEST AFTER INSTALLATION OF ALCOHOL ANTI-ICING SYSTEM FLUID QUANTITY TRANSMITTER UNIT. For test and adjustment procedure, see paragraph 4-92.

4-102. ALCOHOL ANTI-ICING ELECTRIC FUEL PUMP. *(See figure 4-14.)* The alcohol supply pump is mounted adjacent to the supply tank below the fuselage floor and in the area between stations 411 and 429 on the right side of the aircraft. It has a rated output of 55 gallons per hour and operates on the standard aircraft electrical 28-volt system. The inlet port of the pump is connected to the supply tank and the outlet port is connected to the selector panel in the flight compartment. Operation of the pump is controlled through a switch located in the flight compartment.

4-103. REMOVAL OF ALCOHOL ANTI-ICING ELECTRIC FUEL PUMP.

a. Disconnect the electrical connection.

b. Disconnect both the fluid inlet and the fluid outlet pipes.

c. Remove the four attaching bolts that secure the pump mounting bracket to the fuselage structure.

d. Remove the electric pump and mounting bracket assembly.

4-104. INSTALLATION OF ALCOHOL ANTI-ICING ELECTRIC FUEL PUMP. Reverse the removal procedure.

4-105. TEST OF ALCOHOL ANTI-ICING ELECTRIC FUEL PUMP. For test procedure, see paragraph 4-91, preceding.

Note

Do not run the unit with the pump dry (without pumping alcohol) longer than is necessary to check its operation.

4-106. ALCOHOL ANTI-ICING SYSTEM FILTER. *(See figure 4-14.)* The filter is located in the pipe between the tank and the inlet port of the pump. The filter consists of three major parts: a metal cap incorporating the inlet and outlet ports, a bowl to collect sediment, and a filter element. The fluid is drawn through the inlet port, into a passageway in the cap and into the bowl that surrounds the filter element. It is then drawn through the filter element and outlet port by the suction of the pump. The filter element permits passage of the required amount of fluid, but maintains a sufficient restriction to prevent small particles from entering the pump.

4-107. REMOVAL OF ALCOHOL ANTI-ICING SYSTEM FILTER.

a. Turn the shutoff and drain cock valve to the OFF position.

b. Disconnect the fuel pipes at the inlet and outlet ports.

c. Remove the filter unit.

4-108. INSTALLATION OF ALCOHOL ANTI-ICING SYSTEM FILTER. Reverse the removal procedure.

4-109. ALCOHOL ANTI-ICING SYSTEM SHUTOFF AND DRAIN COCK. *(See figure 4-14.)* The alcohol system shutoff and drain cock is located in the tank feed pipe and immediately adjacent to the tank. It is a three-way valve and controls the alcohol supply to the pump, overboard drain, and system shutoff. Placing the valve in the OFF position permits removal of the filter and/or pumping units without the necessity of draining the tank.

4-110. REMOVAL OF ALCOHOL ANTI-ICING SYSTEM SHUTOFF AND DRAIN COCK.

a. Drain the tank.

b. Disconnect the fuel pipes.

c. Remove the drain cock.

4-111. INSTALLATION OF ALCOHOL ANTI-ICING SYSTEM SHUTOFF AND DRAIN COCK. Reverse the removal procedure.

4-112. PROPELLER SLINGER RING AND DISTRIBUTION PIPES. *(See figure 4-15.)* Ice is removed from the propellers by isopropyl alcohol flowing onto each blade from a slinger ring mounted on the propeller hub. The alcohol is directed from the supply pump by pipes to a nozzle, which pour anti-icing fluid into the whirling slinger ring. Centrifugal force throws the alcohol around the inside of the slinger ring and into the blade tubes. These blade tubes direct alcohol onto each blade of the propeller.

CAUTION

Only a propeller specialist must attempt any phase of maintenance or removal of the slinger rings, because they are balanced to the particular assembly on which they were originally installed.

4-113. CARBURETOR THROAT ANTI-ICING SPRAY NOZZLES. *(See figure 4-15.)* The carburetor anti-icing system consists primarily of the necessary pipes, connections, and spray nozzles by which filtered

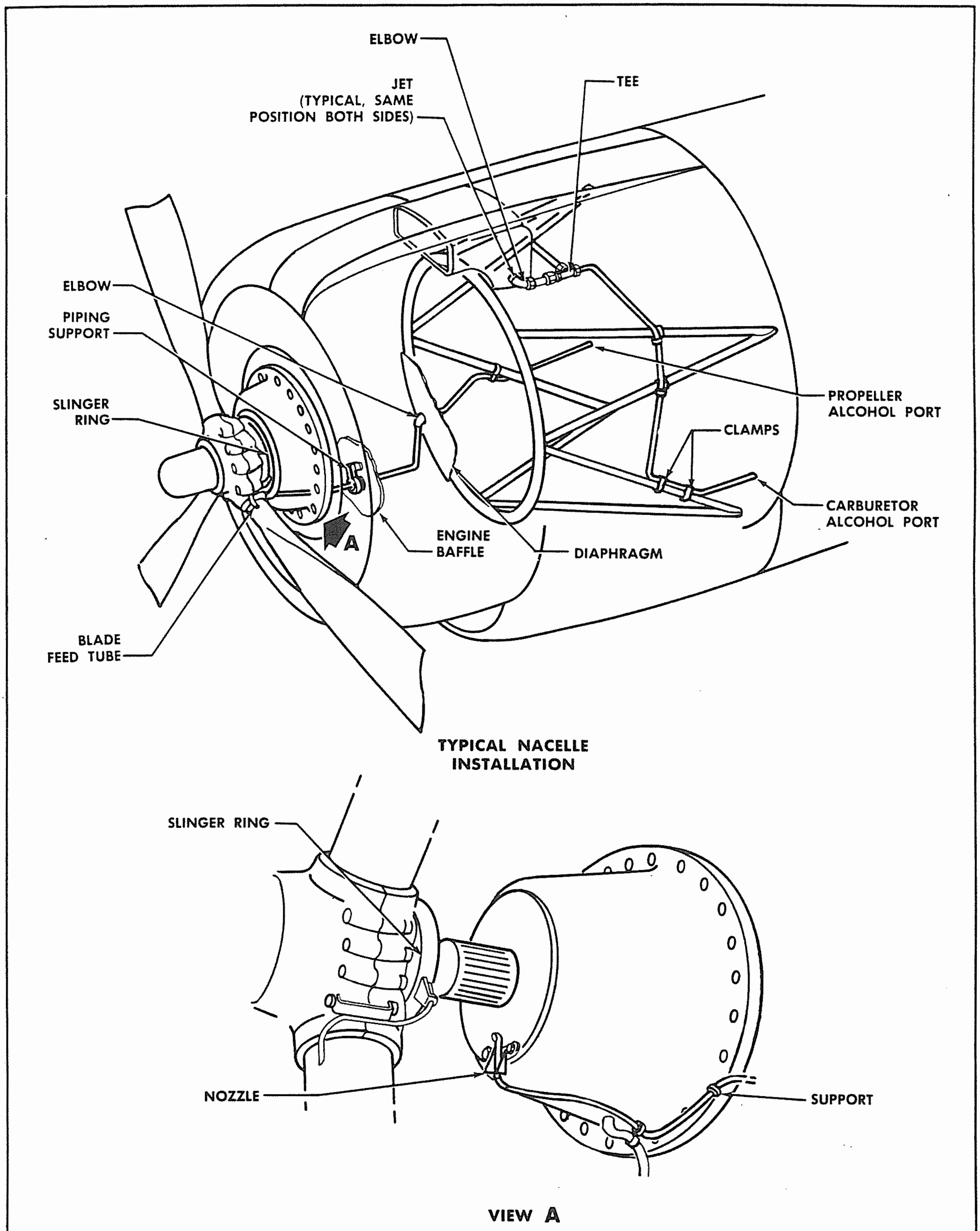

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Figure 4-15. Alcohol Anti-Icing Propeller Slinger Ring, Carburetor Spray Nozzles, and Distribution Piping

alcohol is sprayed into the adapter of the carburetors to remove any accumulation of ice. The jets of alcohol overlap and form a fog of alcohol that covers the screen installed at the bottom of the carburetor adapter.

4-114. REMOVAL OF CARBURETOR THROAT ANTI-ICING SPRAY NOZZLES.

a. Disconnect the supply pipe at the spray nozzle.

b. Remove the bolts securing the spray nozzle to the carburetor throat and remove the nozzle.

4-115. INSTALLATION OF CARBURETOR THROAT ANTI-ICING SPRAY NOZZLES. Reverse the removal procedure.

4-116. ALCOHOL ANTI-ICING SYSTEM SELECTOR PANEL. *(See figure 4-16.)* The alcohol anti-icing system selector panel is located on the right side of the flight compartment at approximately station 37. The three forward valves on the selector panel are used to regulate the flow of alcohol to the propeller slinger rings and the windshield. Since the valves are of the needle type, they may be adjusted to any desired rate of flow required to meet existing conditions. The two aft valves on the selector panel are used to control the flow of alcohol to the carburetor anti-icing spray nozzles, mounted in the airscoop adapters. The carburetor control valves supply alcohol to the carburetors at a predetermined rate only and may not be adjusted. In order to complete the operation of the alcohol anti-icing system, the alcohol pump two-position toggle switch, mounted on the face of the selector panel assembly, must be moved to the ON position.

4-117. REMOVAL OF ALCOHOL ANTI-ICING SYSTEM SELECTOR PANEL.

a. Remove the inlet and outlet pipes.

b. Remove the propeller, windshield, and carburetor valves from the panel.

c. Disconnect the electrical connections.

d. Remove the screws securing the panel to the support and remove the panel.

4-118. INSTALLATION OF ALCOHOL ANTI-ICING SYSTEM SELECTOR PANEL. Reverse the removal procedure.

4-119. PITOT TUBE ANTI-ICING. *(See figure 4-12.)* The pitot heads are kept free of ice by electrical heating elements incorporated in the pitot head. For removal and installation of the pitot tube and de-icing elements, see paragraphs 6-33 and 6-34.

4-120. TEST AFTER INSTALLATION OF PITOT TUBE ANTI-ICING. With engines operating at a minimum of 900 rpm and with the oleo switch manually changed to flight position, turn on the pitot and air conditioning airscoop anti-icing switch, and check each unit according to the placard in the aircraft. If any of the units are inoperative, it is necessary to replace the entire unit.

4-121. AIRSCOOP ANTI-ICING (AIRCRAFT A, B, AND 1 THROUGH 62). *(See figure 4-12.)* A resistance-type heating element is installed in the nose airscoop to prevent the accumulation of ice in this area. Current flow to the heater is indicated on an ammeter mounted on the right overhead electrical panel.

4-122. WINDSHIELD ANTI-ICING. *(See figure 4-13.)* Windshield anti-icing is accomplished through the use of hot air taken from the combustion heater installation and routed between the double windshield panes. Windshield anti-icing is regulated by manual controls located in the flight compartment. The hot air control is connected mechanically to the air valve assembly in the heating and ventilating duct system. For information concerning the windshield volume control valve assembly, see paragraphs 4-37 through 4-39. The windshield alcohol control is installed on the alcohol selector panel (see paragraph 4-116). For removal of the windshield panes, see paragraph 2-290.

4-123. ACCESSORY ANTI-ICING SYSTEM BOLT TORQUE VALUES. Unless otherwise noted, see paragraphs 4-259 and 4-260 for bolt torque values.

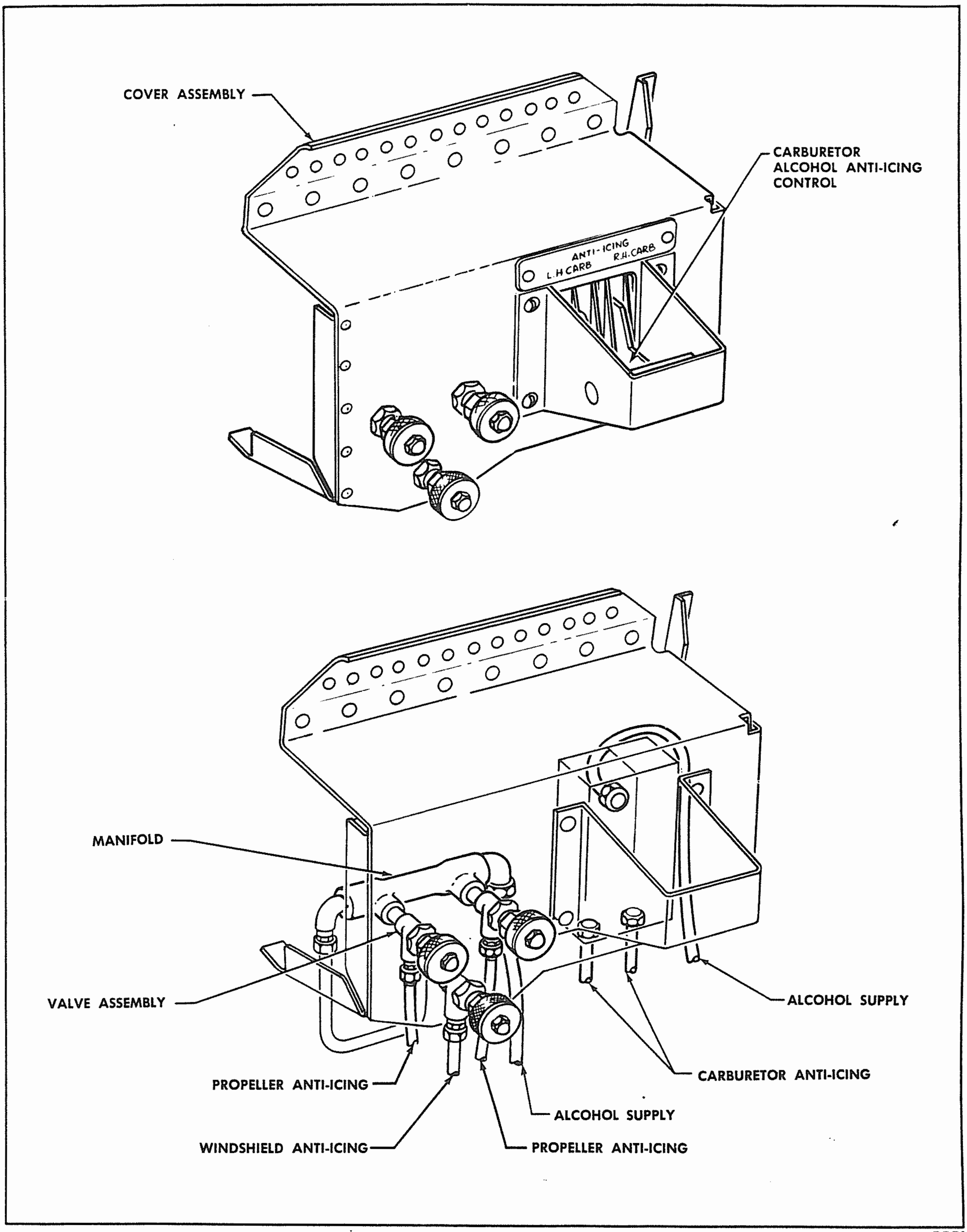


Figure 4-16. Alcohol Anti-Icing Selector Panel

4-124. SURFACE DE-ICING SYSTEM.

4-125. DESCRIPTION. *(See figure 4-17.)* Air-inflated rubber de-icers (sometimes called shoes or boots) remove ice that has formed on the leading edges of the wing and tail surfaces. Span-wise tubular cells, incorporated within the wing de-icers, are inflated alternately, causing a wave-like motion that cracks any ice that has formed. Once the ice has been disintegrated, it is carried away by the force of the air stream. Pressurized air for the operation of the surface de-icing system is supplied by two engine-driven de-icing pumps. Air from each pump passes through an oil separator mounted in each engine section and then passes through a common pressure regulating oil separator in the wing center section. From there it passes on to a distributor valve which directs the flow of air to the de-icers. The engine section oil separators and the pressure regulating oil separator remove the small amounts of engine oil that the air picks up at the pumps. Check valves in the individual pressure pipes from each pump prevent loss of air pressure in the event of pump failure or pipe damage in either of the engine sections. The check valves are located in the nacelle sections aft of the firewall. The de-icers are placed in operation by moving the surface de-icing control, located on the forward side of the bulkhead immediately behind the co-pilot's station, to the ON position. The control is connected by cables to the distributor valve crank, which in turn actuates the electric distributor motor at the same time that the air from the engine-driven pumps is directed to the distributor. When the de-icing switch is in the OFF position, the air from the engine-driven pumps is exhausted overboard and the distributor motor is turned off. The sequence *(see figure 4-18)* in which the de-icers are inflated and deflated is as follows: center tubular cells of the right and left outboard wing de-icers; upper and lower tubular cells of the right and left outboard wing de-icers; center tubular cells of the right and left inboard wing de-icers; upper and lower tubular cells of the right and left inboard wing de-icers; horizontal and vertical stabilizer de-icers.

4-126. TROUBLE SHOOTING OF SURFACE DE-ICING SYSTEM.

Trouble	*Probable Cause*	*Remedy*
a. Pressure gage oscillation peaks irregular.	Pipes or connections between distributor valve and de-icers faulty.	Repair or replace faulty pipes. Tighten loose connections. Replace faulty compression fitting seals.
	De-icers torn, bruised, or punctured.	Repair or replace faulty de-icers.
	Faulty action of relief valve in regulating oil separator.	Check valve for proper operation. Repair or replace as necessary.
b. With pressure gage at normal reading, de-icers do not inflate.	Faulty de-icing switch.	Replace de-icing switch.
	Faulty wiring connections.	Check, clean and repair connections.
	Distributor valve motor inoperative.	Replace distributor valve.
	Faulty distributor valve reduction gearing or valve rotor coupling.	Replace distributor valve.
c. Cycling period too long.	Dirty distributor valve motor commutator.	Clean commutator.
	Worn distributor valve motor brushes.	Replace spring and brush assemblies.
	Faulty external wiring to distributor valve causing intermittent operation of motor.	Repair or replace faulty wiring.
	Thrust adjusting screws improperly adjusted.	Adjust screws.
	Valve rod or seals improperly adjusted.	Replace faulty seals; run in new seals.
d. Pressure gage oscillation—peak pressure is low.	Leaky tubing or connections between pump and distributor valve.	Repair or replace faulty lines. Tighten loose connections. Replace faulty compression seals.
	Incorrectly installed regulating oil separator seal.	Replace seal and install correctly.
	Faulty pressure regulating valve in regulating oil separator.	Replace regulating oil separator.
	Faulty gage or pipe.	Replace gage or pipe.
	Vacuum relief valve screen clogged.	Clean or replace screen.
	Pumps not delivering sufficient air.	Replace pumps.
e. Pressure gage oscillation—peak pressure is high.	Pressure regulating valve in regulating oil separator inoperative in closed position.	Repair or replace faulty regulating oil separator.
	Faulty pressure gage.	Replace faulty gage.
f. Pressure gage shows no pressure while shoes are operating.	Faulty gage or pipe.	Replace faulty gage or pipe.
	Faulty electrical circuit.	Check out circuit; replace faulty parts.

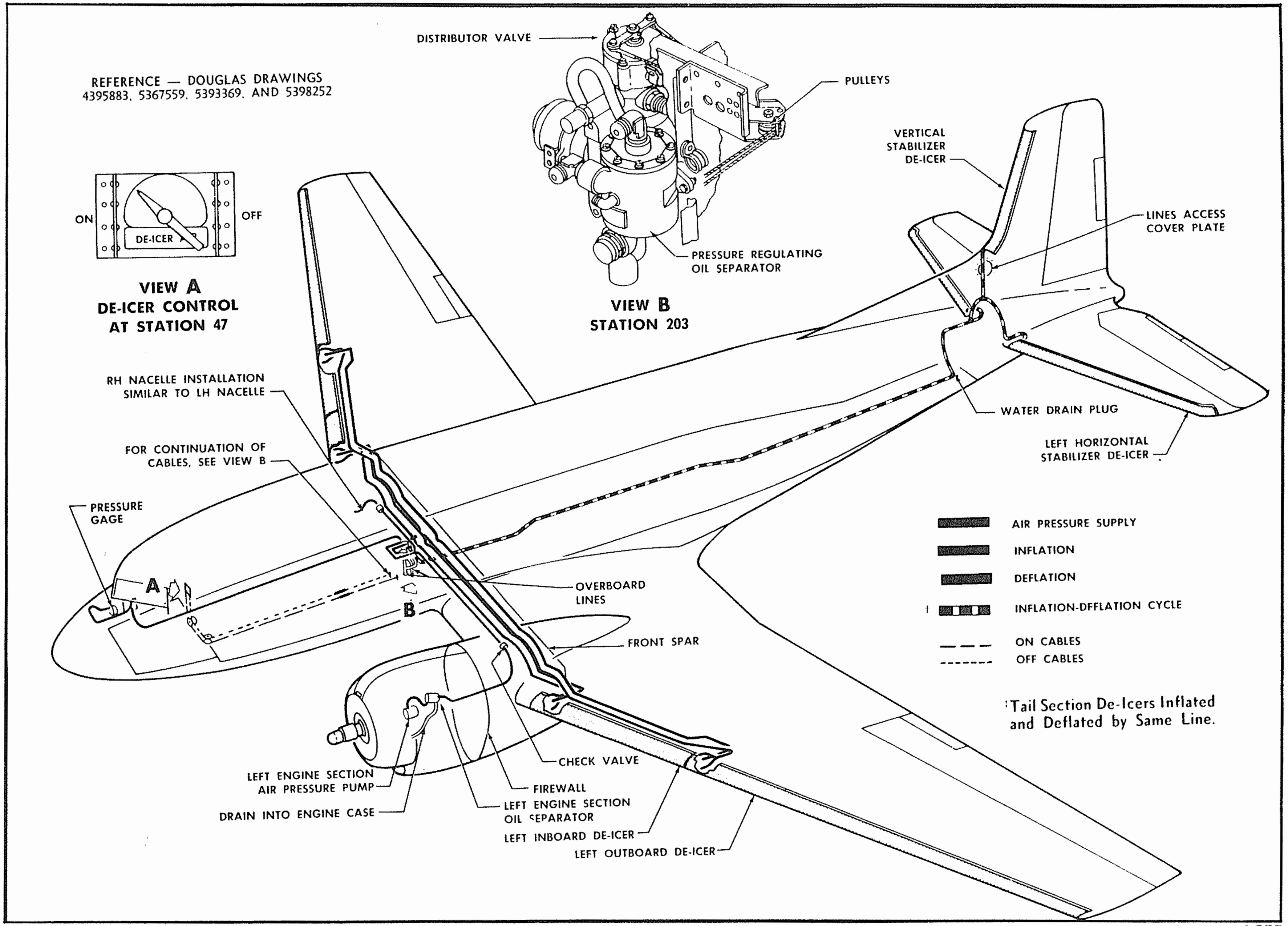


Figure 4-17. Surface De-Icing System

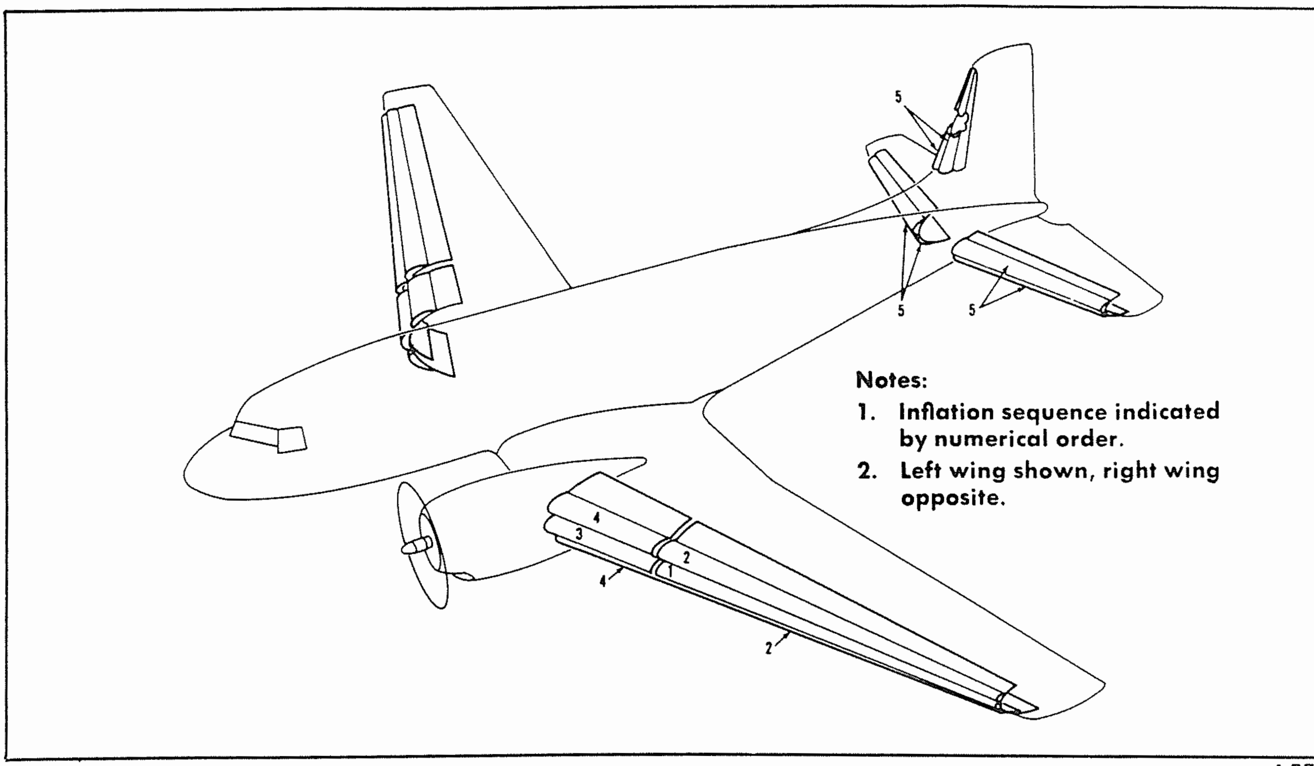


Figure 4-18. Surface De-Icer Inflation Sequence

Trouble	*Probable Cause*	*Remedy*
f. *(Continued)*	Faulty distributor valve.	Repair or replace valve.
	Pressure regulating valve in regulating oil separator inoperative in open position.	Replace regulating oil separator.
	Broken air pressure pipe or open connections.	Replace broken pipes; refit open connections.
g. Excessive discharge overboard from regulating oil separator.	Plugged firewall separator screen or orifice setting.	Remove and clean screens and fittings in firewall separator.
	Faulty pump, allowing excessive discharge of oil.	Replace pump.

4-127. TEST OF SURFACE DE-ICING SYSTEM.

a. After all de-icers have been installed and connected, attach an auxiliary air pipe through a pressure regulating valve to the connection on the main pressure supply pipe to the Eclipse distributor valve. This connection is accessible after removing a small plug. Set the pressure regulator to a maximum of 10 psi.

b. Turn the de-icing system control to ON.

c. Turn on the auxiliary air supply and observe the de-icers. Watch the air pressure on the instrument panel. If the pressure exceeds eight psi, turn to OFF immediately. Pressure readings will fluctuate with various phases of the cycle, but a peak should be reached five times during each cycle.

d. If pressure in the system is too great, turn the adjusting screw on the bottom of the Eclipse oil separator in a counterclockwise direction until the relief valve reduces the pressure to eight psi.

e. Check the pulsations of the de-icers to see that the pipes have been correctly connected according to the sequence cycle outlined in paragraph 4-125, preceding.

f. If an auxiliary air supply is not available, air must be supplied from the de-icing pumps with the engines running.

4-128. SURFACE DE-ICERS. *(See figures 4-17 and 4-19.)* The wing de-icers incorporate three inflatable tubular cells, which are located in the center area of the de-icer. Adjoining each side of the tubular section is an elastic zone, the outer margins of which are attached to the wing surface with machine screws. Fairing strips, which cover the edges of the de-icers, are held in place by the same machine screws which attach to rivnuts permanently installed in the aircraft wing. The empennage de-icers operate in a manner similar to the operation of the wing de-icers. Construction of the empennage de-icers differs from the wing de-icer construction in that only two inflatable tubes or cells are contained in each empennage de-icer center area.

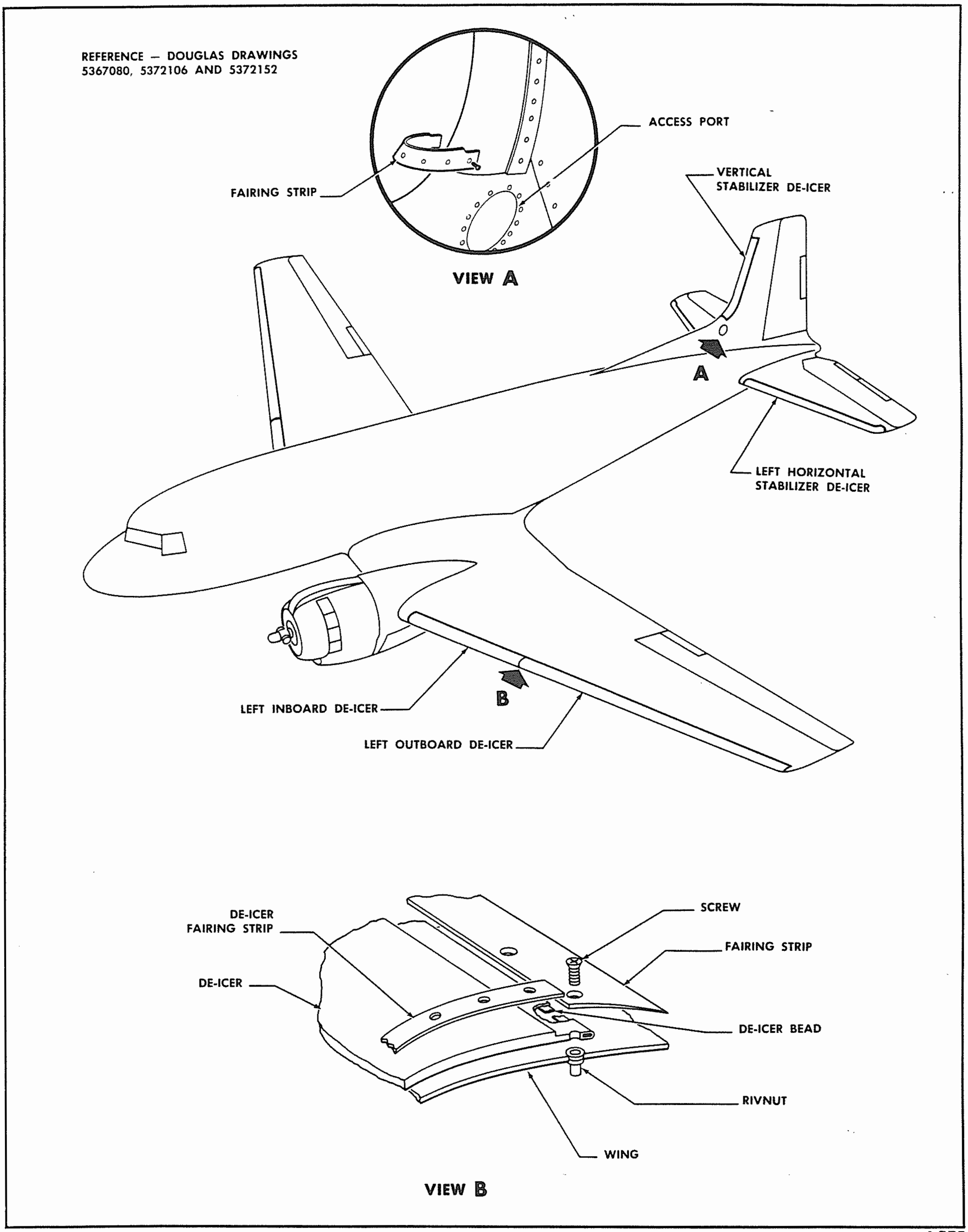


Figure 4-19. Surface De-Icer Installation

Both the horizontal and vertical stabilizer de-icers are operated simultaneously. To avoid damaging the de-icers, observe the following precautions:

a. Do not drag gasoline hoses or cables over the de-icers.

b. Avoid stepping on the de-icers at any time.

c. Install padding on ladders or platforms where they might come into contact with the de-icers.

d. Remove any engine oil from the surface of the de-icers as soon as possible after each flight. Use a soft cloth with a neutral soap-and-water solution, dry cleaning solvent, or Toluol substitute.

CAUTION

All cleaning agents must be used sparingly and wiped off immediately.

4-129. MINOR REPAIR AND REPLACEMENT OF SURFACE DE-ICERS. Frequent inspection of the de-icer surface should be made to locate any small nicks or cuts. De-icers should not be removed for the patching of cuts or tears less than 3/4 inch long. If any small nicks or cuts are found they should be immediately repaired with a cold patch, thus preventing them from becoming larger and necessitating the removal of the de-icer. Patching of the surface de-icers must be done neatly, for the de-icers cannot function properly if the surface is covered with rough patches. Always use molded de-icer patches wherever size permits, since these patches are made with feathered edges and present a minimum of discontinuity to the surface. Cold patching should be considered as a temporary measure. Since numerous cold patches greatly affect the efficiency of the de-icer, the de-icer should be removed periodically so that the cold patches can be removed and the damage permanently repaired by the vulcanizing process. For repair of surface de-icer damage that is less than 3/4 inch in length, use the materials supplied in the B. F. Goodrich Cold Patch Kit No. 3040, and proceed as follows:

a. Clean the surface in the vicinity of the damage with mild soap and water, and wipe dry.

b. Select the correct patch.

c. Place a shield of the proper size over the injury so that the cut-out portion exposes the area to be patched. Hold the shield in place throughout the following operations:

d. Rub the de-icer surface with a cloth moistened in benzol to soften and remove the conductive surface.

e. Roughen the surface with an emery buffer so that approximately 0.003 inch of the surface has been removed. Clean with benzol and allow to dry.

f. Mix a small amount of C-80-S and C-81-S cements in equal portions, and apply one heavy coat to the buffed area. Allow to dry.

g. Remove the fabric backing from the patch and apply one coat of the cement mixture to the exposed surface. Keep the tacky surface of the patch clean after removing the fabric and cementing. Allow to dry. Unlike glue, rubber cement must be thoroughly dry before a patch is applied.

h. Apply the patch to the de-icer.

i. Stick the edge of the patch in place, working the remainder down with a slight pulling action so that the injury is closed. Do not trap air between the patch and the de-icer surface.

j. Roll down the patch with a roller.

k. If the edge is not tight, re-cement and allow to dry. Then roll down.

l. Allow the repair to set 10 to 15 minutes. Wipe the patch and the surrounding area lightly with benzol to remove all excess cement.

m. Apply a coat of B. F. Goodrich Conductive Cement to restore the conductive surface.

4-130. When injuries are over 3/4 inch long, relieve the tension on the de-icer by removing several attachment screws in the area to be repaired. Sheet patching material, reinforced with fabric, is furnished in the De-Icer Cold Patch Repair Kit for large patches. For proper adhesion, these should be cut larger than the injury. This patch material is manufactured so that it will stretch in one direction only. Be sure to cut and apply the patch so its stretch is chord-wise (in the width-wise direction of the de-icer). The application procedure is similar to that outlined in paragraph 4-129, preceding.

4-131. In some instances when there are holes or where a portion of the de-icer has been torn out, a temporary emergency repair may be made. However, it is necessary to place an additional reinforcement on the inside surface. In this case, the following procedure should be used:

a. Remove the de-icer as necessary for accessibility.

b. Buff an area around the edge of the hole on the inner surface until the ribs on the surface have been removed. An area approximately 1½ inches from the hole should be sufficient.

c. Cut a patch of correct size from the roll of fabric-reinforced patching material.

d. Apply as outlined in preceding paragraph 4-129.

e. Turn the de-icer over and repeat the same procedure on the outer surface, using the reinforced sheet patching material.

f. Apply one coat of conductive cement to the outer patch.

g. Dust the inner patch with soapstone.

h. Allow about 1/2 hour to thoroughly dry, and then install the de-icer.

4-132. REMOVAL OF SURFACE DE-ICERS.

a. Remove all screws, clamps, and fairings.

b. As each fairing strip is removed, tag it for position so that it can be reinstalled in its proper place.

c. Disconnect the hose and remove the de-icer from the wing.

d. Plug the hose and mark for identification with the corresponding de-icer tube.

e. Secure the hose inside the wing so that it will be readily accessible when the de-icers are reinstalled.

f. Remove the sealing sheet of rubber from the aircraft and slip it over the de-icer connections. Connection holes in the wing skin should be plugged with special rubber grommets or covered with metal plates.

g. Plug screws should be inserted in each rivnut to keep out foreign material. Any damaged, worn, or loose rivnuts should be drilled out and replaced.

4-133. SURFACE DE-ICER PRE-INSTALLATION PROCEDURES. To facilitate the installation of the de-icers, the following preparations should be made.

4-134. PREPARATION OF LEADING EDGES.

a. If the leading edge is coated with camouflage paint, the area that the de-icers will cover must be rubbed down smooth in order to facilitate installation.

b. If the leading edge is not painted, it should be sprayed with a coating of zinc-chromate primer to prevent any corrosion of the aircraft skin beneath the de-icer.

c. The surface should then be inspected for burred screw heads or sharp skin laps. If these are present, they should be smoothed down to prevent possible de-icer injury.

d. A thin, liquid mixture of talc and non-leaded gasoline should be brushed on quickly. The gasoline will evaporate, leaving a smooth, even film of talc that lubricates the skin for the movement of the de-icer. It must be allowed to dry thoroughly. Other suitable solvents may be used, including water. With water, however, the drying time is considerably lengthened.

4-135. PREPARATION OF FAIRING STRIPS. To prevent chafing of the wing skin, a narrow strip of non-surgical adhesive tape should be placed on the trailing edge of the under side of the fairing. This tape should not extend beyond the fairing.

4-136. PREPARATION OF SURFACE DE-ICERS.

a. To locate the attachment holes to be punched in a new de-icer, a center line is drawn from the top rivnut attachment line through the connection holes in the wing to the bottom rivnut line. The rivnut closest to this line is chosen and its relative position, with respect to the de-icer connections, is marked on the de-icer bead.

b. The proper fairing strip should then be laid along the edge of the de-icer with the corresponding hole for the rivnut matching the mark just made on the de-icer. The remainder of the attachment holes in the de-icer are then marked from the fairing strip.

c. Punch the de-icer attachment holes through the fabric-reinforced rubber at the edge of the de-icer directly beside the metal bead, being careful not to nick the bead wire.

d. Do not punch holes for attachment where the rivnut line is curved. This is done as the actual installation progresses, and only after the de-icer has been slipped along its wire beads until the bead assemblies conform to the curved attachment line. The puckers formed must be at right angles to the leading edge. The de-icer should be slipped along its beads prior to the installation of the straight portions and this can best be done by clamping the end of the bead wire in a bench vise.

e. The back of the de-icer should be thoroughly dusted with talc. Do not use a liquid mixture of talc on the de-icer.

4-137. INSTALLATION OF SURFACE DE-ICERS. *(See figures 4-20 and 4-21.)*

a. Install studs in alternate rivnuts along the upper surfaces of the wings and horizontal stabilizers, and along one side of the vertical stabilizer.

b. Remove cover plates from the leading edges for air pipe connections at the inboard ends of the surface de-icers.

c. Completely attach the de-icers to the lower sides of the wings and horizontal stabilizers and to one side of the vertical stabilizer, inserting attaching screws in all rivnuts. Holes for the screws must be pierced through the rubber. Non-surgical adhesive tape should be applied to the stabilizer and wing skin at the point of contact of the metal fairing strip. No tape should be applied to the fairing strip itself at this point. Non-surgical adhesive tape should be applied to the under side of the metal fairing strip at its leading edge only. No tape should be applied to the rubber de-icer.

d. Place a liberal amount of talc on the inside surface of the de-icer and on the skin of the surface to which it is being attached. This operation facilitates the movement of, and prevents the seizure of, the de-icer.

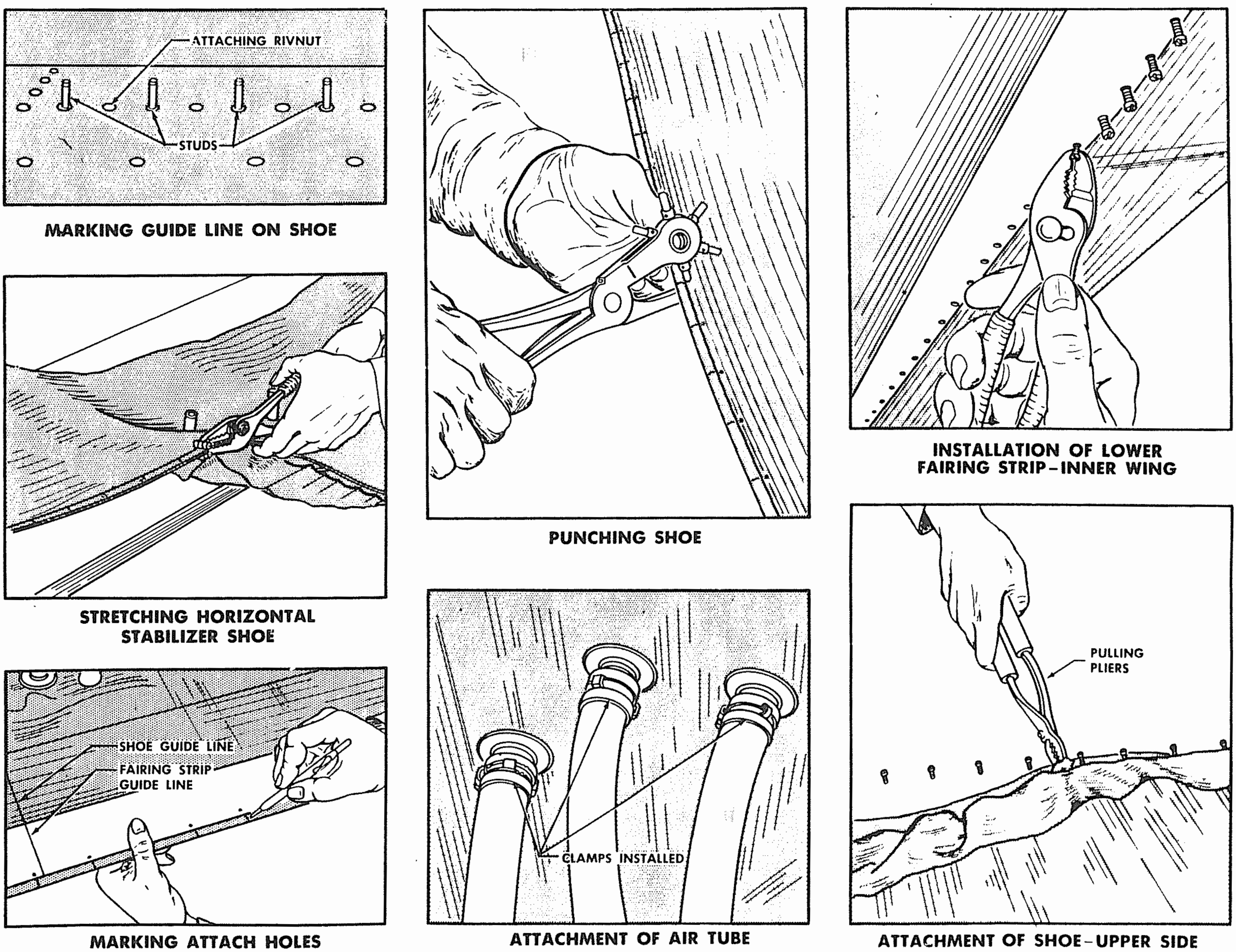


Figure 4-20. Surface De-Icer Installation Procedure

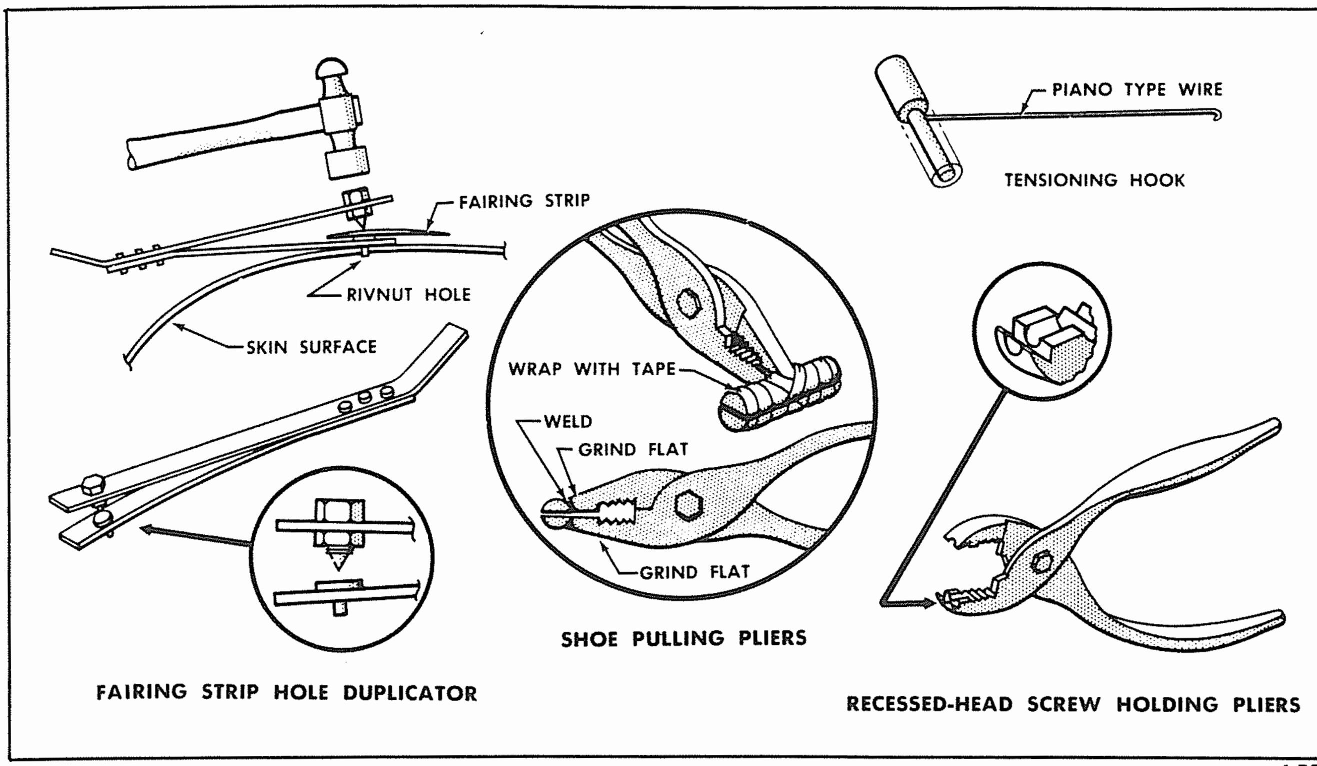


Figure 4-21. Installation Tools for Surface De-Icers

e. Attach the tubes to the de-icer pipes.

f. Stretch the de-icer over the wing leading edge and hook it over the studs on the upper surface. As the rubber is pulled over each stud, it should be immediately pushed down against the skin to lessen the strain on the stud.

g. Place fairing strips over studs and insert screws in the remaining rivnuts. Replace studs with screws and tighten.

h. The vertical stabilizer de-icer is installed in a similar manner. Attach one side; connect the tubes to the de-icer pipes; then, starting at the bottom, attach the other side of the surface de-icer to the vertical stabilizer.

4-138. SURFACE DE-ICING PUMPS. Two Type B-12, engine-driven, rotary, four-vane, positive-displacement pumps supply air pressure for the surface de-icing system. One pump is mounted on the rear case of each engine.

4-139. REMOVAL OF SURFACE DE-ICING PUMP.

a. Disconnect the pressure and air inlet pipes.

b. Remove bolts attaching the pump to the rear case of the engine.

c. Remove the pump.

4-140. INSTALLATION OF SURFACE DE-ICING PUMP. Reverse the removal procedure.

4-141. SURFACE DE-ICING OIL SEPARATORS. *(See figure 4-22.)* Two engine section oil separators are installed in the de-icing system. One is mounted on the engine mount within each engine accessory section just above the de-icing pump. The oil separator removes the major portion of the oil suspended in the air and returns it to the engine case.

4-142. REMOVAL OF SURFACE DE-ICING OIL SEPARATOR.

a. Disconnect the inlet and outlet pipes.

b. Remove the bolts attaching the oil separator to the mounting bracket on the engine mount and remove the separator.

4-143. INSTALLATION OF SURFACE DE-ICING OIL SEPARATOR. Reverse the removal procedure.

4-144. SURFACE DE-ICING PRESSURE REGULATING OIL SEPARATOR. *(See figure 4-23.)* A pressure regulating oil separator is installed in the wing center section. In addition to separating the remaining traces of oil from the air, this unit also prevents overloading of the de-icer pumps by relieving excessive air pressure. Pressure in the de-icer system is limited by an adjustable spring and plunger type relief valve provided in the bottom of the wing center section separator. Oil is removed from the air passing through the separator by a removable copper mesh filter. The oil collects in the bottom of the separator and drains overboard. A small port is provided on the separator for the connection of the system pressure gage pipe.

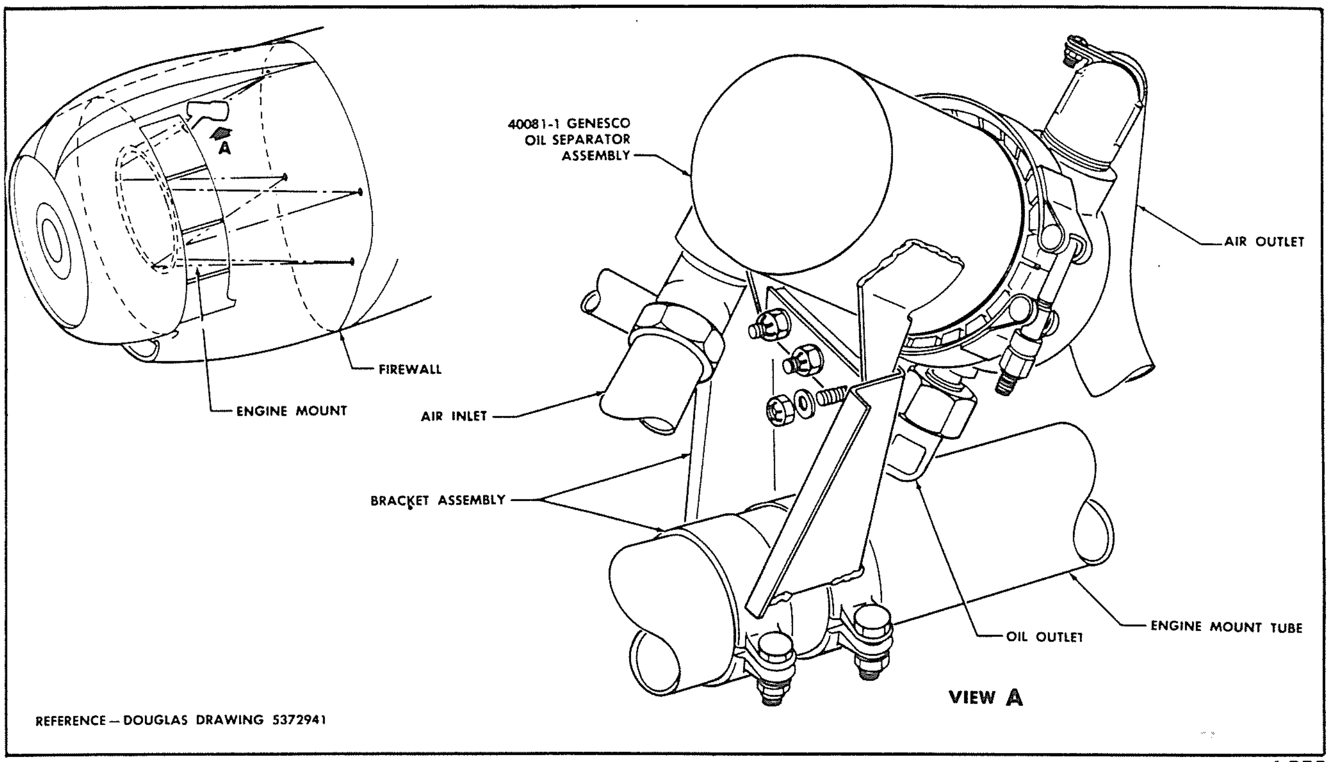


Figure 4-22. Surface De-Icer Oil Separator Installation

4-145. REMOVAL OF SURFACE DE-ICING PRESSURE REGULATING OIL SEPARATOR.

a. Disconnect the inlet and outline pipes from the oil separator.

b. Remove the bolts attaching the separator to the structure and remove the separator.

4-146. INSTALLATION OF SURFACE DE-ICING PRESSURE REGULATING OIL SEPARATOR. Reverse the removal procedure.

4-147. ADJUSTMENT OF SURFACE DE-ICING PRESSURE REGULATING OIL SEPARATOR.

a. Operate all engines at 1500 rpm.

b. Place the de-icers in operation by moving the wing de-icer to the ON position.

c. Break safety wire and loosen the lower hollow hexagonal nut located on the side of the regulating oil separator.

d. Adjust the setting of the relief valve so that the maximum reading on the de-icer pressure gage on the main instrument panel reads 8 ($\pm$ 1/2) psi. Turn the separator slotted shaft clockwise to increase the pressure gage reading, and counterclockwise to decrease the reading.

Note

In the event that the proper setting cannot be obtained, remove the relief valve and replace it with a new one.

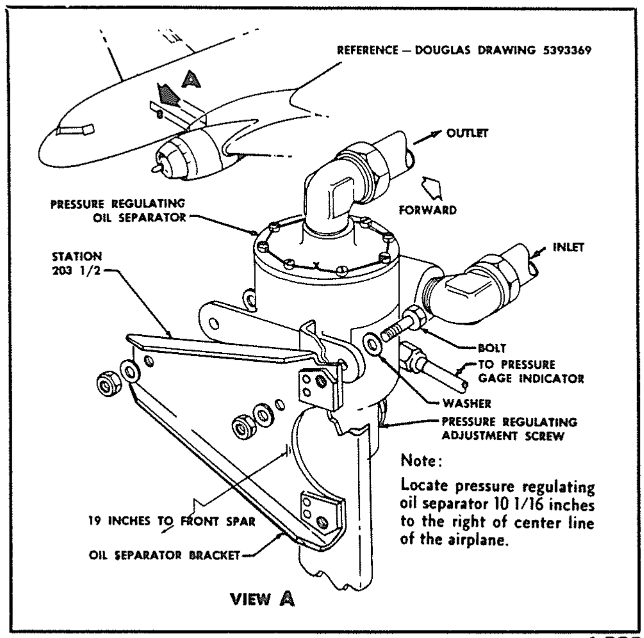


Figure 4-23. Surface De-Icer Pressure Regulator Oil Separator

4-148. SURFACE DE-ICING SYSTEM PRESSURE GAGE. *(See figure 4-17.)* A gage to indicate the pressure in the de-icer system is mounted on the main instrument panel. It is connected to the regulating oil separator in the wing center section.

4-149. REMOVAL OF SURFACE DE-ICING SYSTEM PRESSURE GAGE.

a. Remove the screws securing the instrument to the panel.

b. Remove the instrument from the panel and disconnect the pipes.

c. Remove the gage.

4-150. INSTALLATION OF SURFACE DE-ICING SYSTEM PRESSURE GAGE. Reverse the removal procedure.

4-151. SURFACE DE-ICING DISTRIBUTOR VALVE. *(See figure 4-24.)* The surface de-icing system distributor valve is located in the wing center section. It controls the flow of air for the inflation and deflation of the de-icers. The complete assembly consists of a rotor-type air valve that is driven by an electric motor through reduction gearing. The operating positions of the valve and the sequences of de-icer inflation are determined by the positions of the rotor. One revolution of the rotor completes the full cycle of operation for the de-icer system. The electric motor is controlled by a switch incorporated on the valve itself. The motor and valve are put in operation by moving the surface de-icer control to the ON position. However, when the control is moved to the OFF position, the motor continues to operate until the switch on the valve opens, which occurs when the rotor reaches the pump exhaust port position. In this position the air from the pumps is discharged overboard. When the surface de-icer control is OFF, the rotor will be in the pump exhaust position and, therefore, no pressure will be indicated on the gage.

4-152. REMOVAL OF SURFACE DE-ICING DISTRIBUTOR VALVE.

a. Remove the control cables from the crank.

b. Remove the attached piping.

c. Disconnect the electrical connection to the motor.

d. Remove the mounting bolts.

e. Remove the distributor valve.

4-153. INSTALLATION OF SURFACE DE-ICING DISTRIBUTOR VALVE. Reverse the removal procedure; then tension the cables in accordance with the Cable Rigging Tension Chart, figure 2-9, and safety the turnbuckles.

4-154. SURFACE DE-ICING CHECK VALVES. Check valves in the individual pressure pipes from each pump prevent the loss of air pressure in the event of pump failure or pipe damage in either of the engine sections. The check valves are located in the nacelle sections aft of the firewall.

4-155. REMOVAL OF SURFACE DE-ICING CHECK VALVE.

a. Disconnect the attaching pipes.

b. Remove the check valve.

4-156. INSTALLATION OF SURFACE DE-ICING CHECK VALVE. Reverse the removal procedure.

4-157. SURFACE DE-ICING CONTROLS. *(See figure 4-25.)* The surface de-icing controls incorporate a two-way cable system that extends aft from the de-icer control panel, located on the forward side of the bulkhead aft of the co-pilot's seat, to the de-icer pressure distributor valve at station 203. The distributor valve, controlling the flow of air for the de-icers, is turned on or off by the operation of the control on the de-icer control panel.

4-158. SURFACE DE-ICING CONTROL CABLES. *(See figure 4-25.)* The surface de-icing control cables extend from the de-icer control panel bellcrank at station 47 aft to the crank on the distributor valve at station 203.

4-159. REMOVAL OF SURFACE DE-ICING CONTROL CABLES.

a. Remove the right center floor panel between stations 156 and 195.

b. Remove the safety wire and disconnect the two turnbuckles located just outboard of the pulley bracket at station 203. Thread the cable ends for removal.

c. Remove the cable guard pins from the pulley bracket at station 203.

d. Disconnect the rear section of both the ON and OFF cables from the bellcrank at the top of the distributor valve.

e. Remove the rear section of the cables through the floor panel opening.

f. Remove the center floor panel adjacent to the hydraulic panel and the right floor panel over station 97.

g. Remove the necessary cable grommets from the fairleads at stations 156 and 97.

h. Remove the cable guard pins from the pulley brackets at station 47.

i. Disconnect the forward ends of the ON and OFF cable assemblies from the bellcrank on the control assembly located on the forward side of the bulkhead at station 47.

j. Remove the forward section of the cables through the floor opening adjacent to the hydraulic panel.

(Continued on page 307)

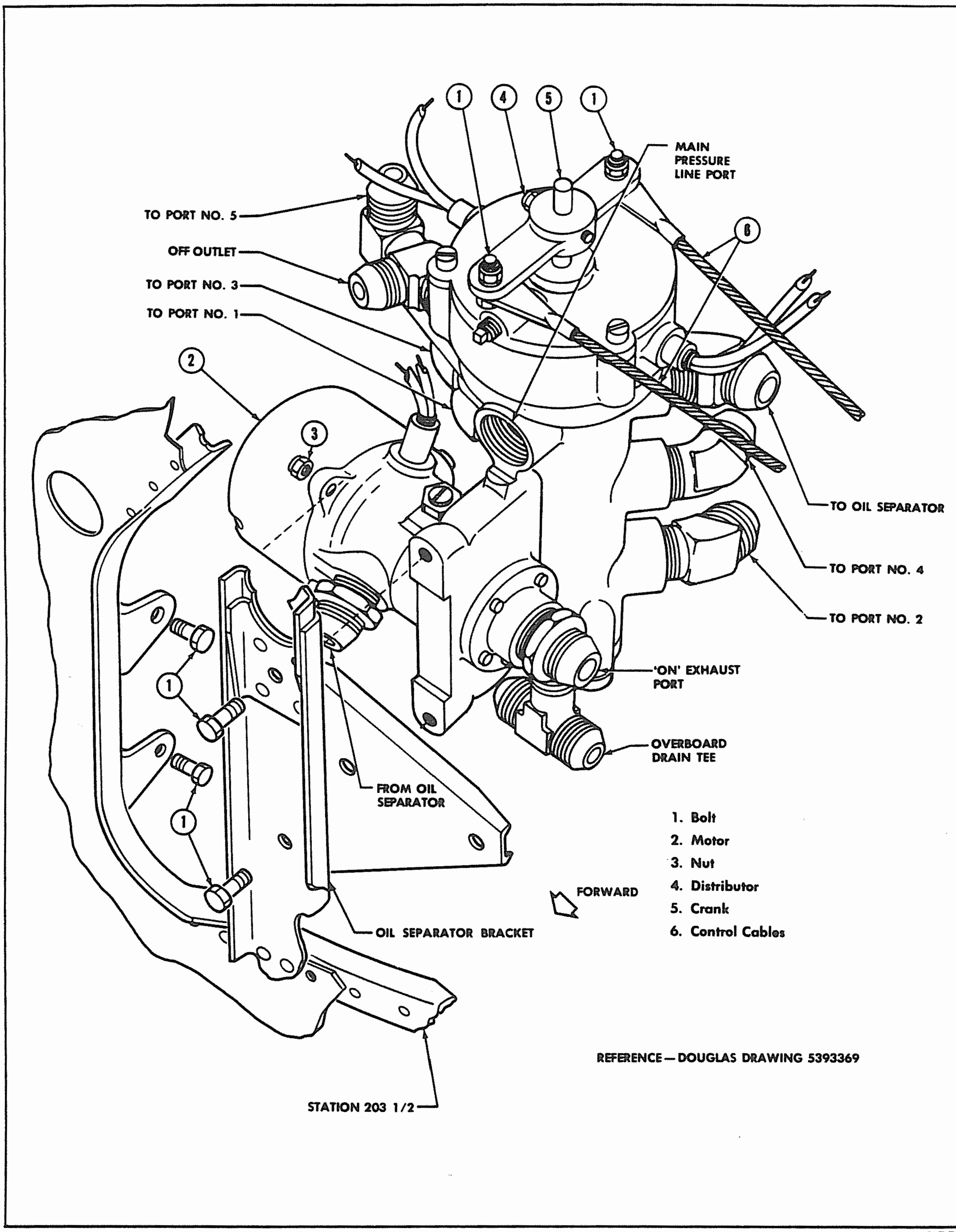
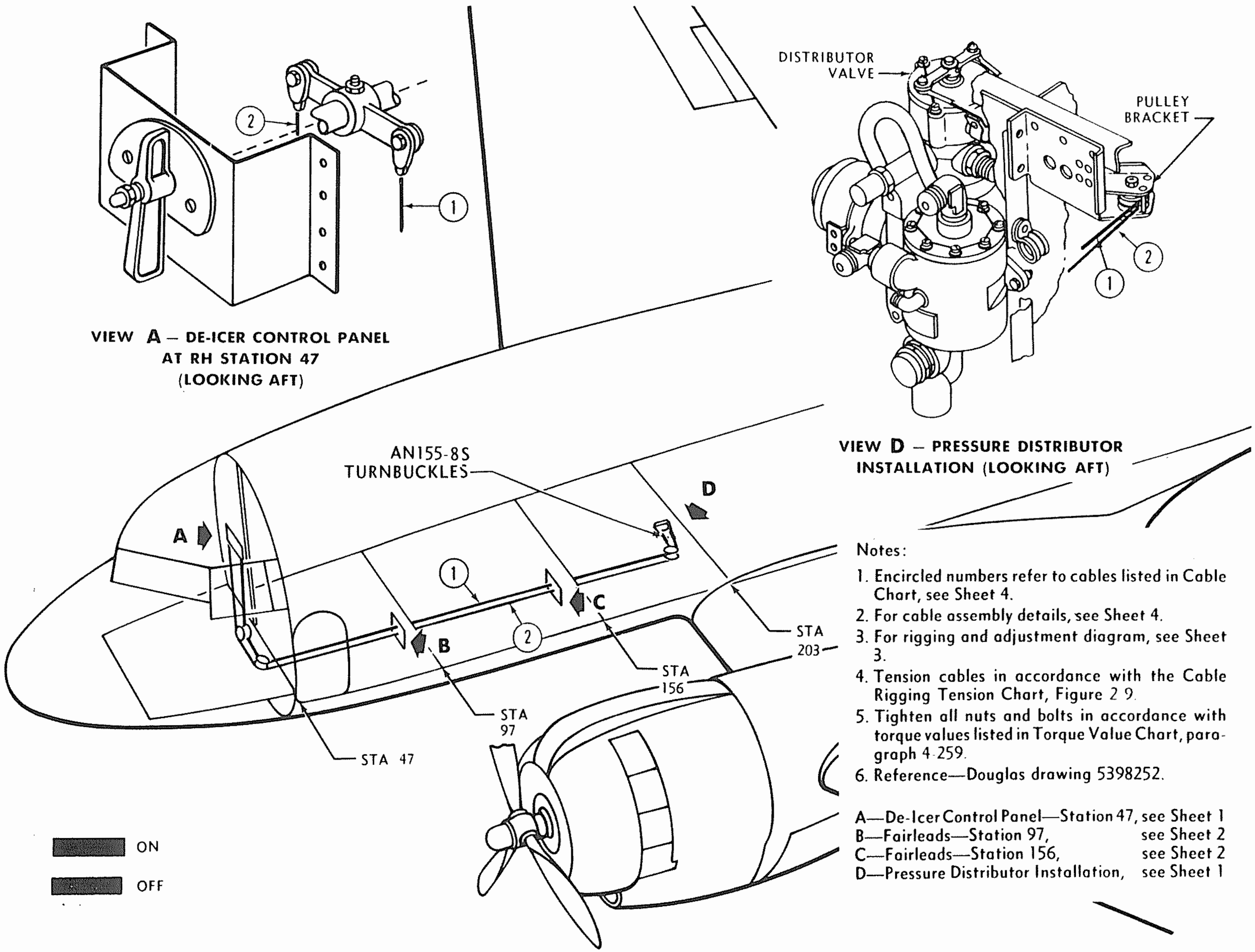


Figure 4-24. Surface De-Icer Distributor Valve



Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 4.
2. For cable assembly details, see Sheet 4.
3. For rigging and adjustment diagram, see Sheet 3.
4. Tension cables in accordance with the Cable Rigging Tension Chart, Figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart, paragraph 4-259.
6. Reference—Douglas drawing 5398252.

A—De-Icer Control Panel—Station 47, see Sheet 1
B—Fairleads—Station 97, see Sheet 2
C—Fairleads—Station 156, see Sheet 2
D—Pressure Distributor Installation, see Sheet 1

Figure 4-25 (Sheet 1 of 4 Sheets). Surface De-Icing Control System — Key Drawing

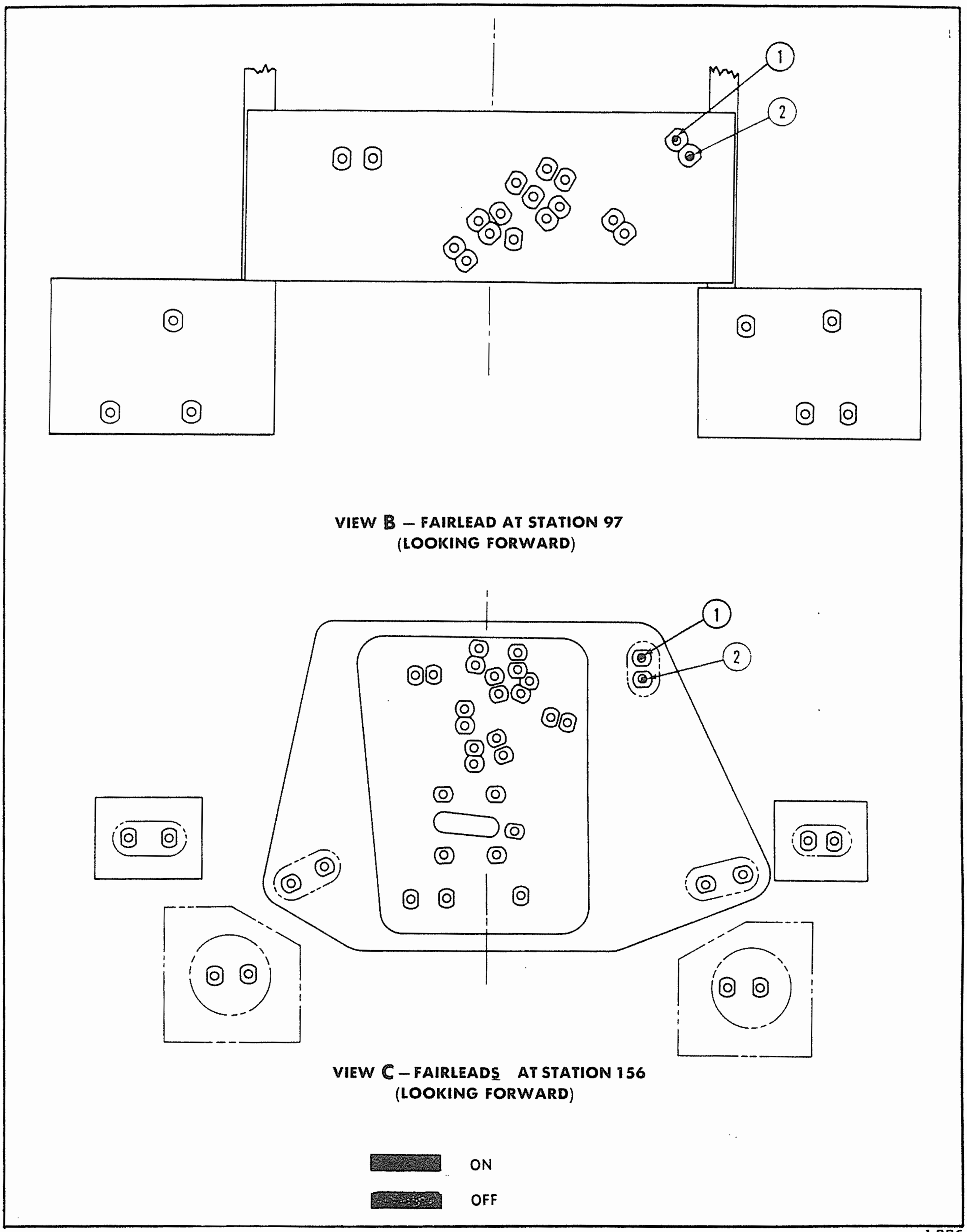
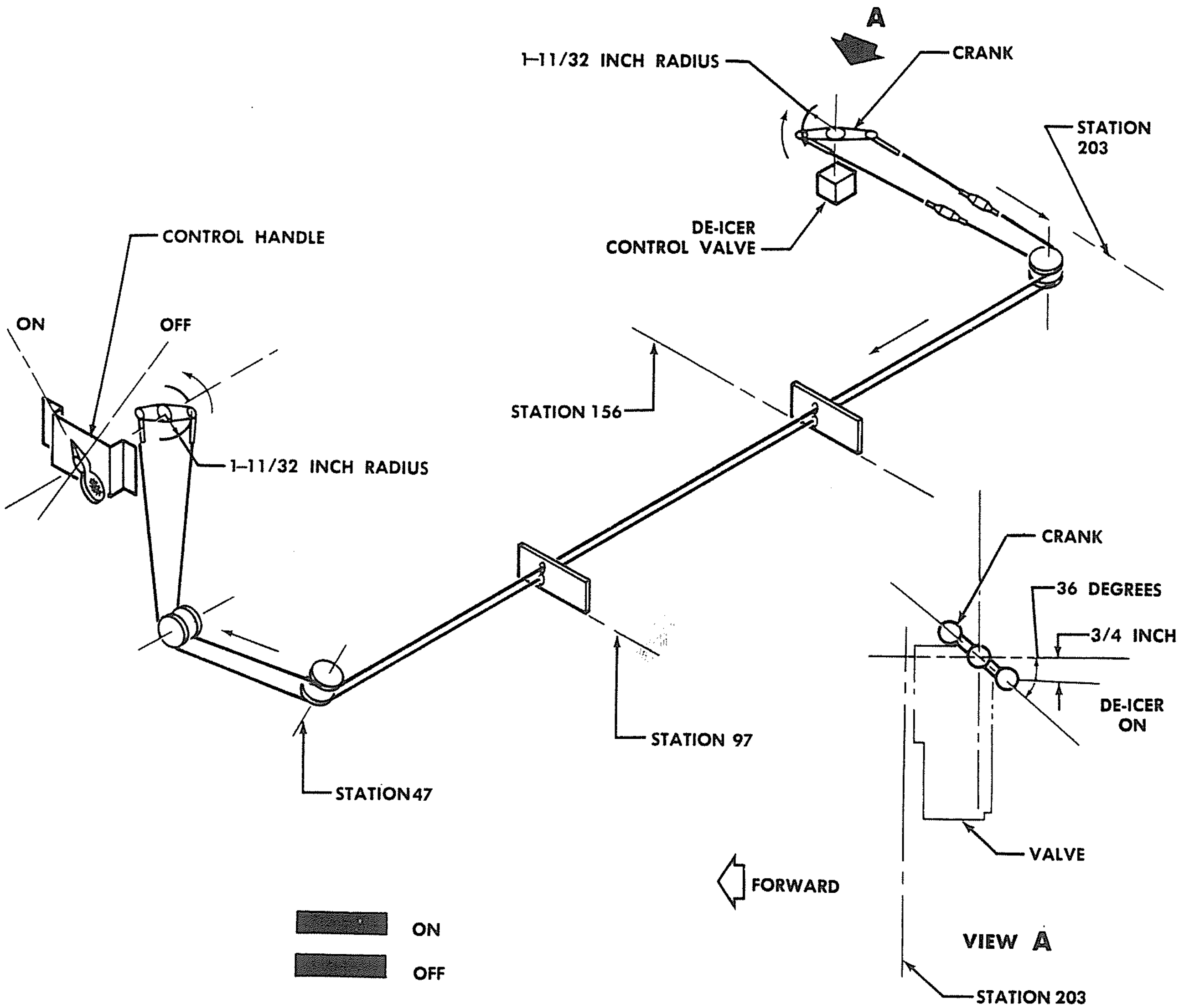


Figure 4-25 (Sheet 2 of 4 Sheets). Surface De-Icing Control System — Fairleads, Stations 97 and 156



DE-ICER ADJUSTMENT INSTRUCTIONS

1. Set de-icer control valve to the ON position, (rotated full clockwise against the integral stops).
2. Install crank to dimension shown in View A.
3. Set control handle to the ON position against the stop and tension the cables, with the turnbuckles at station 203, in accordance with the Cable Rigging Tension Chart, see figure 2-9.
4. Turn control handle to the OFF position (rotated clockwise until the handle bears against the opposite end of the stop). The de-icer valve should be in full OFF position (rotated counterclockwise to the integral stop).
5. Safety the turnbuckles.

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Figure 4-25 (Sheet 3 of 4 Sheets). Surface De-Icing Control System — Adjustment Diagram

SURFACE DE-ICING CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQD.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)
1	4398798-1 De-icer on	1	A	(L_1) 162 1/2 (L_2) 43	1/16 dia 7x7 flex	1111636	AN669S2 RH	AN669S2 LH	1111636
2	4398798-501 De-icer off	1	A	(L_1) 152 1/2 (L_2) 53 1/4	1/16 dia 7x7 flex	1111636	AN669S2 RH	AN669S2 LH	1111636

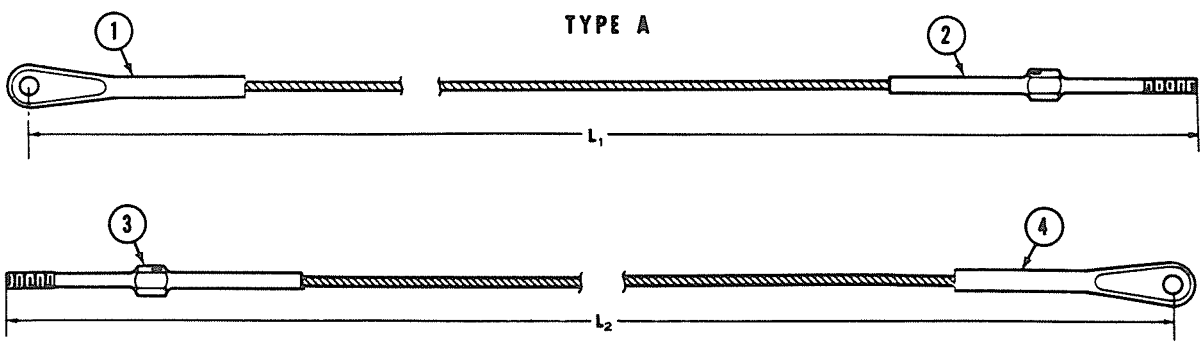

Figure 4-25 (Sheet 4 of 4 Sheets). Surface De-Icing Control System — Cable Chart and Cable Assemblies

(Continued from page 302)

4-160. INSTALLATION OF SURFACE DE-ICING CONTROL CABLES.

a. Connect the forward surface de-icing control cables to the bellcrank on the control assembly, located on the forward side of the bulkhead at station 47, and route them aft through the aircraft, installing guard pins and fairlead grommets as necessary.

b. Attach the aft surface de-icing control cables to the bellcrank at the top of the distributor valve and route them forward, installing necessary guard pins.

c. Connect the cables just outboard of the pulley bracket at station 203 with turnbuckles.

d. Tension cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

e. Safety the turnbuckles.

f. Replace the floor panels in the aircraft.

4-161. ADJUSTMENT OF SURFACE DE-ICING CONTROLS.

(See figure 4-25, Sheet 3.)

4-162. SURFACE DE-ICING SYSTEM BOLT TORQUE VALUES. Unless otherwise noted, see paragraphs 4-259 and 4-260 for bolt torque values.

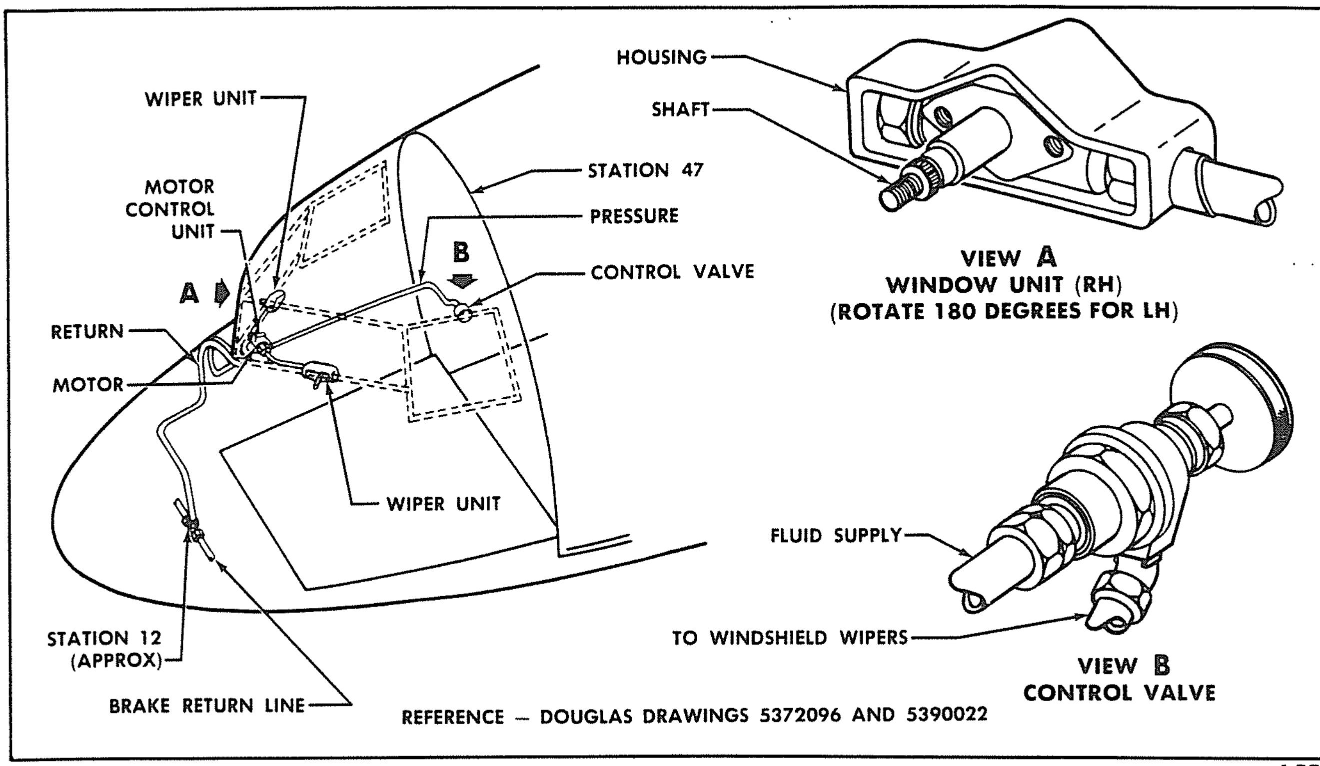

Figure 4-26. Windshield Wiper System

4-163. WINDSHIELD WIPER SYSTEM.

4-164. DESCRIPTION. *(See figures 4-26 and 4-27.)* Hydraulically actuated windshield wipers are provided on the pilots' windshields. Hydraulic pressure for the operation of the windshield wipers is obtained directly from the pressure regulator or the accumulator. Fluid pressure is directed to the windshield wiper control valve, located below the hydraulic reservoir sight gage, on the hydraulic control panel, which controls the operating speed of the wiper blades. From the control valve the fluid flows to the window actuator units. The returning fluid from the windshield wiper actuators flows to a tee in the brake return pipe, from which point it is returned to the reservoir. To operate the wipers, slowly open the control valve until the desired speed of the blades is obtained. If icing conditions exist, the wiper blades should be started immediately and heat applied to the windshield by opening the windshield anti-icing control valve, located at the base of the flight compartment bulkhead to the left and aft of the co-pilot's seat. Stop the windshield wipers with the blades pointing in their most inboard position.

4-165. TROUBLE SHOOTING OF WINDSHIELD WIPER SYSTEM. Hydraulic failure of the windshield wiper system may be traced to any of the three following sources:

External leakage of fluid.

Internal leakage of fluid within the system.

Mechanical failure of parts in the system.

CAUTION

Install the alighting gear ground locks before attempting any trouble shooting operations on the hydraulic system.

4-166. Before attempting any trouble shooting, make certain that the air pressure in the accumulator and the fluid in the reservoir is sufficient. The hydraulic system must be free of air for proper operation. Bleed out the system completely by operating its units through several cycles, with power supplied by a test stand. Following any rework which involves the breaking of a fitting or pipe, the system must be bled before proper operation can be restored.

Trouble	*Probable Cause*	*Remedy*
a. System inoperative.	Lack of fluid in reservoir.	Check fluid quantity.
	External leaks.	Make visual check.
	Internal leaks.	Replace faulty unit.
b. Faulty or erratic operation of the blades.	Faulty window unit.	Replace faulty unit.
	Faulty control valve.	Replace control valve.
	Faulty actuator unit.	Replace faulty unit.
	Distorted flexible actuating shaft.	Replace shaft.

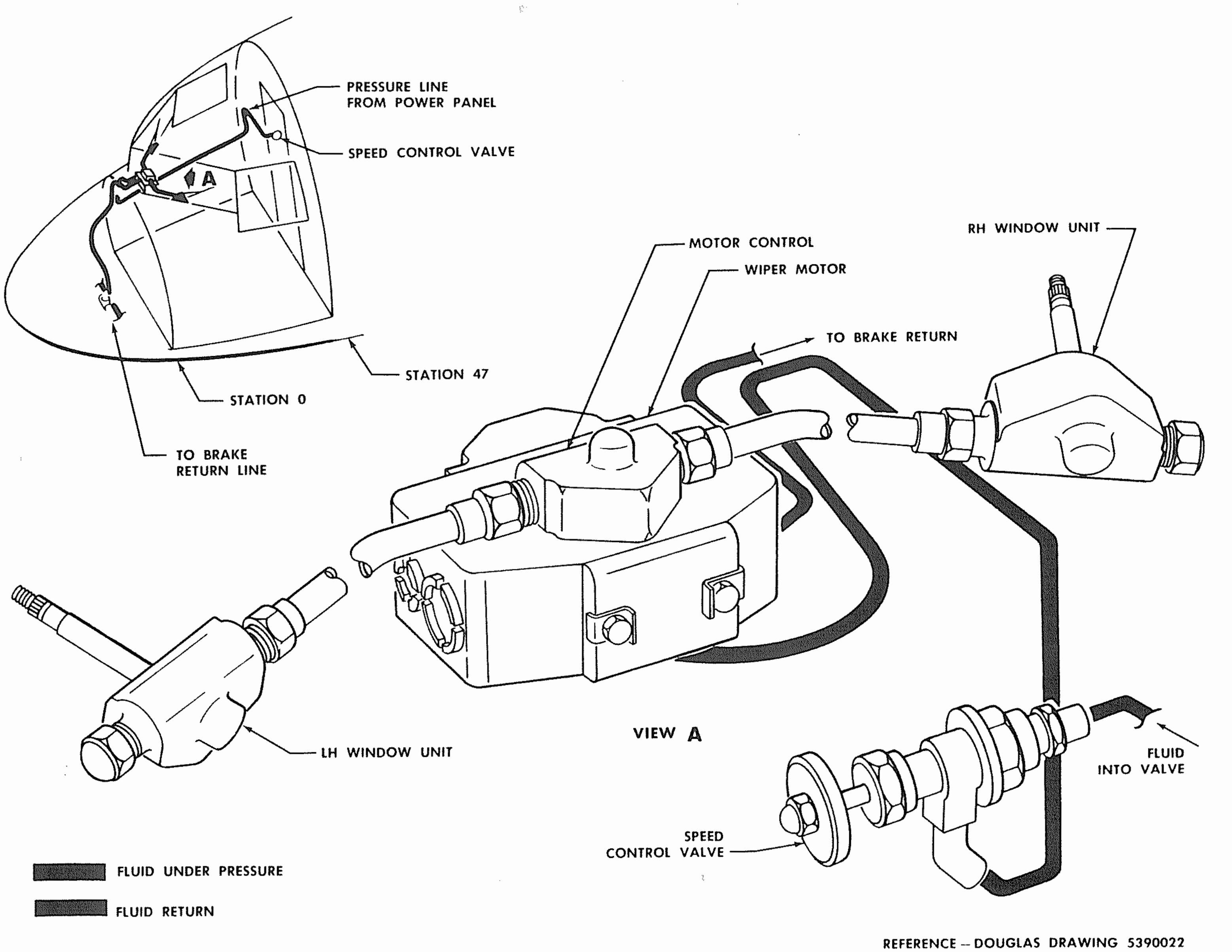

Figure 4-27. Windshield Wiper Hydraulic System

4-167. WINDSHIELD WIPER CONTROL VALVE. The windshield wiper control valve, located below the hydraulic reservoir sight gage on the hydraulic system power panel, is a needle-type valve that controls the quantity of hydraulic fluid that flows to the windshield wiper motor and consequently controls the speed of the windshield wipers. The control valve is provided with an orifice that prevents overspeeding of the wiper motors *(see figure 4-26).*

4-168. REMOVAL OF WINDSHIELD WIPER CONTROL VALVE.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and drain the hydraulic fluid from the system.

b. From the hydraulic system power panel, disconnect and cap the pipes leading to the windshield wiper speed control valve.

c. Remove the valve mounting screws from the face of the hydraulic panel and remove the valve.

4-169. INSTALLATION OF WINDSHIELD WIPER CONTROL VALVE. Reverse the removal procedure.

4-170. WINDSHIELD WIPER ACTUATING UNITS. The windshield wiper actuating units are located on top of the glareshield, behind the windshield unit. The windshield wiper actuating units consist of one motor and attached motor control unit, which serve both of the wiper blades. The hydraulically actuated motor is geared to the control unit, which operates the blades by means of flexible actuating shafts connected to each window unit *(see figure 4-26).*

4-171. REMOVAL OF WINDSHIELD WIPER ACTUATING UNITS.

a. Relieve the hydraulic system pressure by operating the wing flaps until the hydraulic system pressure gage indicates zero, and drain the hydraulic fluid from the system.

b. Disconnect and cap the hydraulic pipes leading to the hydraulic motor.

c. Disconnect the tubing and flexible shafts at the motor control unit.

d. Remove the actuating units.

4-172. INSTALLATION OF WINDSHIELD WIPER ACTUATING UNITS. Reverse the removal procedure.

4-173. TEST AFTER INSTALLATION OF WINDSHIELD WIPER ACTUATING UNITS.

a. Inspect the motor and hydraulic pipes for hydraulic fluid leaks. This should be done by operating the wiper blades very slowly.

CAUTION

Wet the windshield with water before operating the wiper blades. Never operate the blades on dry glass.

b. While the wiper blades are operating, check to see that the blade pressure is constant over the entire range and that the distance from the blade to the lower windshield is approximately equal at both extremes of the blade travel. The wiper blade must not touch the lower frame of the windshield in either of its extreme positions. If it does, the wiper arm is out of time with the drive motor.

c. Turn off the control valve. If the wiper does not make a clean sweep throughout the full arc of operation, inspect the wiper blades for damage and deterioration. Test the spring tension of the blades with a spring scale. Lift the blades with a scale hooked under the wiper arm at the center portion of the wiper blade. Lift up until the blade barely leaves the windshield. The spring scale should indicate 32 ounces. The spring tension can be adjusted by means of the screw at the base of the wiper-arm assembly.

4-174. WINDSHIELD WIPER SYSTEM BOLT TORQUE VALUES. Unless otherwise specified, see paragraphs 4-259 and 4-260 for bolt torque values.

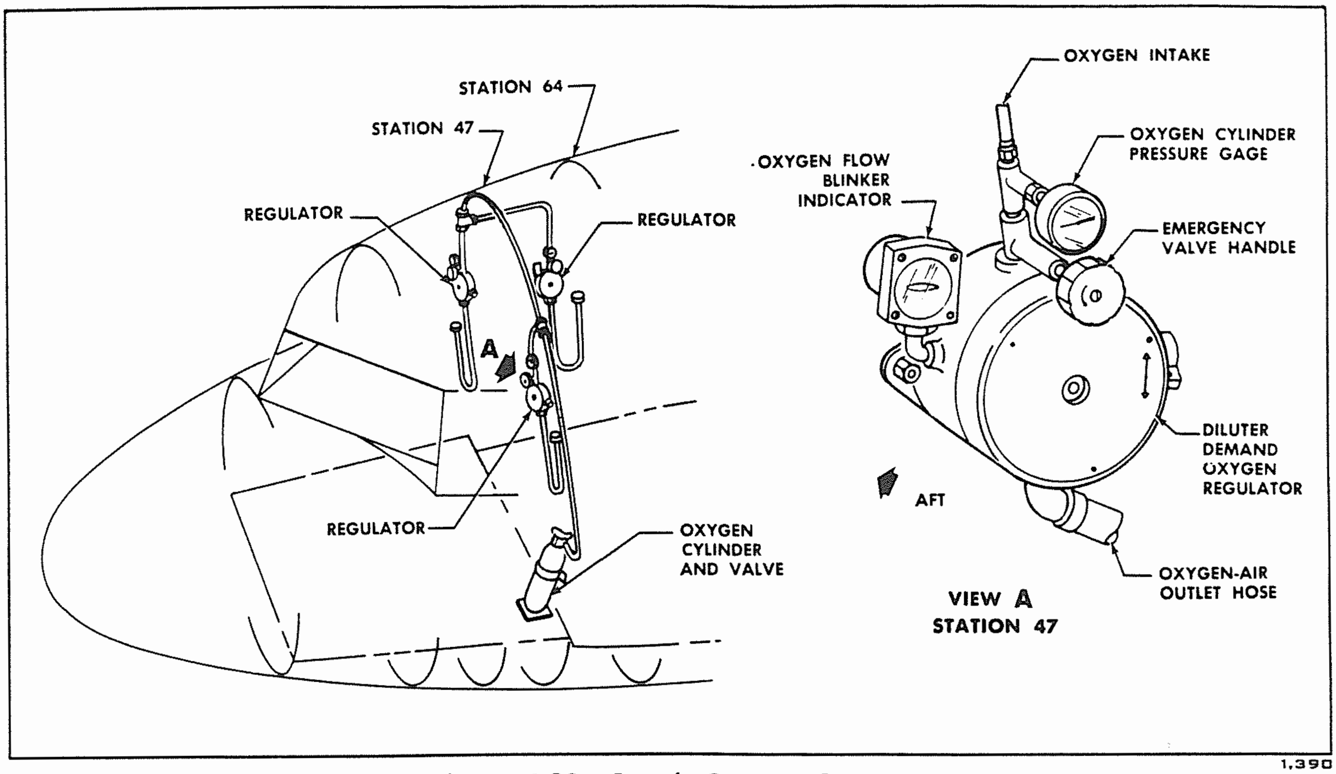

Figure 4-28. Crew's Oxygen System

4-175. OXYGEN SYSTEM.

4-176. DESCRIPTION. *(See figure 4-28.)* A high-pressure diluter-demand oxygen system supplied from one cylinder is installed in the aircraft. The oxygen system components consist of three oxygen outlets connected by rigid pipes, a 295-cubic-inch capacity supply cylinder mounted in the cockpit, three diluter-demand oxygen regulators equipped with integral gages, flow indicators, and the necessary pipes and fittings. The oxygen system, charged at approximately 1800-psi pressure, is capable of supplying 22 cubic feet of oxygen at atmospheric pressure for maintenance of flow requirements to three crew members for approximately one hour at an altitude of 15,000 feet *(see figure 4-29).*

4-177. Aircraft A through D are provided with an additional and separate oxygen system in the main cabin for passenger use *(see figure 4-30).*

4-178. TROUBLE SHOOTING OF OXYGEN SYSTEM.

Trouble	*Probable Cause*	*Remedy*
a. No reading on pressure gage.	No oxygen in system.	Replace empty cylinder with full one.
b. Incorrect pressure gage reading.	Clogged pipes.	Check pipes for stoppage.
	Defective regulator gage.	Replace regulator.

Trouble	*Probable Cause*	*Remedy*
c. No oxygen available at outlet.	Coupling plugged.	Replace coupling.
	Regulator inoperative.	Replace regulator.
d. Oxygen pressure dropping when regulator valves are closed.	Loose couplings in pipe.	Check and tighten loose connections.
e. Oxygen has unnatural odor.	Foreign matter oxidizing in pipe. (Hot spot in pipe may indicate location.)	Blow out pipe with dry air or nitrogen; replace dirty parts or replace cylinder if necessary.
f. Oxygen leaks at cylinder valve.	Dirt in valve.	Clean or replace leaky part.
g. Leak around fitting.	Joint loose or fitting defective.	Tighten joint or replace fitting.
h. Regulator inoperative.	Punctured diaphragm.	Remove regulator.
	Outlet elbow dust plugs not removed at installation.	Take out dust plugs and reinstall regulator.

4-179. FILLING OXYGEN SYSTEM. Replenish the oxygen supply in the system by removing the empty cylinder and installing a charged cylinder (see paragraphs 4-191 and 4-192).

PASSENGER'S OXYGEN DURATION (HOURS)

OXYGEN SUPPLY:
127.4 CUBIC FEET
AT 1800 PSI

Note:
To obtain oxygen duration per passenger, divide by the number of passengers.

CYLINDER PRESSURE (PSI)	ALTITUDE (THOUSAND FEET) 10	15	20	25
1800	30.	28.3	23.	13.9
1500	24.3	22.6	16.5	11.
1200	17.9	16.8	13.9	8.1
900	12.1	11.6	9.3	5.8
600	5.8	5.8	4.7	2.9
300	DESCEND BELOW 10,000 FEET			

FIGURES INDICATE CONTINUOUS-FLOW OXYGEN DURATION

CREW MEMBERS OXYGEN DURATION (HOURS)

OXYGEN SUPPLY:
22 CUBIC FEET
AT 1800 PSI

Note:
To obtain oxygen duration per crew member, divide by the number of crew members.

CYLINDER PRESSURE (PSI)	ALTITUDE (THOUSAND FEET) 10	15	20	25
1800	5.2 .7	4.9 .8	4.0 1.0	2.4 1.3
1500	4.2 .6	3.9 .7	3.2 .8	1.9 1.0
1200	3.1 .4	2.9 .5	2.4 .6	1.4 .8
900	2.1 .3	2.0 .3	1.6 .4	1.0 .5
600	1.0 .1	1.0 .2	.8 .2	.5 .3
300	DESCEND BELOW 10,000 FEET			

BLACK FIGURES INDICATE DILUTER LEVER "NORMAL"

RED FIGURES INDICATE DILUTER LEVER "100%"

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Figure 4-29. Oxygen Consumption Table

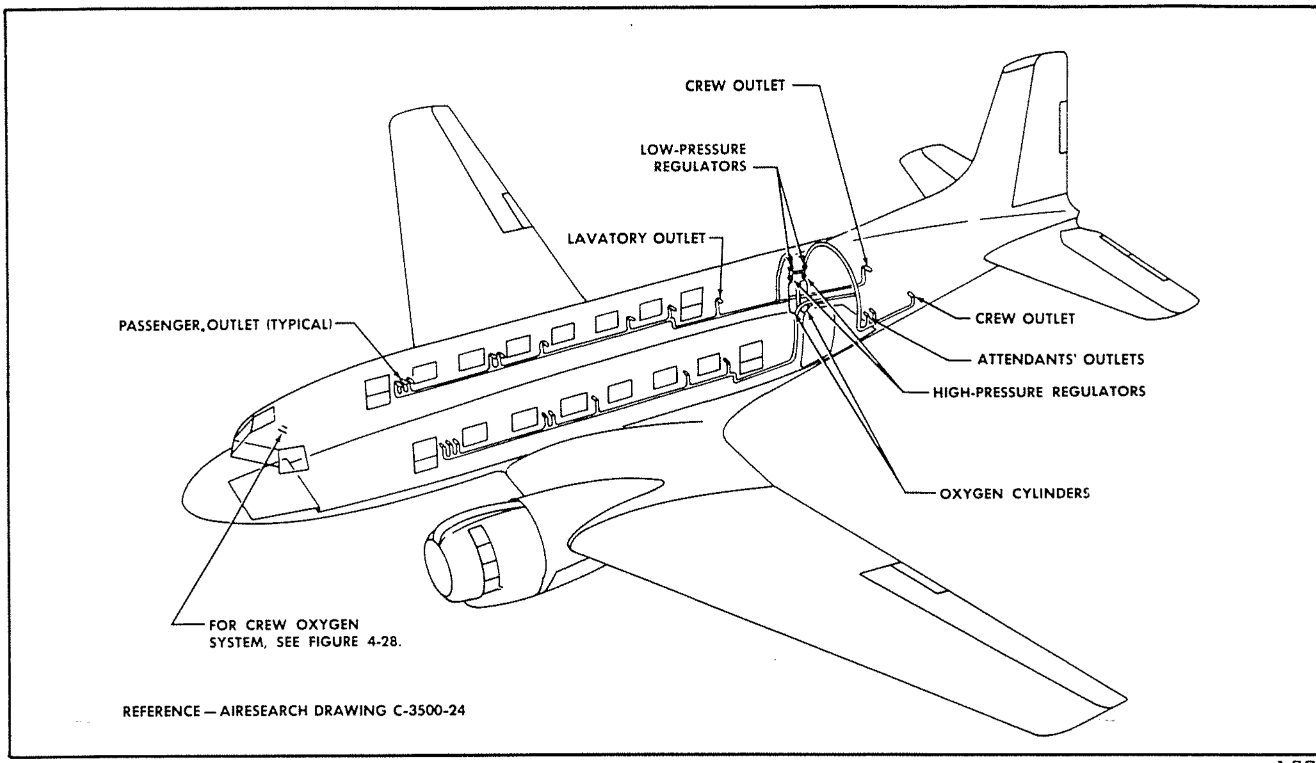

Figure 4-30. Main Cabin Oxygen System Schematic (Aircraft A through D)

4-180. BLEEDING OXYGEN SYSTEM. It is necessary to bleed the oxygen system before working on any of the system components. Observe the precautions given in the warning following, and proceed as follows:

WARNING

Keep all oxygen equipment free from oil, grease, dirt, and moisture. Do not smoke or allow combustible matter to come near high-pressure oxygen equipment.

a. Close the oxygen cylinder shutoff valve.

b. Connect a hose to either the pilot's or co-pilot's regulator, or to both.

c. Open the EMERGENCY valve to bleed the pipes. Permit the oxygen to flow overboard until the pipe is empty.

d. Disconnect the hose from the regulator or regulators to which it was attached.

e. Cap or plug any open fittings to prevent contamination of the pipes.

f. Return the EMERGENCY valve to the closed position.

4-181. OXYGEN SYSTEM TEST AND CHECK-OUT PROCEDURE.

4-182. The following equipment will be required to test the oxygen system: an oxygen recharger assembly; a test pressure gage assembly, consisting of a test pressure gage, a shutoff valve, and a zero-to-2000-psi adapter; one zero-to-100-psi test pressure gage; and a zero-to-30-psi test pressure gage assembly. The adapter must be suitable for connecting a test pressure gage between the oxygen system and the oxygen system test assembly, with the shutoff valve between the test pressure gage and the recharger assembly.

WARNING

When making tests, keep all oxygen equipment free from oil, grease, dirt and excessive moisture. Do not smoke and do not allow combustible matter to come in contact with high-pressure oxygen.

4-183. Test the system as follows:

a. Release all oxygen from the system (see paragraph 4-180).

b. Disconnect the cylinder from the oxygen system, leaving the regulator and cylinder pressure gage at-

tached to the aircraft system. Cap or plug the fitting to prevent contamination of the pipes while the cylinder is disconnected.

c. Make certain that the regulator handle is in the OFF position by turning the handle of the regulator counterclockwise and connect the test equipment to the aircraft oxygen supply system.

d. Slowly turn the handle of the test regulator and fill the system to 2000 psi as indicated by the pressure test gage. When the required pressure has been reached, turn the regulator OFF, close the test equipment shutoff valve, and disconnect the oxygen recharger assembly system.

e. Allow five minutes for temperature equalization; then tap the test pressure gage lightly and record the pressure. After 15 minutes, recheck the pressure. No pressure drop is allowed.

f. If there is a drop in pressure in the system, locate the leak, using soap or bubble solution. Remove all soap or bubble solution by washing the parts with water, and then dry them with a clean cloth.

g. Check the pressure on the aircraft cylinder pressure gages. They should read within 50 psi plus or minus of the test gage.

h. Release pressure from the system and remove the equipment.

i. Reconnect the oxygen supply system.

4-184. OXYGEN SYSTEM PIPES AND FITTINGS. Oxygen pipes and fittings consist of approximately 12 feet of soft annealed copper tubing ($^3/_{16}$ inch OD by 0.035) and 20 bronze fittings, which convey oxygen from the cylinder to the diluter-demand regulators *(see figure 4-28)*. All tubing and fittings are pressure-tested to 3000 psi, and the system (after installation) is tested to a full-cylinder pressure of 1800 psi. No torque values are specified, but care should be taken not to overtighten the fittings. When replacements are made, no deviation from specified materials is allowable. The pipes and fittings must be kept free of grease, oil, excessive moisture, and dirt to prevent contamination of the oxygen supply and/or dangerous explosions.

4-185. PERIODIC LEAKAGE TEST OF OXYGEN PIPES AND FITTINGS. All pipe fittings and diluter-demand regulator connections should be tested periodically for leakage by painting with soap solution or bubble fluid. Remove all soap or bubble solution by washing with water and dry with a clean cloth.

4-186. CLEANING OXYGEN PIPES AND FITTINGS. If oxygen from a fresh cylinder becomes contaminated because of small accumulations of grease or foreign matter in the tubes or fittings, clean the pipe as follows:

a. Bleed the oxygen system (see paragraph 4-180).

b. Disconnect and remove tubes and fittings.

c. Clean oil and grease from pipes and fittings by a vapor-degreasing method, using either a trichlorethylene vapor degreaser (Specification AN-T-37a) or a mild alkaline cleaner. Then dry the pipes and fittings by baking them in an oven at 121° to 149°C (250° to 300°F) for one hour or until dry.

d. As an alternative method, immerse and flush all parts with a mild alkaline cleaner for 10 minutes. Flush tubing with cold water, then with hot water 71° to 82°C (160° to 180°F) and then dry in an oven at 66° to 93°C (150° to 200°F).

WARNING

Do not clean oxygen pipes or fittings with carbon tetrachloride, which, in the presence of oxygen, will corrode the pipes and generate deadly chlorine gas. Also avoid the use of leaded gasoline, precipitation naphtha, or dry cleaning solvents.

4-187. REMOVAL OF OXYGEN SYSTEM PIPES AND FITTINGS. For removal of oxygen pipes and fittings, proceed as follows:

a. Bleed the oxygen system (see paragraph 4-180).

b. Disconnect and remove the pipes and fittings.

c. Clean all parts and fittings (see paragraph 4-186).

d. Cap, seal, and bag all pipes and fittings after cleaning.

4-188. INSTALLATION OF OXYGEN SYSTEM PIPES AND FITTINGS. Reverse the removal procedure, with the following additional steps, exercising extreme care in all assembly operations to prevent grease, oil, dust, excessive moisture, and dirt from entering pipes or fittings.

Note

Do not remove protective caps, bags, or seals until ready to make connections.

a. Assemble male threads bare unless seizure is encountered, in which case use Rectorseal No. 15. Do not apply this compound to the first two threads of male fittings.

Note

Do not allow Rectorseal to come in contact with interior fabrics as it is very difficult to remove.

b. Wipe excess compound from the connection after tightening.

4-189. OXYGEN CYLINDERS. One oxygen supply cylinder (NAF 1135-295) is installed at station 47 on the left side of the flight compartment just aft of the pilot's seat. A single strap holds the cylinder in formed aluminum brackets. A shutoff valve is incorporated with the cylinder. The assembly weighs 14.7 pounds when fully charged with 1800-psi pressure, and the total capacity of the cylinder is 22 cubic feet of oxygen. There is no provision for refilling the cylinder aboard the aircraft. When empty, it must be removed and replaced with a charged cylinder of the same type.

4-190. REPLACEMENT OF OXYGEN CYLINDERS. If the oxygen cylinder is damaged in any way, it must be replaced with a new or serviceable unit. No repairs can be made on oxygen cylinders.

WARNING

Release of oxygen under pressure is a dangerous fire hazard.

4-191. REMOVAL OF OXYGEN CYLINDER. To remove the oxygen cylinder, proceed as follows:

a. Turn the shutoff valve on the cylinder to OFF.

b. Bleed the system (see paragraph 4-180).

c. Disconnect the pipe at the oxygen cylinder.

WARNING

Take every precaution to prevent entrance of dirt, excessive moisture, and grease into the oxygen system. Foreign matter in the system may cause an explosion when the system is refilled.

d. Plug the end of the system pipe and cap the cylinder.

e. Loosen the nut and release the strap assembly holding the cylinder to its supports.

f. Remove the cylinder.

4-192. INSTALLATION OF OXYGEN CYLINDER. Reverse the removal procedure, being careful to note that the gages indicate a full cylinder by a reading of 1800 (± 50) psi.

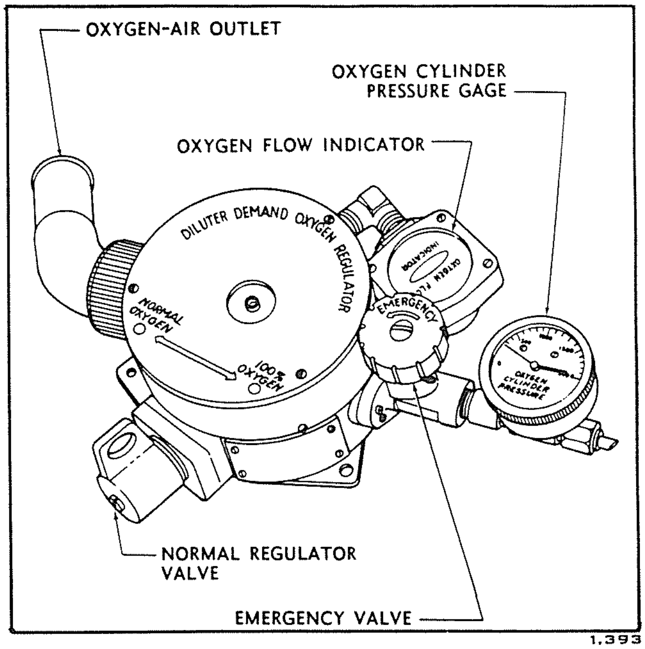

Figure 4-31. Diluter-Demand Oxygen Regulator

4-193. OXYGEN GAGES. There is no separate oxygen gage installation, but each diluter-demand oxygen regulator is equipped with an integral gage that registers pipe pressure calibrated in psi from zero to 2000. These gages are designed to take the full 1800-psi pressure of the oxygen cylinder and there is no separate gas pressure reducer in the pipe. For information on oxygen consumption, see figure 4-29.

4-194. OXYGEN REGULATORS. The three oxygen regulators are of the diluter-demand type, in which air is mixed with oxygen in the proper proportions for a given altitude and in a quantity demanded by the user *(see figure 4-31).* This regulator furnishes air only from sea level to 10,000 feet and supplies oxygen diluted with air in the correct variable ratio for the altitude attained. The regulator is controlled by an automix lever mounted on the regulator. This lever has two positions, NORMAL OXYGEN for normal diluter-demand operation and 100 PER CENT OXYGEN for pure oxygen demand operation. The diluter-demand regulator also has an emergency valve, which is indicated by a red knob. The emergency valve bypasses the demand valve, and when turned to the ON position, provides a continuous flow of 100 per cent oxygen to the mask. The emergency valve may be used when it is necessary to revive a crew member or when an oxygen mask leaks badly.

CAUTION

Use of the emergency valve when the pressure is below 50 psi should be avoided except in extreme emergencies.

4-195. REPLACEMENT OF OXYGEN REGULATORS. If an oxygen regulator is defective, replace it. This unit is a precision instrument and its maintenance must be performed only by specialized personnel.

4-196. REMOVAL OF OXYGEN REGULATOR. To remove the diluter-demand regulator, proceed as follows:

a. Bleed the oxygen from the system (see paragraph 4-180).

b. Remove the oxygen regulator.

4-197. INSTALLATION OF OXYGEN REGULATOR. Reverse the removal procedure.

WARNING

Before installing a regulator, be sure to remove the dust plugs from the outlet elbow and from all the fittings. After the regulator is installed, make certain that the emergency valve or valves are closed before opening the cylinder valve.

4-198. FLOW TEST OF OXYGEN REGULATORS. To test the oxygen regulators, proceed as follows:

a. Connect a diluter-demand mask to the regulator to be tested.

b. Turn the auto-mix lever on the diluter-demand regulator to the 100 PER CENT OXYGEN position.

c. Put on the mask and breathe oxygen in a normal manner. The flow indicator must function freely with each breath. Return the lever to NORMAL OXYGEN.

d. Turn the emergency valve full on for 10 to 20 seconds. During this time there must be continuous and unrestricted flow of oxygen.

e. Return the emergency valve to the OFF position.

4-199. MAIN CABIN OXYGEN SYSTEM (AIRCRAFT A THROUGH D). *(See figure 4-30.)* A high-pressure oxygen system for passengers is provided on aircraft A through D. Two interconnected oxygen cylinders are provided; oxygen may be used from either one of these cylinders or from both. The cylinders are installed in the coat rack on the right side of the main cabin at approximately station 490. When charged at 1800 psi, each cylinder contains 63.7 cubic feet of oxygen, a total supply of 18 man-hours at 15,000 feet, or enough to supply 18 men for one hour at that altitude. The cylinders are connected through a rigid supply pipe to 21 outlets, to which continuous flow masks can be attached. There is one outlet for each seat position, one outlet installed in the lavatory, and one outlet at each of the two crew bunks in the crew's sleeping quarters.

4-200. MAIN CABIN OXYGEN SUPPLY CYLINDERS (AIRCRAFT A THROUGH D).

4-201. REMOVAL OF MAIN CABIN OXYGEN SUPPLY CYLINDERS (AIRCRAFT A THROUGH D). To remove the oxygen supply cylinders, proceed as follows:

a. Close the shutoff cock.

b. Bleed the system.

c. Disconnect the oxygen cylinder.

d. Loosen the clamp and remove the cylinder.

4-202. INSTALLATION OF MAIN CABIN OXYGEN SUPPLY CYLINDERS (AIRCRAFT A THROUGH D). Reverse the removal procedure, making certain that the gage indicates a full-cylinder reading of 1800 (± 50) psi.

WARNING

Do not smoke or allow flame or combustible material in the vicinity while installing the oxygen cylinders. Keep all oxygen equipment free from dirt, oil, grease, and excessive moisture.

4-203. OXYGEN SYSTEM BOLT TORQUE VALUES. Unless otherwise noted, see paragraphs 4-259 and 4-260 for bolt torque values.

4-204. FIRE EXTINGUISHER AND DETECTION SYSTEMS.

4-205. DESCRIPTION. *(See figures 4-32 and 4-33.)* A mechanically controlled carbon dioxide fire extinguisher system is provided for the control of fires in the engine nacelles and in the cabin heater compartment. The system is controlled from the flight compartment by a selector valve and two discharge handles. These controls are located on the fire emergency control panel on the floor of the flight compartment, immediately aft of the control pedestal. The selector valve selects the area to which the CO_2 is to be directed; and the two discharge handles, one for each CO_2 cylinder, release the CO_2 through distribution pipes to the fire zone.

4-206. Cable systems attached to the fuel, oil, and hydraulic emergency shutoff valves, and the generator air blast tube shutoff are controlled by two actuator handles located on the fire emergency control panel.

4-207. Hand-operated CO_2 fire extinguishers supplement the mechanically operated system to provide fire control in the flight compartment, the main cabin, and in the baggage compartments.

4-208. A warning light on the left electrical panel in the flight compartment is operated by an automatic fire detection system to indicate a fire in the heater compartment. Engine section fires are indicated by warning lights on the pilot's main instrument panel.

4-209. When the aircraft is on the ground, fire control in the nacelle sections may be accomplished by a ground equipment CO_2 hose nozzle inserted through an access door provided for this purpose.

4-210. Two celluloid disc discharge indicators, one red and one yellow, are installed in fittings attached to the fuselage skin and can be seen from outside the aircraft. Both CO_2 cylinders are connected to the discharge indicators by overboard discharge pipes. If a CO_2 cylinder should discharge through a ruptured safety seal, the red celluloid indicator disc will be blown out, indicating discharge of one or both CO_2 cylinders due to thermal expansion (see paragraph 4-243, following). If one or both cylinders are discharged intentionally, the CO_2 pressure acts on a plunger that punches out the yellow indicator disc to indicate normal system discharge. The discs are easily replaceable.

4-211. For the sequence of operation for the control of fire, see the applicable handbook of flight operation.

4-212. TROUBLE SHOOTING OF FIRE EXTINGUISHER AND DETECTION SYSTEMS.

WARNING

Use extreme care in all operations involving CO_2 cylinder valves and CO_2 cylinder controls. Note all warnings included in this section.

Trouble	*Probable Cause*	*Remedy*
a. Discharge handle is pulled and no CO_2 is discharged to area selected.	CO_2 cylinder empty.	Recharge cylinders and check thermal and normal system discharge indicators.
	Cables rigged incorrectly.	Rig cables correctly (see paragraph 4-240).
	Cylinder control head fouled.	Check for foreign objects. Control head is sometimes locked for safety during rigging. Be sure to remove locking device.
b. CO_2 is discharged into wrong location.	CO_2 pipes incorrectly routed or broken.	Reroute pipes correctly and replace as required.
c. Empty CO_2 cylinders.	Excessive pressure causing discharge from cylinder valve safety outlet.	Recharge cylinder to correct pressure and weight. Replace safety seal.
	Defective safety seal installation.	Recharge cylinder and replace safety seal and washers.
	Loose safety disc plug retainer, allowing leakage at safety discharge outlet.	Tighten plug retainer to a torque of 350 inch-pounds. Recharge cylinder.
	Leakage at cylinder valve pilot check caused by foreign matter or nick on pilot check seat.	Replace pilot check or reface pilot check seat.
	CO_2 leaking past pilot check into control head and exerting sufficient pressure in pressure-controlled cylinder to discharge cylinder.	Disassemble manual control head and examine plunger, piston shaft, piston shaft bushing, and packings. Dress down nicks or burrs on plunger or shaft with crocus cloth. Replace packings or parts, or replace entire control head.
	Nicks or foreign matter on cylinder valve main check seat, creating leakage through valve outlet.	Replace main check or reface main check seat.

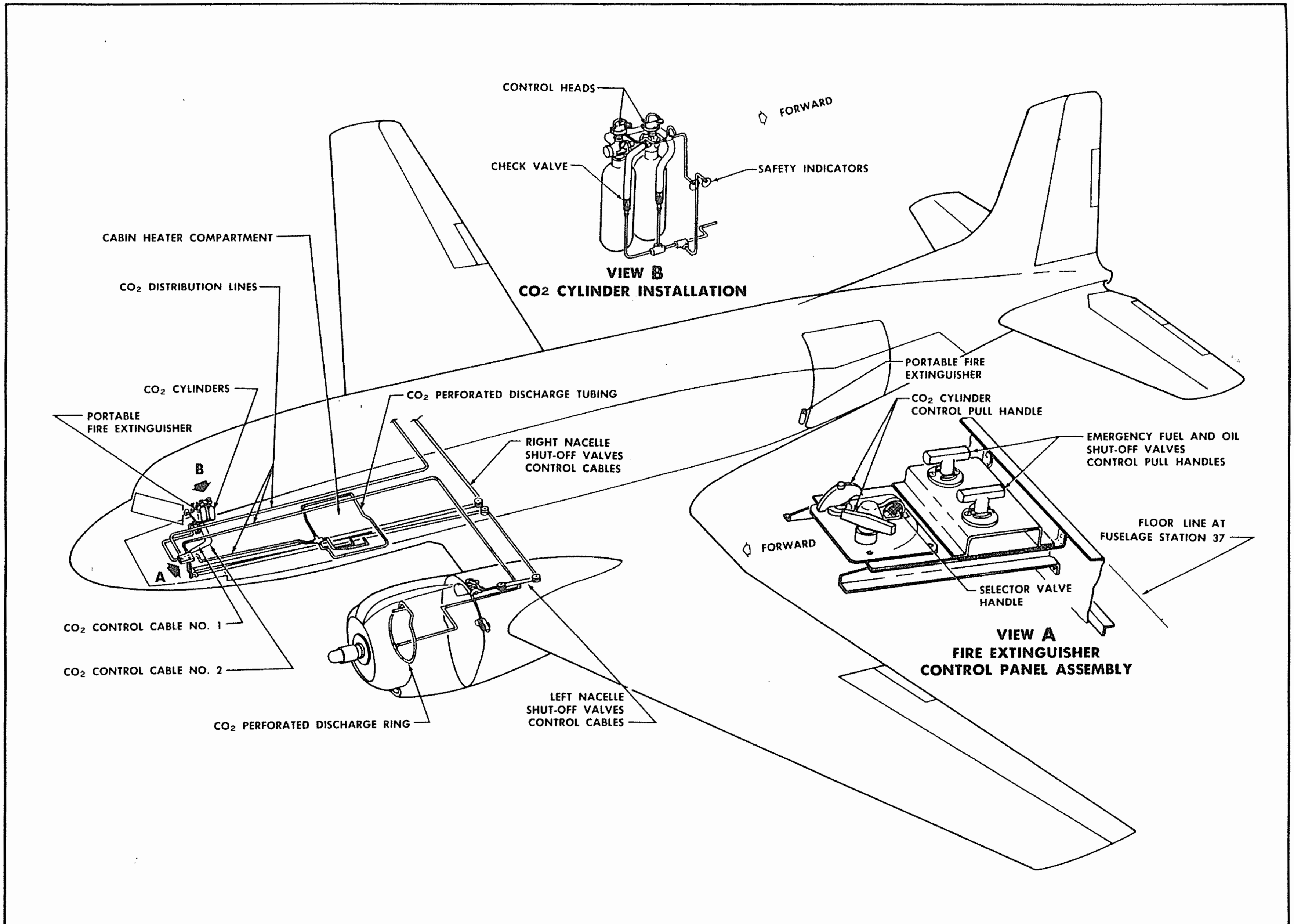

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Figure 4-32. Fire Extinguisher System

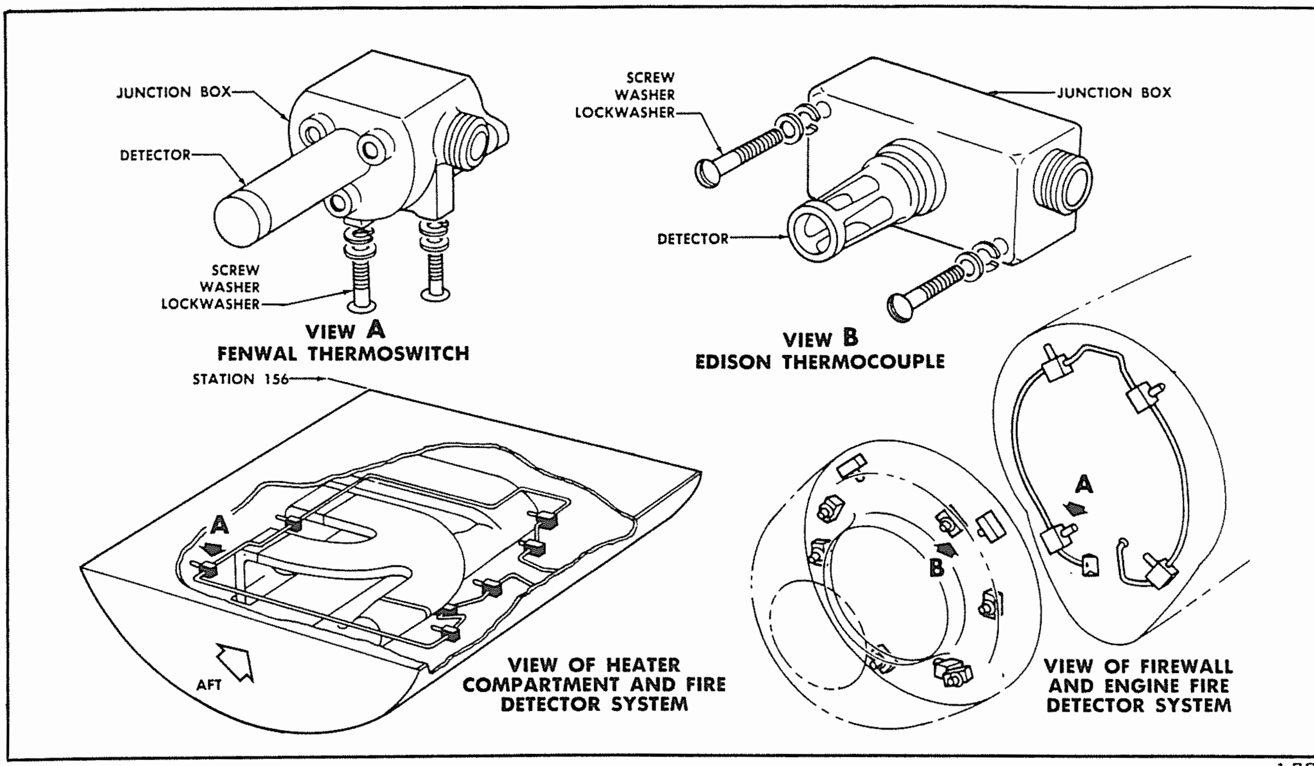


Figure 4-33. Fire Detection System

Trouble	*Probable Cause*	*Remedy*
d. Fire detector light bulbs fail to illuminate when detector test switch is operated.	Bulbs burned out.	Replace bulbs with tested bulbs.
	Tripped circuit breaker.	Check for cause of circuit breaker opening; reset circuit breaker.
	Thermocouple fire detector system sensitive relay box defective.	Plug in a tested relay box.
	High resistance between pins B-A, D-C, F-E, and H-G of Cannon plug in relay box of thermocouple-type fire detector system.	If resistance exceeds 9 (±1) ohms, check for high-resistance connections.
	Ground in circuit between pins B-A, D-C, F-E, and H-G of Cannon plug in relay box of thermocouple-type fire detector system.	Locate ground and repair circuit as required.
	Fire detector thermocouple out of order.	Check each thermocouple for continuity, polarity, and tightness of connections.

4-213. PERIODIC TEST OF FIRE EXTINGUISHER SYSTEM. *(See figure 4-32.)* A functional check of the fire extinguisher system should be made every six months. It is general practice to use the CO_2 cylinders in the system for test purposes rather than incorporate provisions for the use of a ground equipment cart.

Note

After testing make sure that the CO_2 cylinders are replaced with fully charged cylinders.

4-214. TEST OF FIRE EXTINGUISHER SYSTEM IN LEFT NACELLE AND HEATER COMPARTMENT AREA.

a. Pull the handle for the left emergency shutoff valves.

b. Turn the selector valve on the fire extinguisher control panel to the LH ENGINE position.

c. Operate the discharge control for the No. 1 cylinder. Check for a full, vigorous discharge into the left nacelle area.

d. Before discharge is complete, select the heater compartment area and check for vigorous discharge of CO_2 into that area.

e. Check the operation and rigging of the emergency shutoff valves.

4-215. TEST OF FIRE EXTINGUISHER SYSTEM IN RIGHT NACELLE.

a. Pull the handle for the right emergency shutoff valves.

b. Turn the selector valve on the fire control panel to the RH ENGINE position.

c. Operate the discharge control for the No. 2 cylinder. Check for a full, vigorous discharge into the right nacelle area.

d. Check the operation and rigging of the emergency shutoff valves.

4-216. PRESSURE AND LEAK TEST OF FIRE EXTINGUISHER SYSTEM VALVES AND PIPES.

a. Discharge one of the CO_2 cylinders into a preselected system.

b. Check all pipes and valves in the system for leaks.

4-217. TEST OF FIRE DETECTION SYSTEM. *(See figure 4-33.)* Two push-button-type fire detection system test switches are installed to test the continuity of the fire detector circuits. One button is used to test both the left and right engine section fire detectors simultaneously. It is located on the main instrument panel adjacent to the engine section fire warning lights. The other button, which is used to test the heater compartment fire detector circuit, is installed on the overhead electrical panel adjacent to the heater compartment fire warning light. When the button for either circuit is depressed, the lights for that circuit should glow. Failure of the light to illuminate indicates that its circuit is inoperative.

Note

Each warning light incorporates two bulbs so that if one burns out, the remaining bulb will illuminate. The button should be depressed for 2 to 10 seconds to allow time for the lights to illuminate.

4-218. FIRE CONTROL PANEL ASSEMBLY. *(See figure 4-34.)* The fire control panel is located under an access door in the flight compartment floor immediately aft of the control pedestal. The control assembly consists of a selector valve for directing the flow of CO_2 into the desired area; two CO_2 discharge handles, which release CO_2 gas into the area preselected by the position of the selector valve handle; and emergency fluid shutoff valve control handles for shutting off the supply of inflammable liquids aft of the firewall of either nacelle. The complete control panel assembly cannot be removed. Access to any of its component parts that may require removal, such as the selector valve and the control cable assemblies, may be gained through the snap-type access cover plate marked FIRE EXTINGUISHER-EMERGENCY SHUTOFF VALVES.

4-219. FIRE EMERGENCY SHUTOFF VALVES. *(See figure 4-35.)* Four emergency shutoff valves, one each for the fuel supply pipe, hydraulic oil supply pipe, engine oil supply pipe, and generator air blast tube, are installed in each nacelle. The shutoff valves for the fuel, hydraulic oil, and engine oil are located aft of the firewall. The generator air blast tube shutoff valve is located forward of the firewall near the duct take-off from the oil cooler air inlet duct. The fuel and hydraulic oil shutoff valves are interconnected by linkage; the engine oil shutoff valve is actuated by an Arens control connected to the fuel shutoff valve bellcrank; and the generator air blast tube shutoff valve is actuated by a cable control connected to the hydraulic oil shutoff valve bellcrank. These valves shut off the flow of inflammable liquids to the engine section and the direct blast of air in the engine accessory section. The valves are all operated by one handle located on the fire extinguisher control panel.

4-220. REMOVAL OF FIRE EMERGENCY SHUTOFF VALVES. *(See figures 4-35 and 4-36.)* To remove an emergency shutoff valve, proceed as follows:

Note

Before removing the engine fuel system emergency shutoff valve, close the center wing tanks and outer wing tank fuel selector valves and drain the strainers. Before removing the engine hydraulic system emergency shutoff valve, completely relieve system pressure and drain the supply tank. Drain the oil tank and sump before removing the engine oil system emergency shutoff valve. Provide suitable containers for collection of spillage during draining operations.

a. Remove the taper pin attaching the valve control arm to the valve shaft and remove the arm.

Note

The valve control arms for the fuel and hydraulic shutoff valves are joined by a connecting link and the two arms and the link are removed as a unit; the Arens control for the oil emergency shutoff valve, and the generator blast tube shutoff cable are also connected to these valve arms and must be disconnected before the arms are removed.

b. Disconnect the fluid supply pipe from the inlet side of the valve by unscrewing the pipe assembly nut.

c. Disconnect the fluid supply pipe from the outlet side of the valve by removing the hose and clamps.

d. Remove the stop nuts from the bolts which secure the valve to its mounting bracket.

e. Spring the fluid supply pipes aside and remove the shutoff valve from its bracket.

4-221. TEST OF FIRE EMERGENCY SHUTOFF VALVES BEFORE INSTALLATION. Before installing a valve, test it with 15-psi pressure for external and internal leakage. No external leakage is permitted. Internal leakage must not exceed 15 drops per minute for the oil or fuel hydraulic valves.

(Continued on page 327)

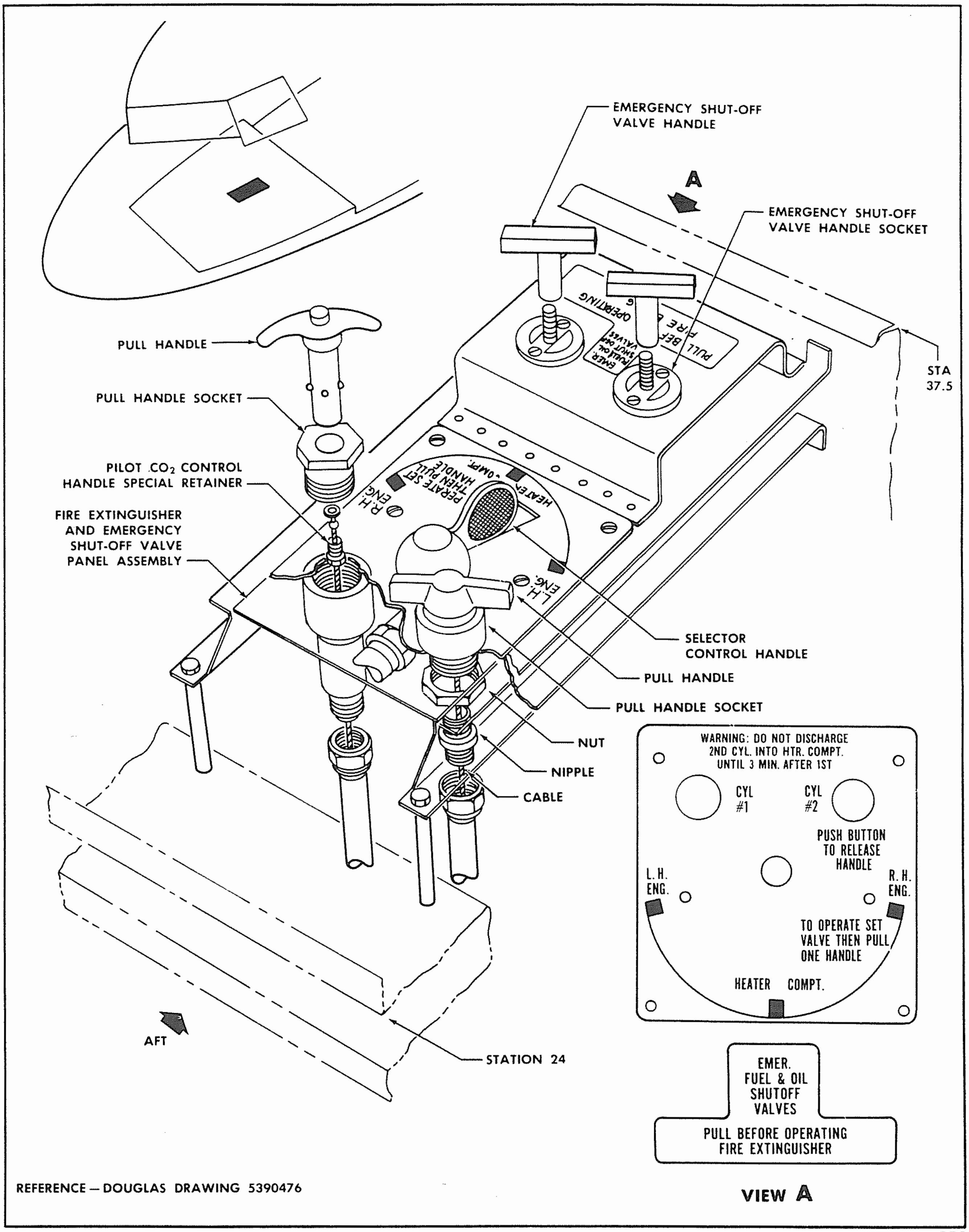


Figure 4-34. Fire Control Panel

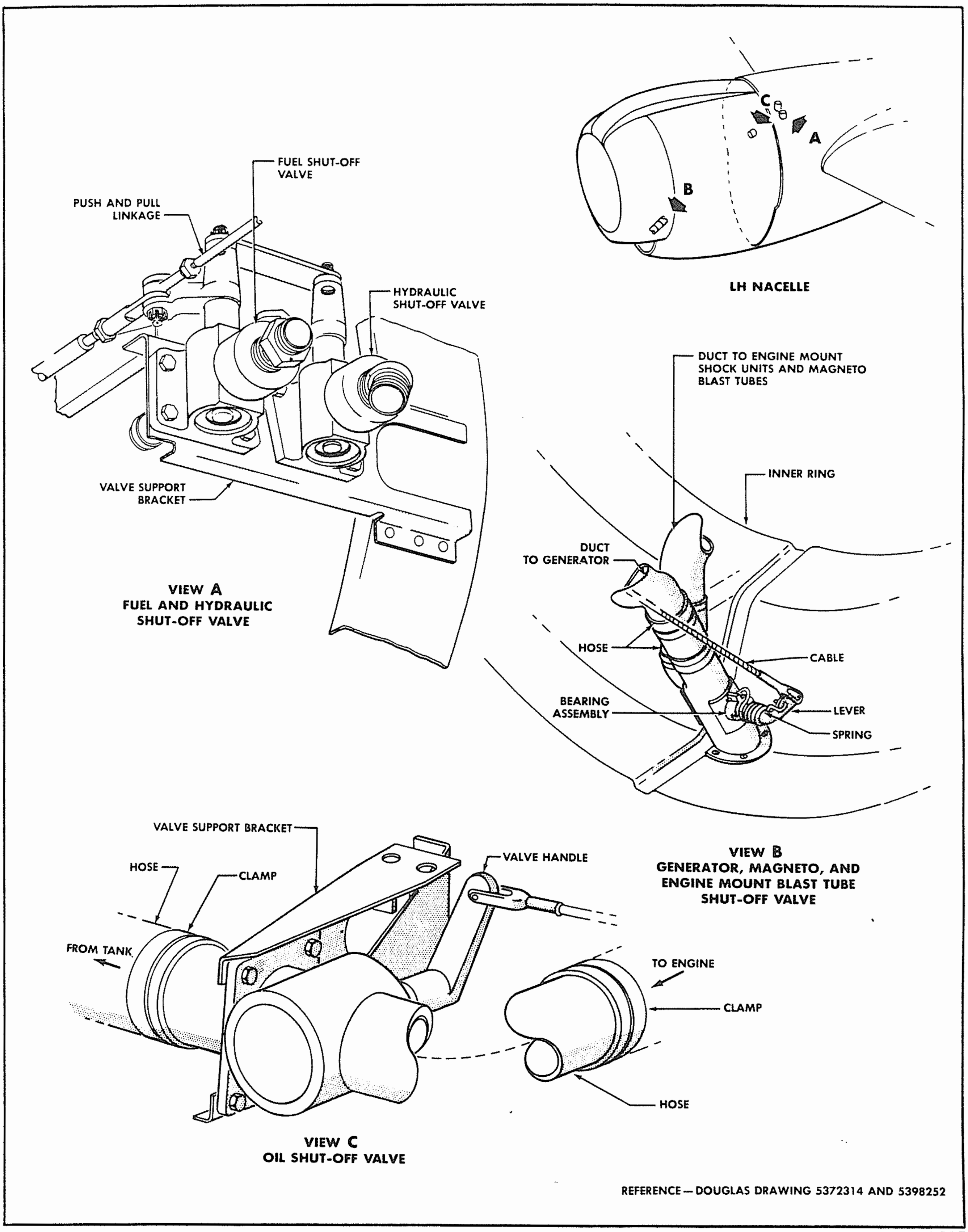
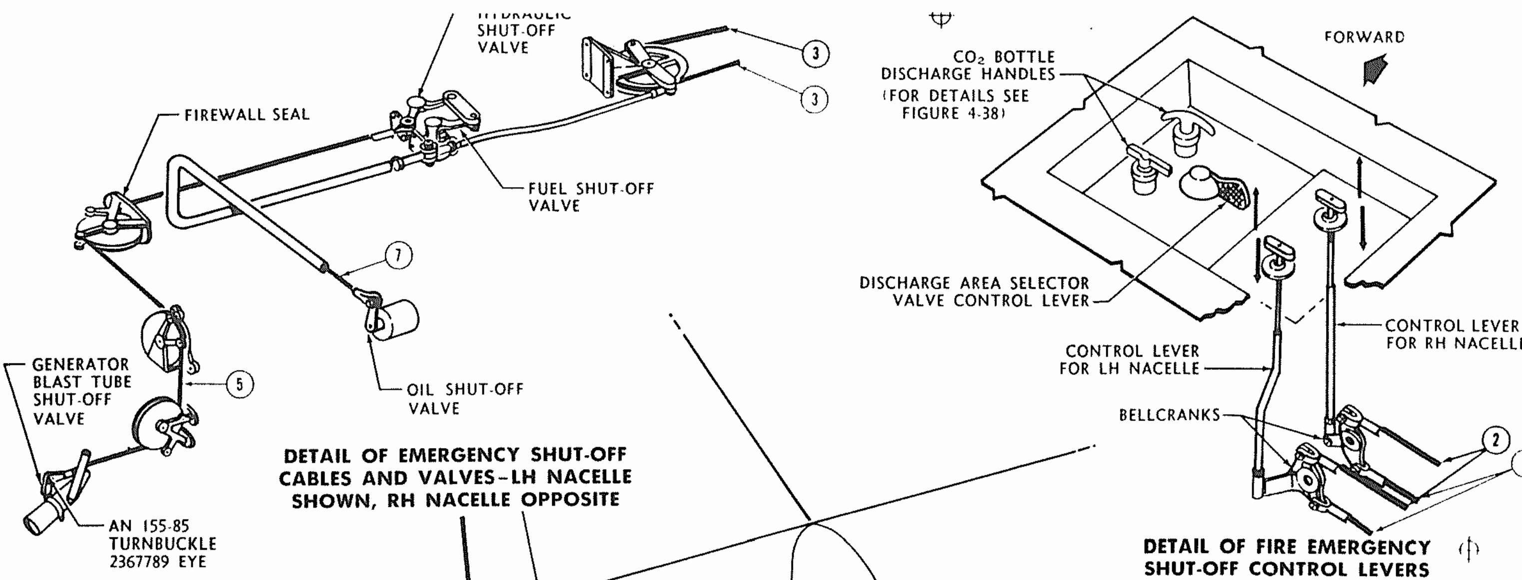
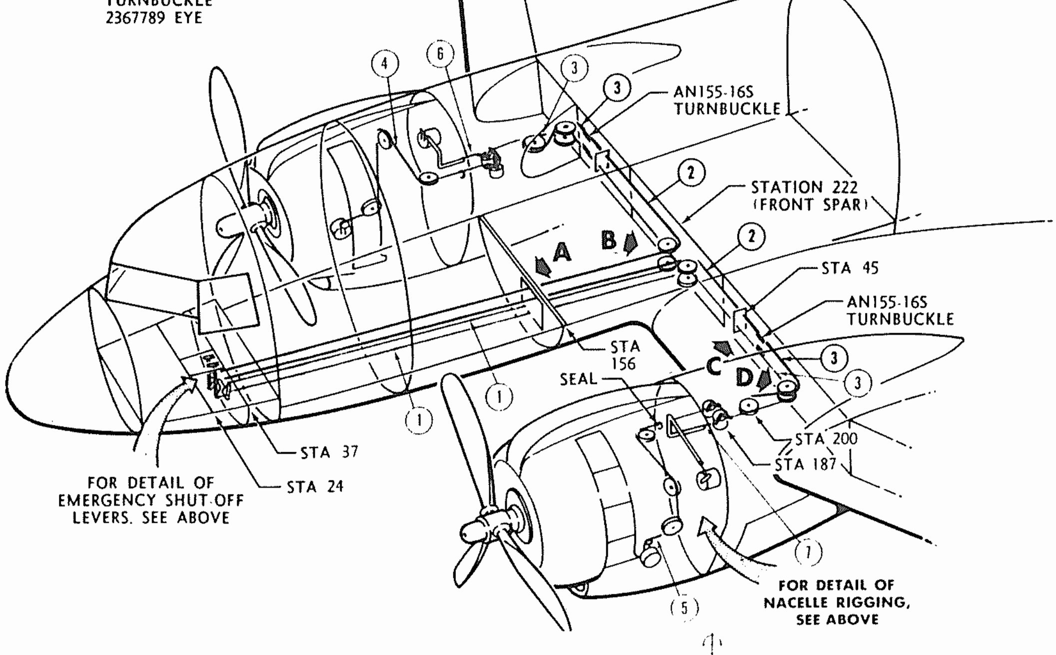


Figure 4-35. Fire Emergency Shutoff Valves





Notes:

1. Encircled numbers refer to cable listed in Cable Chart, see Sheet 4.
2. For cable assembly details, see Sheet 4.
3. For Rigging and Adjustment Diagram, see Sheet 3.
4. Tension cables in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart, Paragraph 4-260.
6. Reference—Douglas drawings 5372314, 5372513, 5372514, 5392589 and 5398252.

A — Fairlead—Station 156, see Sheet 2
B — Pulley Bracket—Front Spar, see Sheet 2
C — Fairlead—Wing Station 45, see Sheet 2
D — Pulley Brackets—Nacelle Front Spar, see Sheet 2

FLUID SHUT OFF — OPEN

FLUID SHUT OFF — CLOSED

Figure 4-36 (Sheet 1 of 4 Sheets). Fire Emergency Shutoff Control System — Key Drawing

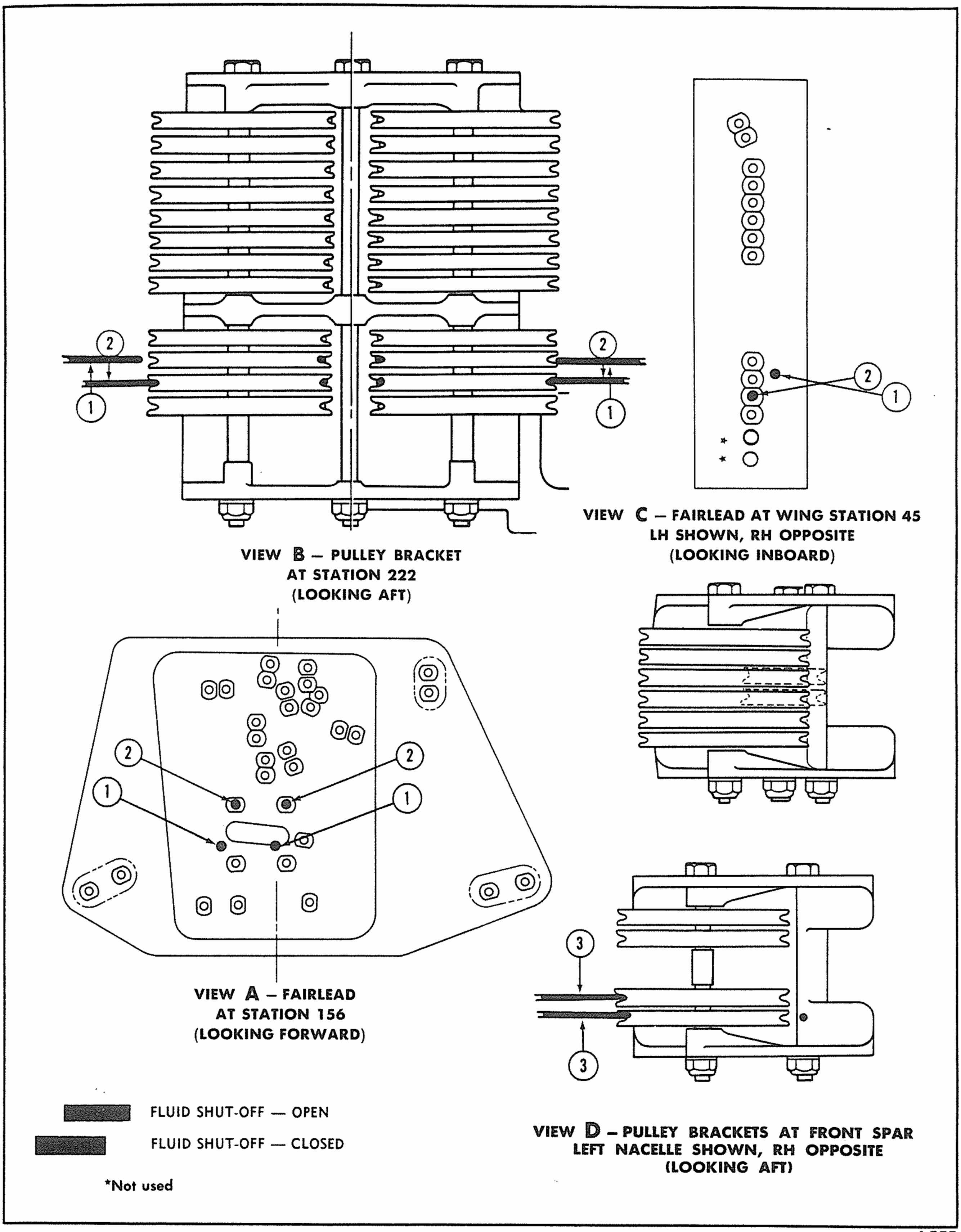
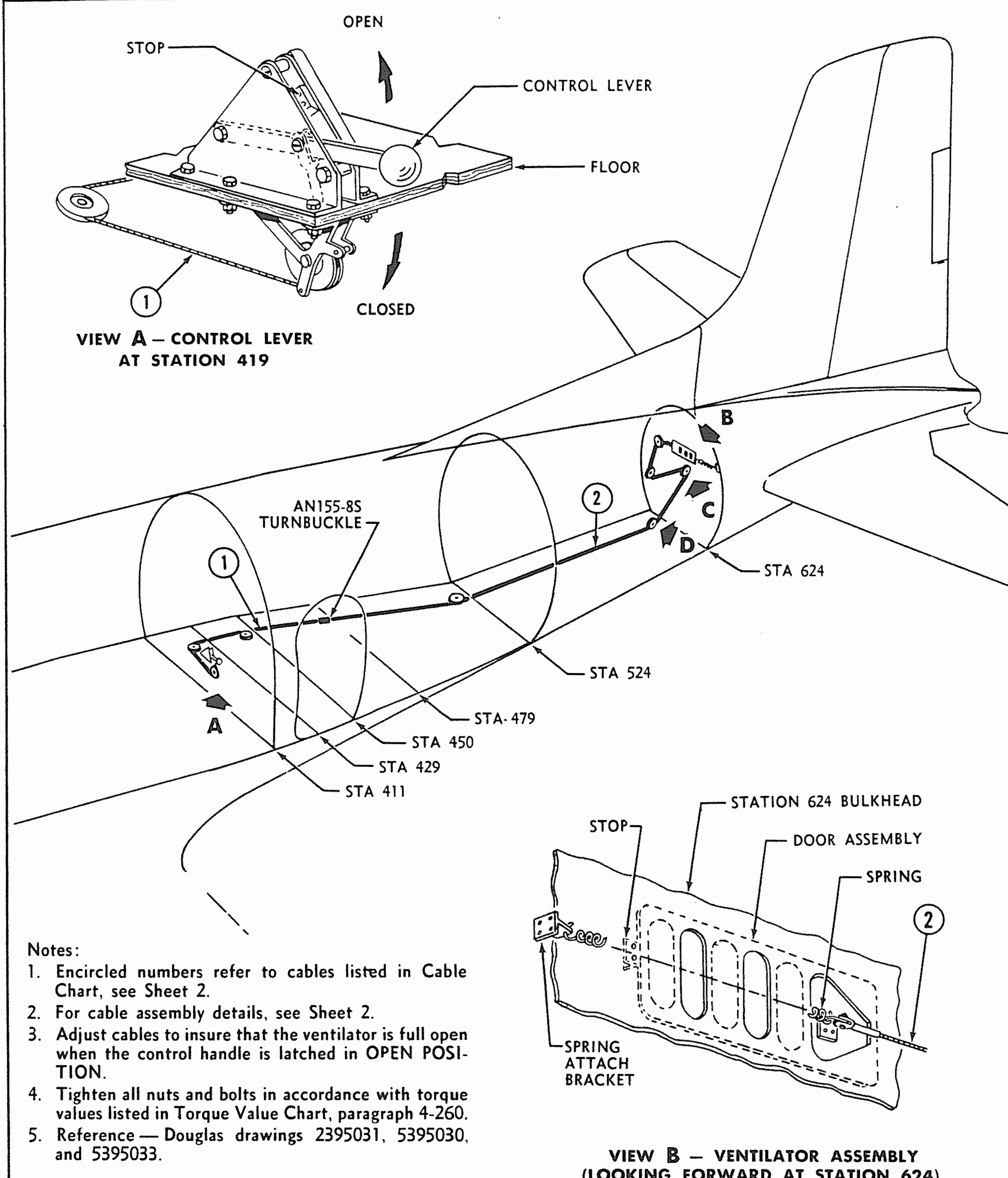


Figure 4-36 (Sheet 2 of 4 Sheets). Fire Emergency Shutoff Control System — Fairleads, Fuselage Station 156 and Wing Station 45; Pulley Brackets, Station 222 and Nacelle Front Spar



Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 2.
2. For cable assembly details, see Sheet 2.
3. Adjust cables to insure that the ventilator is full open when the control handle is latched in OPEN POSITION.
4. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart, paragraph 4-260.
5. Reference — Douglas drawings 2395031, 5395030, and 5395033.

A—Control Lever—Station 419, see Sheet 1
B—Ventilator Assembly—Station 624, see Sheet 1
C—Pulley Bracket—Station 646, see Sheet 2
D—Pulley Bracket—Station 623, see Sheet 2

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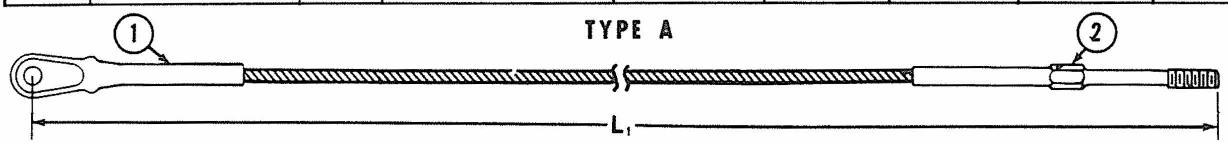
Figure 4-36 (Sheet 3 of 4 Sheets). Fire Emergency Shutoff Control System — Adjustment Diagram

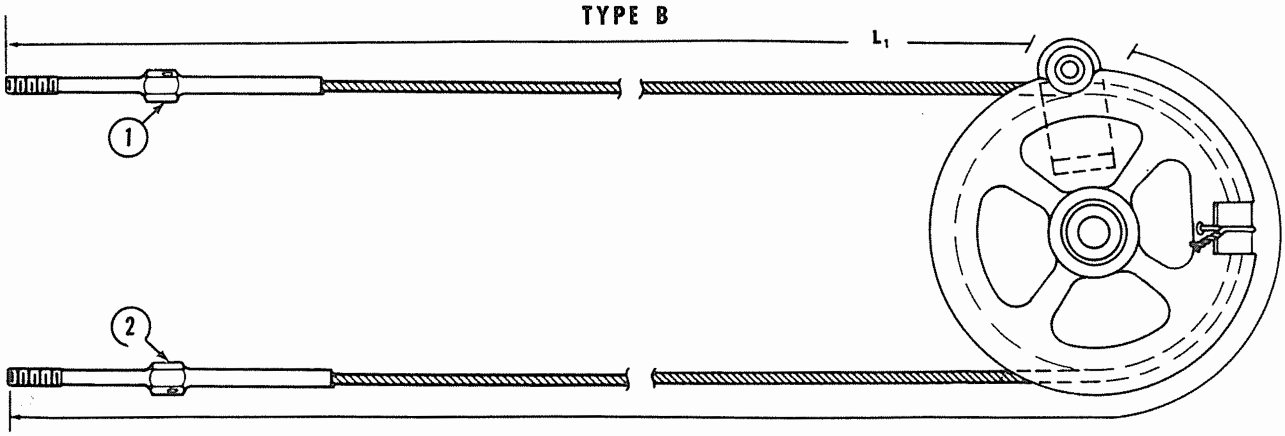
FIRE EMERGENCY SHUT-OFF CABLE CHART

Cable Ref No.	Douglas Cable Assembly Drawing No.	No. Req	Type	Cable Length (L_1)	Cable Size	Fittings 1	Fittings 2	Fittings 3	Fittings 4
1	3391412-519	2	A	221 1/2	3/32 dia 7x7 flex	AN667-3	AN669-S3 LH		
2	3391412-521	2	A	210 1/2	3/32 dia 7x7 flex	AN667-3	AN669-S3 LH		
3	4390268-3	2	B	155	3/32 dia 7x7 flex	AN669-S3 RH	AN669-S3 RH		
4	3391412-505	1	C	93	1/16 dia 7x7 flex	AN667-2	AN669-S2 LH		
5	3391412-503	1	C	77	1/16 dia 7x7 flex	AN667-2	AN669-S2 LH		
6	2188830-4	1	D	(L_1) 38 1/2 (L_2) 43 (L_3) 45 5/8	3/64 dia Arens No. FC125	1188338	AN3164R	AN3164R	REB3N
7	2188830-2	1	D	(L_1) 31 (L_2) 35 1/2 (L_3) 38 1/2	3/64 dia Arens No. FC125	1188338	AN3164R	AN3164R	REB3N

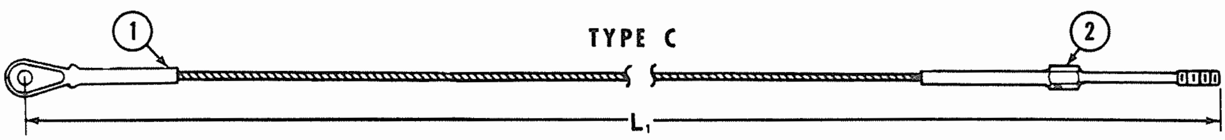
TYPE A



TYPE B



TYPE C



TYPE D

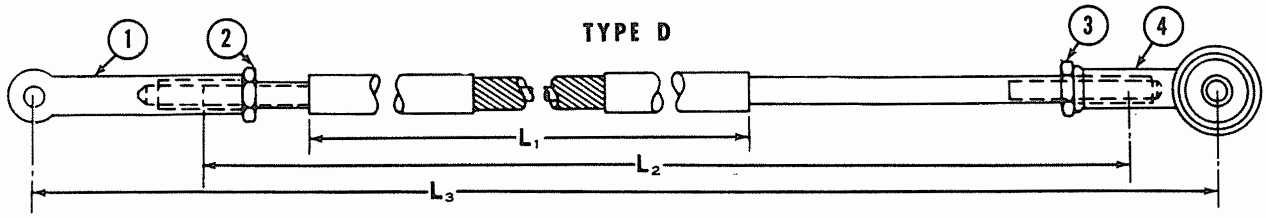


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Figure 4-36 (Sheet 4 of 4 Sheets). Fire Emergency Shutoff Control System — Cable Chart and Cable Assemblies

(Continued from page 320)

4-222. INSTALLATION OF FIRE EMERGENCY SHUTOFF VALVES. Reverse the removal procedure.

4-223. FIRE EMERGENCY SHUTOFF CONTROLS. *(See figure 4-36.)* The fire emergency shutoff control handles are located on the cockpit floor just aft of the control pedestal. Movement of each handle actuates a two-way cable system. The cable systems extend aft, from the fire control panel, through the fuselage and outboard to the fuel, hydraulic, oil, and generator blast tube shutoff valves, located in the nacelles. The shutoff valves, in each nacelle, operate simultaneously upon actuation of the respective control handle.

4-224. FIRE EMERGENCY SHUTOFF CONTROL CABLES. *(See figure 4-36.)* The fire emergency shutoff control cables extend from the fire control panel assembly, located in the flight compartment floor just aft of the control pedestal, aft through the lower part of the fuselage and outboard through the wings to the fuel, hydraulic, oil, and generator blast tube shutoff valves in the nacelles.

4-225. REMOVAL OF FIRE EMERGENCY SHUTOFF CONTROL CABLES.

a. Remove the necessary access doors on the bottom of the fuselage and center wing sections. Remove the necessary floor panels in the aircraft, between the control pedestal and the wing front spar, to gain access to the control cables.

b. Disconnect the fire emergency shutoff cables at the turnbuckles outboard of the fairleads at wing stations 45. Thread the cable ends for removal.

c. Remove the necessary guard pins, fairlead grommets, and firewall fire seals.

d. Disconnect the outboard cables from the drums at nacelle stations 200 and draw the cables into the nacelles.

e. Disconnect the generator blast tube shutoff valve cables at the turnbuckle ends. Thread the cable ends.

f. Disconnect the cables at the hydraulic shutoff valves, and pull them aft into the nacelles.

g. Disconnect the Arens flexible cables at the oil and fuel shutoff valves and draw them aft through the conduits into the nacelles.

h. Disconnect the forward fire emergency shutoff cables from the control level bellcranks at station 37.

i. Draw the cables through the floor panel aft of the control pedestal. Pull the cables slowly to prevent chipping of the micarta pulleys. Use two men to perform this operation, one to pull the cables and the other to guide the cable ends through the pulley brackets. and fairleads.

4-226. MINOR REPAIR AND REPLACEMENT OF FIRE EMERGENCY SHUTOFF CONTROL CABLES.

a. Inspect all pulleys, fairleads, and fire seals for general condition. Replace any badly worn fairleads, fire seals, or damaged pulleys.

b. Check all cables and replace any cable that shows more than six broken wires in any one-inch length (see paragraph 2-51).

c. Examine the Arens flexible controls in the nacelles for general condition and replace if found faulty.

4-227. INSTALLATION OF FIRE EMERGENCY SHUTOFF CONTROL CABLES.

a. Connect the forward fire emergency shutoff control cables to the control lever bellcranks at station 37. Route the cables aft through the fuselage and outboard through the wings, installing guard pins and fairlead grommets as necessary.

b. Route the Arens flexible cables forward through the nacelle conduits and connect the ends to the fuel and oil shutoff valves in the nacelles.

c. Attach the generator blast tube shutoff valve cables to the hydraulic shutoff valves and route the cables through the nacelles, replacing guard pins and fire seals as necessary. Align the soft steel grommets of the fire seals axially and concentrically with the cables. Tighten the retaining nuts and bolts.

d. Connect the cables to the generator blast tube shutoff valves with turnbuckles.

e. Connect the outboard fire emergency shutoff valve control cables to the drums at nacelle stations 200 and route them inboard through the wings.

f. Connect the cables with turnbuckles outboard of wing stations 45.

g. Tension all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

h. Safety all turnbuckles.

i. Replace the access doors and floor panels.

4-228. ADJUSTMENTS AND OPERATIONAL TESTS OF FIRE EMERGENCY SHUTOFF CONTROLS. For a complete description of the procedure to be followed in adjusting and testing the fire emergency shutoff controls, see figure 4-36.

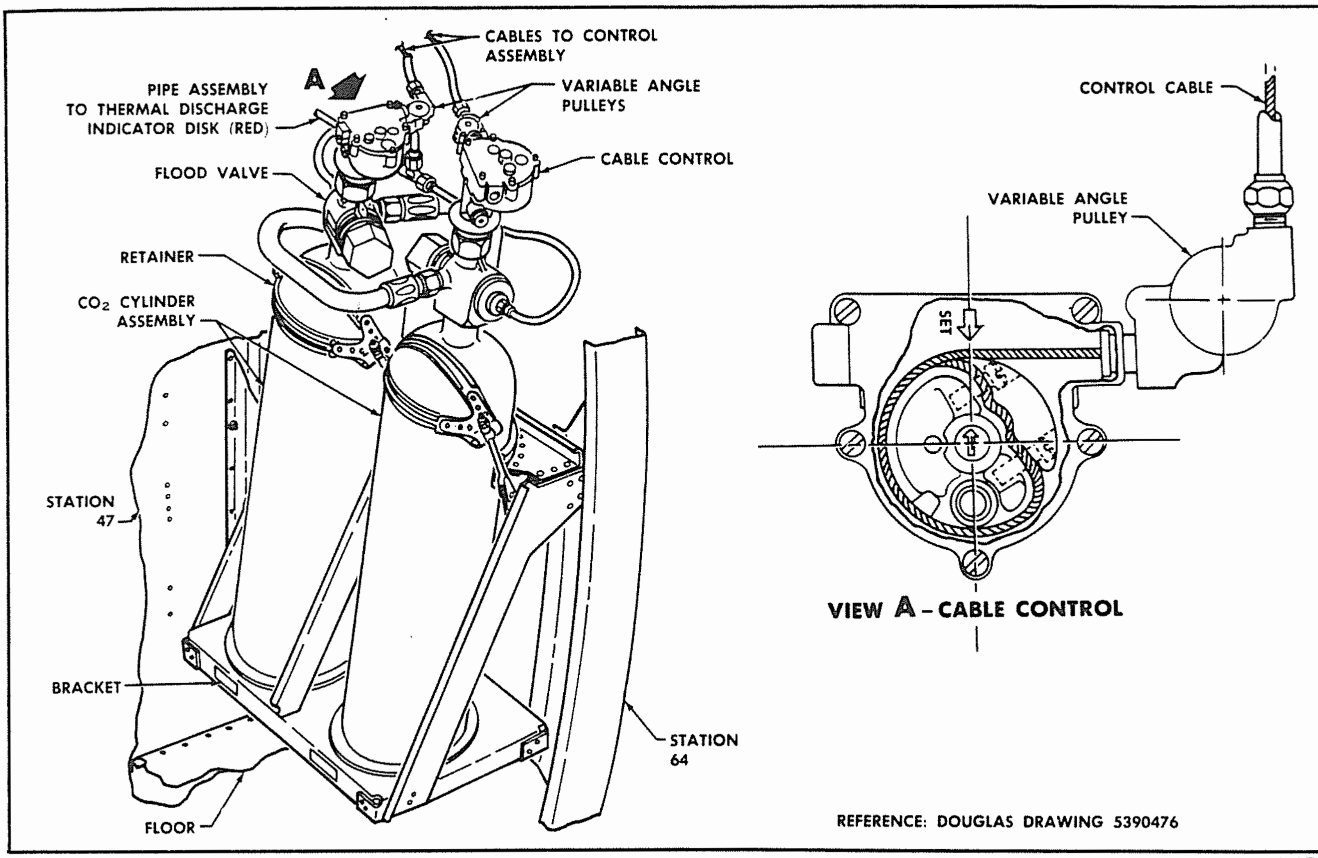


Figure 4-37. CO_2 Cylinder, Flood Valves, and Control Head Installation

4-229. CO_2 CYLINDERS AND FLOOD VALVE ASSEMBLIES. *(See figure 4-37.)* Two non-winterized regular carbon dioxide cylinders equipped with flood valves are installed on the right side of the fuselage, between stations 47 and 64. They are mounted vertically on a supporting platform and brackets, which are attached permanently to the fuselage frames and longerons. Padded clamps with adjustable hold-down bolts hold the cylinders tightly in place. Each cylinder and flood valve assembly weighs approximately 30.8 pounds when fully charged with 12.6 pounds of CO_2 gas. The cylinders should be removed and checked periodically for the correct weight, which is stenciled on the side of each cylinder.

4-230. A ¾-inch flood valve is installed on top of each cylinder to hold the pressure of the CO_2 charge. The two valves are connected by piping through a tee into the main fire extinguisher pipe. Each valve is operated by a series control head (see paragraph 4-234). The cylinder flood valves are each provided with a safety seal, which will rupture if the cylinder is overcharged or if the pressure is increased as a result of thermal expansion. If the safety seal ruptures, the cylinder will discharge outside the aircraft through an overboard discharge pipe. The valve is designed so that the safety seal is easily replaceable.

4-231. REMOVAL OF CO_2 CYLINDER AND FLOOD VALVE ASSEMBLY. To remove a CO_2 cylinder and flood valve assembly, proceed as follows:

WARNING

Be sure to follow exact instructions for removal of charged CO_2 cylinders. A charged CO_2 cylinder can become a very dangerous missile if gas is accidentally released while the cylinder is not clamped in place. Before removing a cylinder, always remove the control head from the flood valve first. When installing a cylinder, install the control head last. When disconnecting the flexible hose, disconnect it at the flood valve.

a. Disconnect the control head from the cylinder flood valve and support the control head, together with its cable, free of the cylinder.

b. Disconnect the discharge hose from the cylinder flood valve outlet.

c. Loosen the clamps and remove the cylinder.

4-232. CHARGING CO_2 CYLINDERS. The correct adapters and blow-off valves for the 3/4-inch cylinder flood valves must be included in the charging assembly.

Note

Use only dry CO_2. Any moisture will freeze in the tubes and valves, possibly rendering the extinguisher system inoperative.

a. Connect the CO_2 supply to the outlet of the cylinder valves.

b. Place the cylinder on a scale and note its exact weight.

c. Open the CO_2 supply valve and charge the cylinder with 12.6 pounds of CO_2. Perform this operation carefully, so as not to overcharge the cylinder. If the cylinder should be overcharged, reduce the pressure by using a blow-off head, which can be screwed to the top (control head outlet) of the cylinder valve. When blowing down to the desired weight, make certain that the weight of the blow-off head (approximately seven ounces) is considered. When the cylinder charge is reduced to the proper weight, remove the blow-off head.

d. With the CO_2 charge at the proper weight, shut off the recharging pump, close the CO_2 supply valve, loosen the supply pipe at the cylinder valve, bleed the pipe, and disconnect it. The cylinder valve pilot check is closed during charging.

e. Test the cylinder valve and safety seal retainer for leakage by placing the cylinder in a horizontal position so that the discharge outlet is turned up. Then test the main valve for leakage by pouring water into the outlet, filling it almost to the top. Make certain that no water spills over the top of the outlet, as it may enter the valve passages and prevent operation of the valve. Appearance of bubbles indicates leakage at the main check. Place the valve cover *only* (including the safety seal retainer) into a small cup of water. Bubbles indicate leakage at the safety seal. Tilt the cylinder so that the hole at the side of the control head connection well (which leads to the valve piston head), is higher than the hole in the center of the well (pilot check stem hole). With the valve held in this position, pour enough water into the control head connection well to cover the center hole, but do not permit it to run into the side hole. Water trapped in the small passage leading from the side hole will freeze and possibly prevent operation of the valve. Bubbles in the water covering the pilot check stem hole indicate leakage at the pilot check.

f. After completing a water test of the valves, wipe all parts perfectly dry. If any places cannot be reached with the drying cloth, blow them out with compressed air.

4-233. INSTALLATION OF CO_2 CYLINDER AND FLOOD VALVE ASSEMBLY. Reverse the removal procedure, with the following additional steps:

WARNING

When installing a cylinder, always install the control head last. A charged CO_2 cylinder can become a very dangerous missile if gas is accidentally released while the cylinder is not clamped in place.

a. Set the holding clamps only hand-tight until they are aligned by the completion of all tubing connections.

b. Before installing the series cable gas control head, be sure that the shaft-head arrow is aligned with the arrow in the cover.

c. Before tightening the series cable gas control head on the cylinder flood valve, check to see that the arrow setting has not been changed.

CAUTION

Check the operation of the control head before connecting it to the cylinder valve.

4-234. SERIES CABLE CYLINDER CONTROL HEAD. *(See figures 4-37 and 4-38.)* The cable-operated control heads are mounted on top of each cylinder flood valve. The control heads are operated manually by the two handles at the forward end of the fire extinguisher control panel. When either of these handles is pulled, a cable-operated sheave in the top of the control head rotates, forcing a plunger down and releasing pressure that opens the cylinder flood valve.

4-235. REMOVAL OF SERIES CABLE CYLINDER CONTROL HEAD.

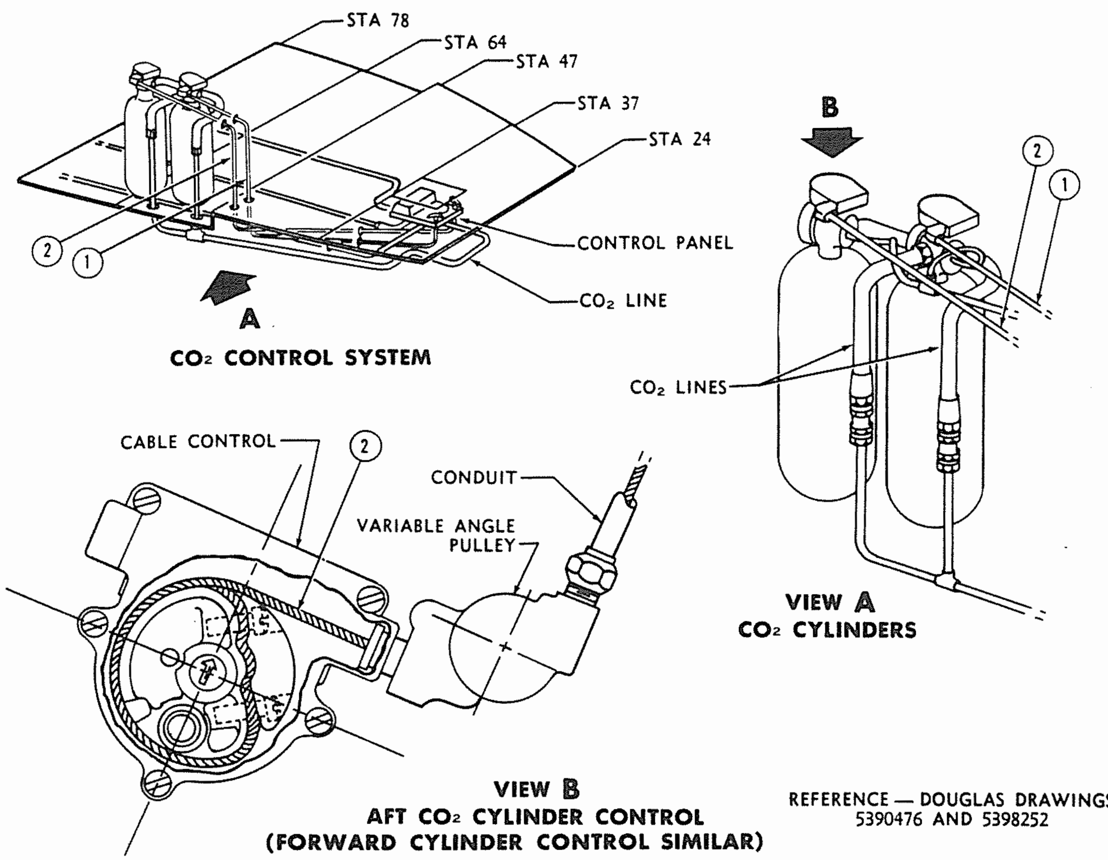
a. Remove the control head from the cylinder.

WARNING

A charged CO_2 cylinder can become a very dangerous missile if gas is released while the cylinder is not clamped in place. Always be sure to remove the control head with the cylinder clamped in place. Do not attempt to remove the cylinder in order to remove the control head at a work bench.

b. Remove the cover screws and take off the cover.

c. Raise the sheave far enough to take off the cable clamp by removing the two attaching screws; remove the cable from the sheave.



REFERENCE — DOUGLAS DRAWINGS 5390476 AND 5398252

CO_2 CONTROL HEAD CABLE CONTROL CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEM. DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH L_1	CABLE SIZE	FITTINGS 1	2	3	4	5
1	4369360-5	1	A	85	1/16 dia. 7x7 flex.	RA2487-2	AN818-4	AN819-4		
2	908810 (Kidde)	1	B	180	1/16 dia. 7x7 flex	7870	AN816-4	AN818-4	AN819-4	

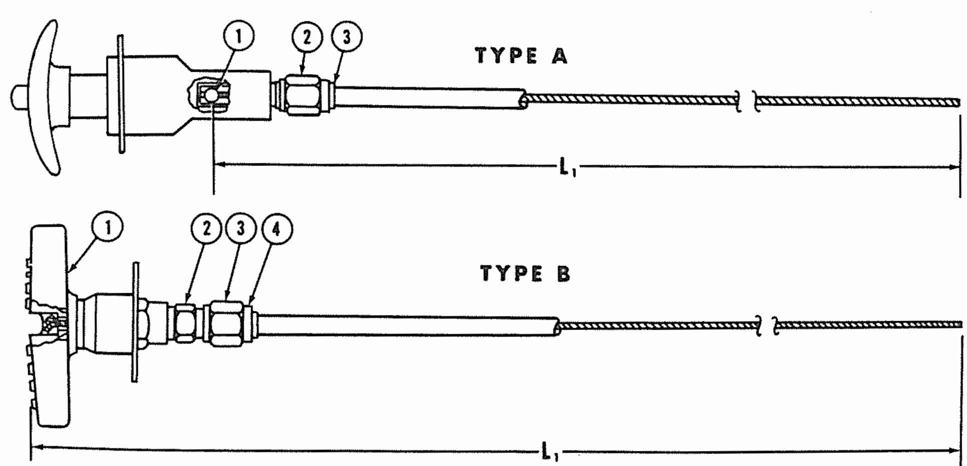


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Figure 4-38. CO_2 Control Head Control System

CAUTION

Perform this operation carefully. Do not move or rotate the control head.

4-236. FUNCTIONAL TEST OF SERIES CABLE CYLINDER CONTROL HEAD BEFORE INSTALLATION.

CAUTION

Make certain that the control head has been disassembled from the cylinder before operating it. If this is not done, loss of gas and probable injury to personnel may result. Be sure to line up the arrows on the shaft so that the control head will be in SET position before it is installed on the cylinder.

With the control head disassembled from the cylinder, check the operation of the series cable control head. Line up the arrows on the shaft and cover so that the control head will be in the SET position; then pull the control handle. The control head plunger should advance. *Reset the control head.*

4-237. INSTALLATION OF SERIES CABLE CYLINDER CONTROL HEAD.

a. Check to be sure that the arrow on the sheave shaft is aligned with the arrow on the control cover (indicating that the head is in the SET position). If it is not aligned, insert a pin through the hole in the shaft and turn the sheave end in the direction indicated on the cover until the arrows line up (a click of the sheave stop will be heard at the correct position). *Be extremely careful not to discharge the cylinder.*

b. Attach the control head to the cylinder by screwing on the swivel nut hand-tight. Do not tighten the swivel nut until the control head is correctly aligned.

4-238. CYLINDER CONTROL HEAD CABLES. *(See figure 4-38.)* The control heads are cable-operated. The cables from the control head sheaves connect to the two handles at the forward end of the fire extinguisher control panel.

4-239. REMOVAL OF CYLINDER CONTROL HEAD CABLE.

a. Carefully detach the cable from the control head sheaves. Thread the cable end for removal.

b. Disassemble the pull handle and pull the cable forward through the tubing.

4-240. INSTALLATION AND RIGGING OF CYLINDER CONTROL HEAD CABLE.

a. Insert the cable into the tubing through the control panel opening.

b. Assemble the pull handle.

c. Place the handle in full down position, making sure the arrows on the control head line up, and install the cable on the control head sheaves.

4-241. FIRE EXTINGUISHER SYSTEM DISTRIBUTION PIPES. *(See figure 4-32.)* The CO_2 gas is discharged inside the nacelles through perforated steel pipes that circle the engine accessories section and lead to the carburetor airscoop. Additional perforated steel pipes encircle the cabin heater compartment and lead to the ventilating air inlet duct and the combustion air intake port. All distribution pipes should be carefully inspected periodically to make certain that neither flow nor discharge openings have been clogged, impaired by minor damage, or distorted.

4-242. FIRE EXTINGUISHER CHECK VALVE. *(See figure 4-32.)* A check valve is installed in both supply pipes immediately adjacent to the flood valves. The locations of these check valves prevent possible waste of CO_2 into an already discharged cylinder.

4-243. SAFETY DISCHARGE INDICATORS AND PIPES. *(See figure 4-39.)* An overboard discharge pipe connects both cylinders with a fitting in the aircraft skin. Two celluloid disc indicators, one red and one yellow, are installed in fittings attached to the fuselage skin and can be seen from outside the aircraft. If a CO_2 cylinder should discharge through a ruptured safety seal, the red celluloid indicator disc will be blown out, indicating that one or both CO_2 cylinders have been discharged by thermal expansion. If one or both cylinders are intentionally discharged, the yellow celluloid indicator disc will be punched out by a plunger which is actuated by the CO_2 pressure, indicating normal system discharge.

4-244. REPLACEMENT OF YELLOW INDICATOR DISCS. The yellow indicator discs are held in position by a snap ring. When a disc has been blown out, the snap ring generally remains in the fitting. It has to be removed before a new disc can be installed and a new snap ring must be placed over the new disc.

4-245. REPLACEMENT OF RED INDICATOR DISCS. The spring in the red piston-type indicator discs *(see figure 4-39)* sometimes is distorted during discharge to such an extent that it cannot be subsequently used again. When this occurs, the entire indicator unit should be replaced.

4-246. PORTABLE FIRE EXTINGUISHERS. *(See figure 4-32.)* Two portable fire extinguishers are installed on brackets equipped with quick-release type clamps at advantageous points in the aircraft. Instructions for their use are stenciled on each cylinder. In most installations, one extinguisher is located on the floor under the pilot's seat, and the other is located just forward of the main cabin door next to the floor.

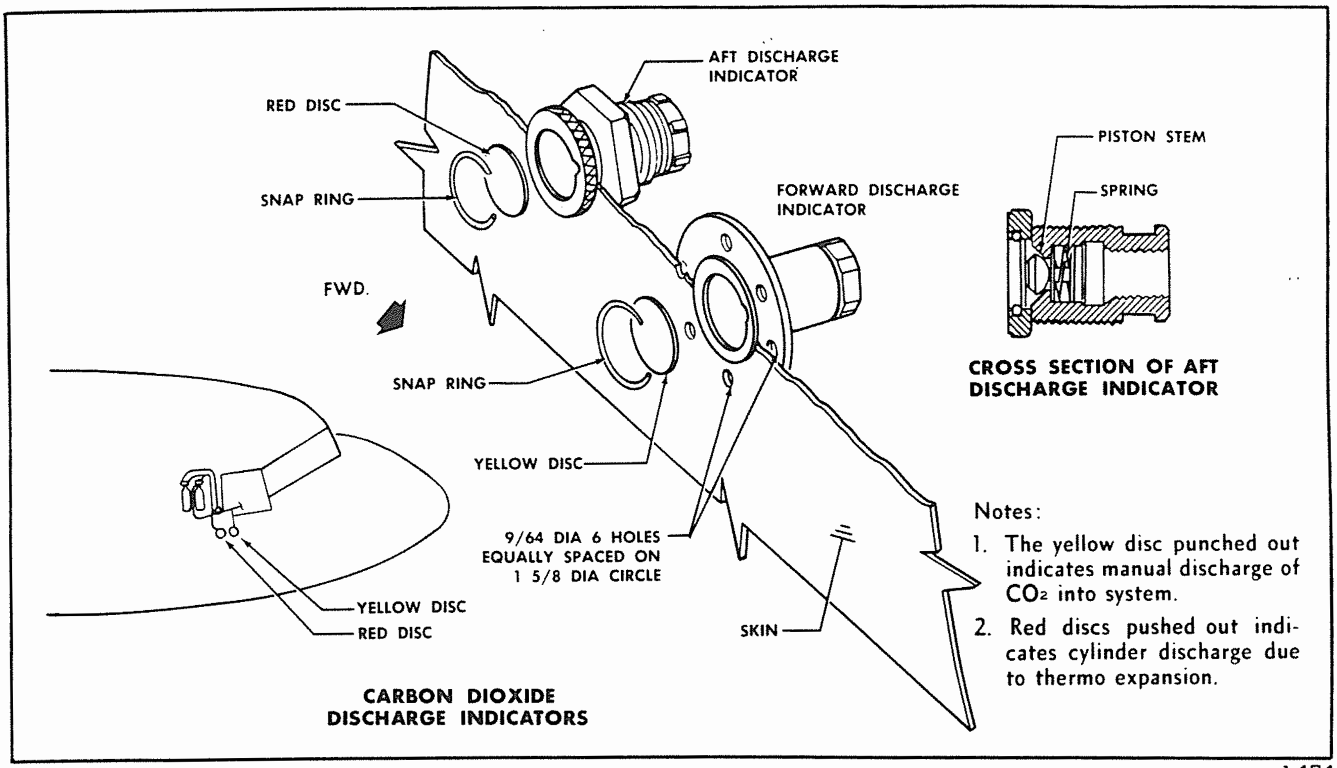

Figure 4-39. Discharge Indicators

4-247. CHARGING PORTABLE CO_2 FIRE EXTINGUISHER.

a. Release the extinguisher and remove it from the aircraft.

b. Remove the discharge horn only and connect the charging adapter.

c. Charge the extinguisher with two pounds of carbon dioxide. The gas will open the valve automatically. Release of the gas pressure in the charging pipe will automatically close the valve.

d. Test for leakage by submerging the tube outlet under water for a few minutes.

e. Remove the recharging adapter and connect the horn.

f. Attach a seal wire through the hole in the valve body and the hole in the trigger.

g. Replace the extinguisher in its bracket in the aircraft.

4-248. OPERATIONAL TEST OF PORTABLE FIRE EXTINGUISHER. Portable fire extinguishers should be given an operational test periodically. Instructions for their use are stenciled on the cylinders.

4-249. ENGINE SECTION FIRE DETECTORS. *(See figure 4-33.)* Ten Edison thermocouple-type fire detectors are mounted in the engine accessory section; six of these fire detectors are mounted in such a way that their elements protrude through the engine diaphragm. Four additional detectors are mounted on the forward side of the firewall. With a rapid rise in temperature, any one of the thermocouples can deliver an impulse that actuates the warning lights. Dual warning lights are installed for each detector circuit as a precaution against the possibility of a burned-out bulb causing the failure of a warning circuit. The lighting of either or both bulbs in any circuit indicates dangerous temperatures in the area served by that circuit.

4-250. CABIN HEATER COMPARTMENT FIRE DETECTORS. *(See figure 4-33.)* Five Fenwal thermal switch-type fire detectors are installed in the cabin heater compartment, one in the heater combustion air inlet duct and one in the wye of the ventilating air duct. Any one of the thermal switch-type fire detectors may deliver an impulse that will actuate the dual warning lights on the left overhead electrical panel. Dual warning lights are installed as a precaution against the possibility of a burned-out bulb causing failure of a warning circuit. The lighting of either or both of the bulbs indicates dangerous temperatures in the heater compartment.

4-251. FIRE EXTINGUISHER AND DETECTION SYSTEM BOLT TORQUE VALUES. Unless otherwise noted, see paragraphs 4-259 and 4-260 for bolt torque values.

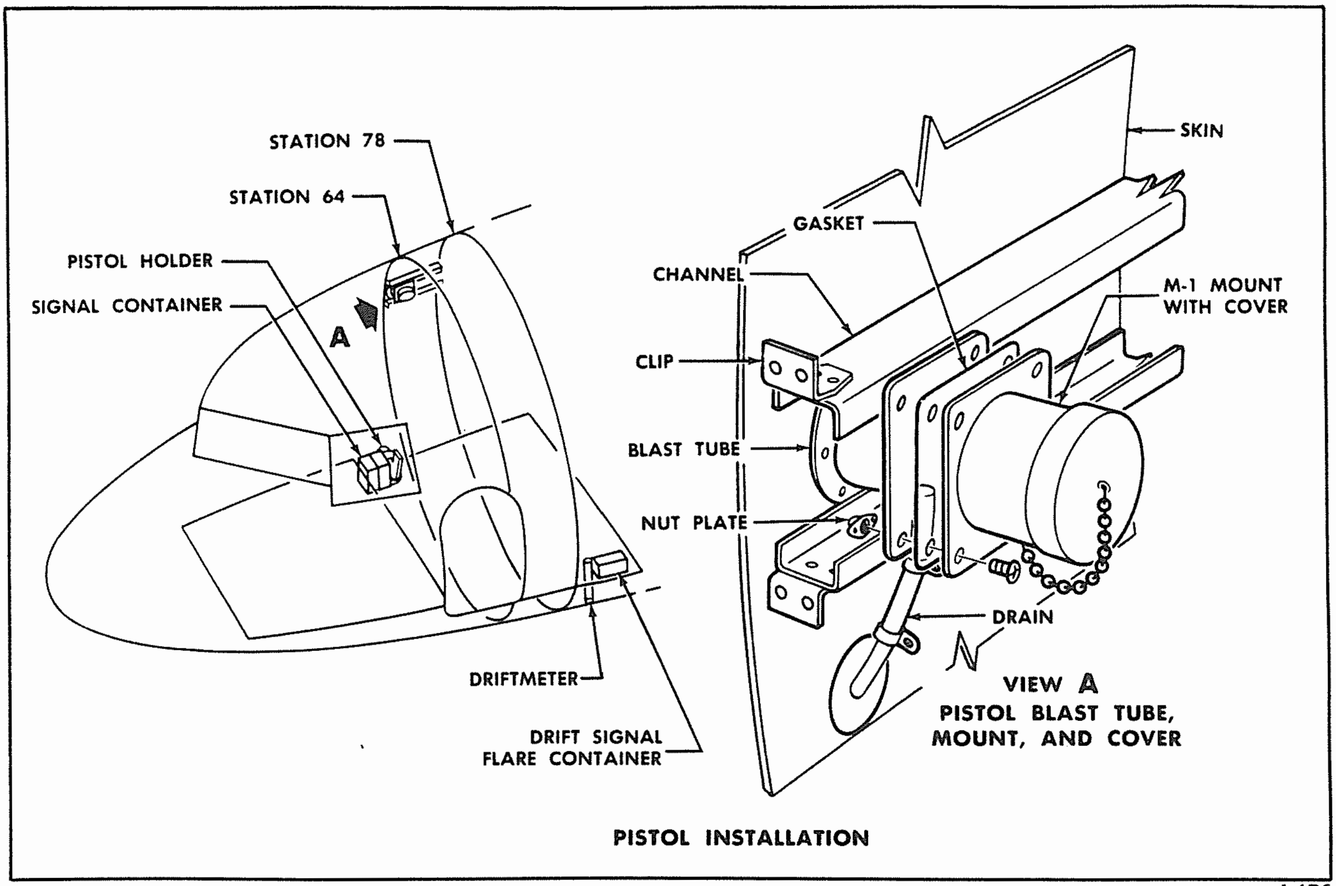

Figure 4-40. Pyrotechnic Equipment

4-252. PYROTECHNICS.

4-253. DESCRIPTION. *(See figure 4-40.)* The aircraft is equipped with a pyrotechnic pistol used for firing warning flares.

4-254. PYROTECHNIC PISTOL SYSTEM. The pyrotechnic pistol system consists of a blast tube located on the right side of the aircraft at the radio operator's station, an M-8 pyrotechnic pistol, and an M-1 mount for securing the pistol to the blast tube. The blast tube and mount serve to steady the pistol and prevents the blast from flaring inside. When not in use, the pistol and signals are stowed in containers attached to the floor adjacent to the radio operator's station at approximately station 60.

4-255. PYROTECHNIC PISTOL. *(See figure 4-40.)* The M-8 pyrotechnic pistol is a double-action, single-loading pistol used to project pyrotechnic signals from an aircraft in flight. To use the pistol, remove it from the holster and insert the end of the barrel into the M-1 mount, which is securely fastened to the blast tube. Twist the barrel to the right while it is being inserted to fasten the barrel into the mount. A "click" will indicate that the barrel is secure. Push the mount unlocking lever on top of the barrel to engage the lock catch. Open the breech by means of the breech-unlocking lever and insert the signal. Lock the breech in position and pull the trigger. To remove the pistol after use, pull the mount unlocking lever, twist the pistol to the left, and pull. The pistol is fired from the M-1 mount, which is fastened to the blast tube.

4-256. PYROTECHNIC BLAST TUBE. The blast tube is a heavy, hollow metal cylinder installed in the fuselage right side, at the radio operator's station. A drain tube extends down from the bottom inboard end of the blast tube to an outlet in the fuselage skin.

4-257. PYROTECHNIC MOUNT AND COVER. The M-1 mount, which is securely fastened to the blast tube, incorporates a spring at each of the four corners to absorb the recoil shock of the pistol. Around the inside of the mount are catches for attaching either the pistol or the pistol mount cover. The cover is provided in order that the hole through the mount to the outside may be kept closed when the pistol is not in use.

4-258. PYROTECHNIC PISTOL HOLDER AND SIGNAL CONTAINER. The A-8 signal container, located on the floor near the radio operator's station, is a satchel made of canvas-like material with a slide fastener for opening and closing. It contains pockets for 20 standard type signals. Signals should be inserted with firing caps up for easy removal. The A-2 pistol holder, a small bag also made of canvas-like material, is attached to the signal container by snap fasteners. The combination forms a portable unit.

4-259. UTILITY SYSTEM BOLT TORQUE VALUES.

4-260. DESCRIPTION. If structural tension bolts are not properly tightened when installed, vibration and fatigue stresses will materially shorten their service life. It is equally dangerous, however, to tighten a bolt excessively, since over-tightening may result in stripped threads, a bent or sheared bolt, or eventual tension failure. It is essential, therefore, to observe the torque limits specified here when installing tension bolts. Unless otherwise noted on the installation drawing, bolts and nuts must be installed free of lubricants. Otherwise, the specified torque reading may produce an incorrect tension load. Torque readings should not be made on bolts that show painting or corrosion, as it is impossible to determine accurately the amount of tension being applied. Bolts and nuts that are to be cotter-pinned may be torqued to the special values shown in order to permit the alignment of their holes. However, these figures must never be exceeded, nor may the nuts be backed off to align the holes, unless they are backed off sufficiently to completely re-torque properly. When re-torquing bolts, the nut must be backed off part of a turn and then retightened to the correct value. This is necessary because more torque may be required to break the nut free than is actually on the bolt. To apply 700 inch-pounds and fail to move the nut does not mean that the nut is torqued to 700 inch-pounds. The nut may have a torque of only 650 inch-pounds, but 750 inch-pounds may be required to break it free. A value in excess of the required torque may also be needed to break the nut free in a counter-clockwise direction. For this reason, the nut should be backed off and retightened, allowing no rotation of the bolt. Whenever possible, the torque reading must be taken on the nut and not on the bolt. Torque wrenches should be calibrated frequently to insure accurate readings. When adapters are used with torque wrenches, compensating torque values must be computed for each adapter and wrench involved. The following chart lists the standard torque values for structural tension bolts and also lists the bolts and nuts which should be installed with the torque values specified. Special torques are listed for all bolts to which the standard values are not applicable.

UTILITY SYSTEM BOLT TORQUE VALUE TENSION CHART

Bolt Size	*AN Type Bolts*	*AN365 and AN310 Nuts*	*Torque Values*	*Cotter Pin Maximum Torque*
10-32	AN-3 AN-173	−1032	20-25 inch-pounds	40 inch-pounds
¼-28	AN-4 AN-174	−428	50-70 inch-pounds	100 inch-pounds
5/16-24	AN-5 AN-175	−524	100-140 inch-pounds	225 inch-pounds
⅜-24	AN-6 AN-176	−624	160-190 inch-pounds	390 inch-pounds
7/16-20	AN-7 AN-177	−720	450-500 inch-pounds	840 inch-pounds
½-20	AN-8 AN-178	−820	480-690 inch-pounds	1100 inch-pounds
9/16-18	AN-9 AN-179	−918	800-1000 inch-pounds	1600 inch-pounds
⅝-18	AN-10 AN-180	1018	90-105 foot-pounds	200 foot-pounds
¾-16	AN-12 AN-182	1216	190-210 foot-pounds	415 foot-pounds
⅞-14	AN-14 AN-184	1414	210-250 foot-pounds	590 foot-pounds
1-14	AN-16 AN-186	1614	310-460 foot-pounds	835 foot-pounds
1⅛-12	AN-18	1812	415-585 foot-pounds	1250 foot-pounds

Bolt Size	*NAS Heat-Treated Bolts*	*Douglas Heat-Treated Bolts*	*Elastic Stop Heat-Treated Nuts*	*Torque Values*
¼-28	NAS-144	S-2076904	12B-048	73-100 inch-pounds
5/16-24	NAS-145	S-2076905	12B-054	145-200 inch-pounds
⅜-24	NAS-146	S-2076906	12B-064	230-280 inch-pounds
7/16-20	NAS-147	S-2076907	12B-070	650-720 inch-pounds
½-20	NAS-148	S-2076908	12B-080	700-1000 inch-pounds
9/16-18	NAS-149	S-2076909	12B-098	100-120 foot-pounds
⅝-18	NAS-150	S-2076910	12B-108	135-155 foot-pounds
¾-16	NAS-152	S-2076912	12B-126	280-300 foot-pounds
⅞-14	NAS-154	S-2076914	12B-144	300-360 foot-pounds
1-14	NAS-156	S-2076916	12B-164	450-665 foot-pounds
1⅛-12	NAS-158	S-2076918	12B-182	605-845 foot-pounds

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z

AIRCRAFT

SECTION V
POWER PLANT
AND
RELATED SYSTEMS

THIS SECTION SUPERSEDES SECTION V OF AN 01-40NK-2
DATED 15 MAY 1952 REVISED 1 NOVEMBER 1952

NOTE: A COPY OF AN 02A-35GK-2, SERVICE INSTRUCTIONS FOR MODEL R-1820-80 ENGINES, SHOULD BE FILED WITH THIS SECTION

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE
AND THE CHIEF OF THE BUREAU OF AERONAUTICS

15 April 1953

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SECTION V

POWER PLANT AND RELATED SYSTEMS

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SECTION V

POWER PLANT AND RELATED SYSTEMS

5-1. POWER PLANT.

5-2. DESCRIPTION. *(See figures 5-1 and 5-2.)* Each demountable power plant consists of an engine mount and the following units: a Wright R-1820-80 engine, engine accessories, cowling, carburetor air induction system, an exhaust system, and an oil cooler and ducting. Detailed maintenance instructions for the engines are not included in this manual (see the applicable technical orders). General information on removal and installation procedures for the engine and the engine mount is outlined in the following paragraphs. For information concerning the oil system, fuel system, electrical system, and the various power plant accessories, refer to the section or paragraph pertaining to the respective systems.

5-3. ENGINE COWLING. *(See figure 5-3.)* The engine cowling covers the engine section of each nacelle forward of the firewall and is designed to provide engine cooling sufficient to maintain temperature limits as specified for best engine performance, carburetor air, oil cooling, and cooling of generator and engine flexible mounts. The cowling consists of the engine section accessory cowling, anti-drag ring and cowl flaps and supporting structure, carburetor airscoop, oil cooler airscoop, the inner ring assembly, the engine seal-ring diaphragm, and the generator and engine mount blast tubes. For information on cowl flaps, see paragraph 5-222.

5-4. ACCESSORY COWLING. The accessory cowling is composed of four detachable stainless steel panels, two panels per side, and provides ready access to each engine accessory section. Camloc fasteners attach the forward ends of the panels to the inner ring assembly and the aft ends to the support ring assembly, (which is attached to the engine mount just forward of the firewall). The sides attach either to the adjacent cowl panels or to the carburetor airscoop fairing or the oil cooler fairing.

5-5. REMOVAL OF ACCESSORY COWLING.

a. Loosen the Camloc fasteners of the top panels (one on each side of the engine accessory section) and lift off the panels.

b. Loosen the Camloc fasteners of the lower panels (one on each side of the engine accessory section), and lift off the panels.

5-6. INSTALLATION OF ACCESSORY COWLING. Reverse the removal procedure.

5-7. ANTI-DRAG RING. The anti-drag ring for each engine is composed of three aluminum-alloy segments held together by quick-attach hook and latch assemblies that, when latched, tighten the anti-drag ring about the bow-ring support. This bow-ring support also supports the cowl flaps. Each segment of the anti-drag ring incorporates three hook assemblies on one end and three latch assemblies on the other end. Dowel pins are provided to align the segments. The end of each segment incorporating latch assemblies also incorporates three doors that cannot be closed unless a locking plunger in the latch assembly has been snapped in, thus providing a positive and foolproof locking of the segment latch assemblies. The doors are locked in the closed position by a Camloc fastener. The top segment of the anti-drag ring incorporates an under-cowl-type carburetor air-intake duct leading to the carburetor airscoop elbow. The anti-drag ring also incorporates a rubber seal, or baffle, so that when the anti-drag ring is installed, all the engine cooling air is forced to pass through the cooling fins on the cylinders and cylinder heads.

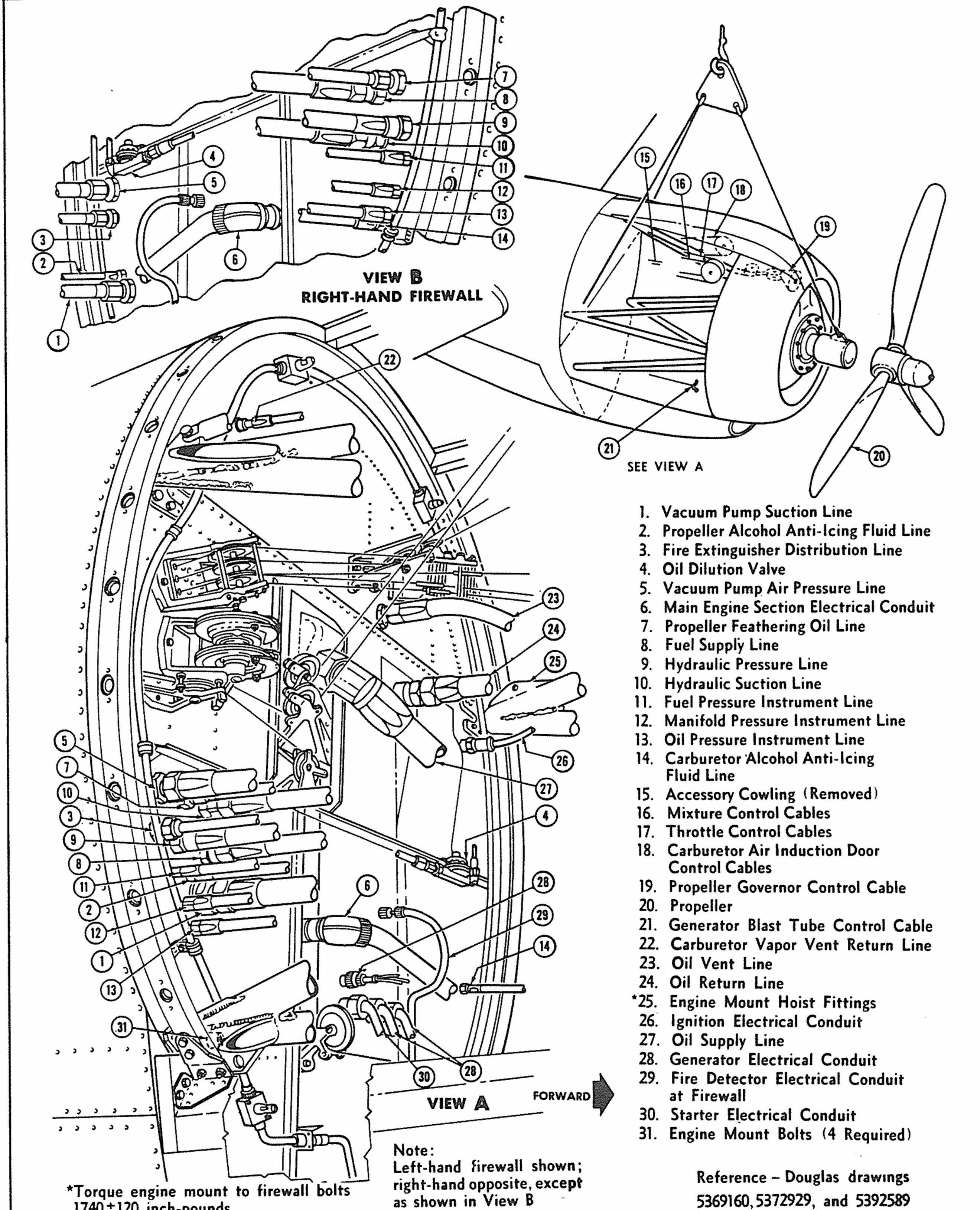
5-8. REMOVAL OF ANTI-DRAG RING.

a. Loosen the Camloc fasteners locking the three doors on each of the anti-drag ring segments to expose the latch assemblies.

b. Using a coin or screw driver, lift the plunger at the center of the hex head of the latch assembly. The travel on this plunger is approximately $\frac{9}{32}$ inch, and it should be easily operable without the use of excessive force.

c. Use a $\frac{3}{4}$-inch box or socket wrench on the hex head of each latch assembly and turn in a counterclockwise direction to loosen the hex head.

d. Remove the two lower segments first. The top segment may then be lifted free.



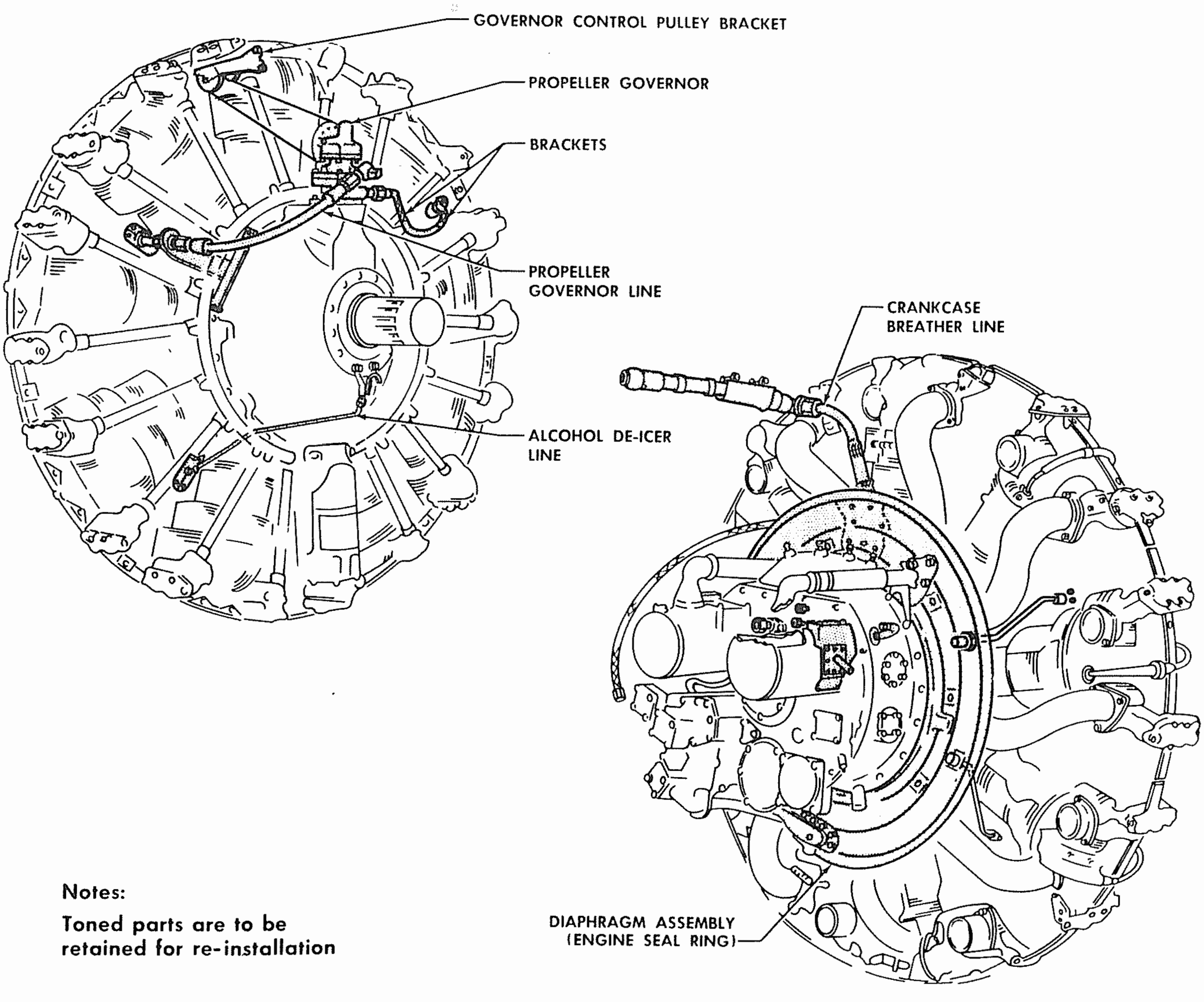
1. Vacuum Pump Suction Line
2. Propeller Alcohol Anti-Icing Fluid Line
3. Fire Extinguisher Distribution Line
4. Oil Dilution Valve
5. Vacuum Pump Air Pressure Line
6. Main Engine Section Electrical Conduit
7. Propeller Feathering Oil Line
8. Fuel Supply Line
9. Hydraulic Pressure Line
10. Hydraulic Suction Line
11. Fuel Pressure Instrument Line
12. Manifold Pressure Instrument Line
13. Oil Pressure Instrument Line
14. Carburetor Alcohol Anti-Icing Fluid Line
15. Accessory Cowling (Removed)
16. Mixture Control Cables
17. Throttle Control Cables
18. Carburetor Air Induction Door Control Cables
19. Propeller Governor Control Cable
20. Propeller
21. Generator Blast Tube Control Cable
22. Carburetor Vapor Vent Return Line
23. Oil Vent Line
24. Oil Return Line
*25. Engine Mount Hoist Fittings
26. Ignition Electrical Conduit
27. Oil Supply Line
28. Generator Electrical Conduit
29. Fire Detector Electrical Conduit at Firewall
30. Starter Electrical Conduit
31. Engine Mount Bolts (4 Required)

*Torque engine mount to firewall bolts 1740±120 inch-pounds

Note:
Left-hand firewall shown; right-hand opposite, except as shown in View B

Reference – Douglas drawings 5369160, 5372929, and 5392589

Figure 5-1. Power Plant Installation



Notes:

Toned parts are to be retained for re-installation

Figure 5-2. Engine Adapting Parts Diagram

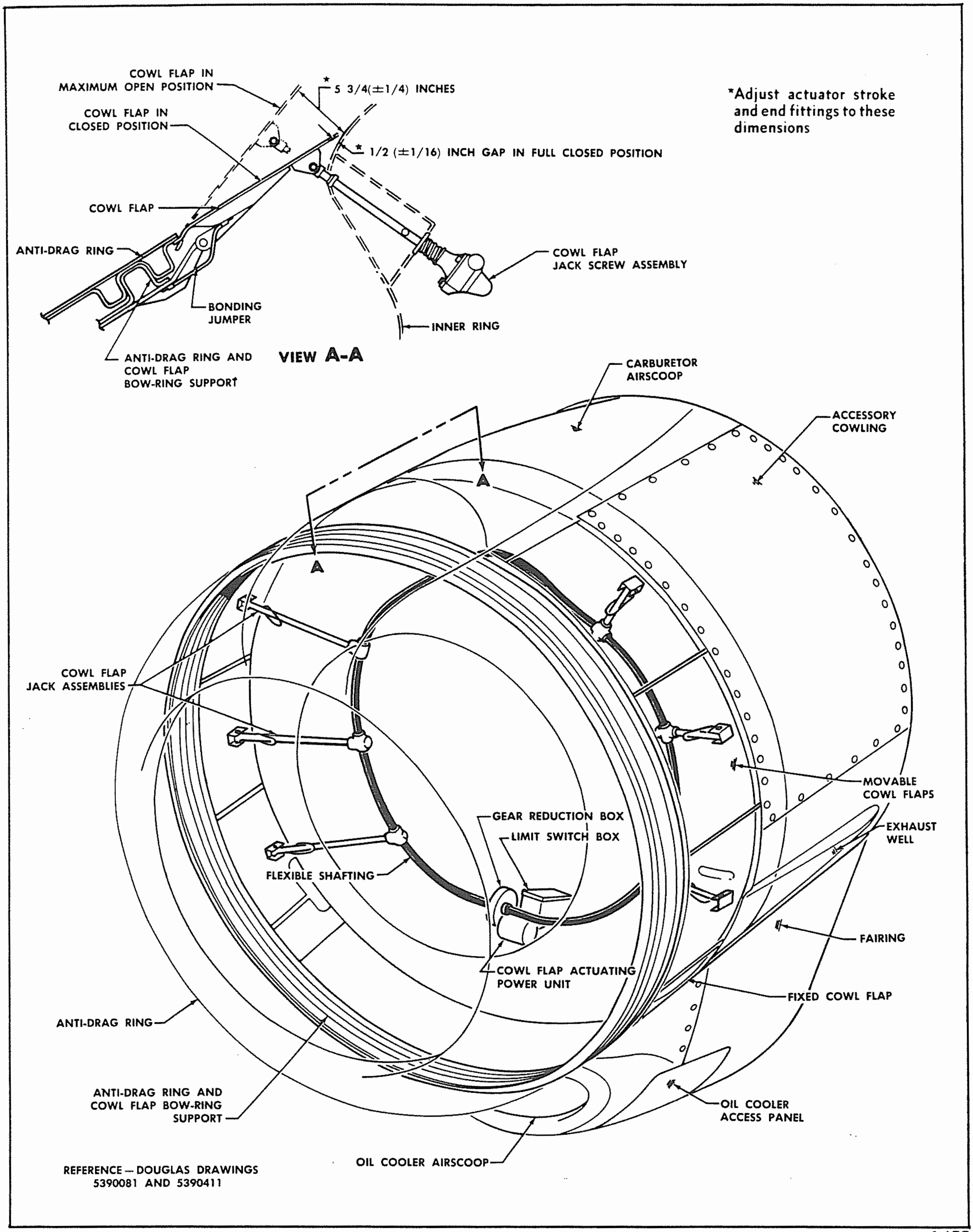


Figure 5-3. Engine Cowling

5-9. INSTALLATION OF ANTI-DRAG RING. Reverse the removal procedure.

5-10. DEMOUNTABLE POWER PLANT. The demountable power plant consists of the engine mount and the following units: one Wright R-1820-80 engine, the engine accessories, the cowling, the carburetor air induction system, the exhaust system, and the oil cooler and ducting.

5-11. REMOVAL OF DEMOUNTABLE POWER PLANT.

Note

Before removal for storage, accomplish preservation procedures as described in applicable technical orders.

a. Remove the engine accessory cowling segments and anti-drag rings (see paragraphs 5-5 and 5-8).

b. Remove the propeller (see paragraph 5-202).

c. Place the fire shutoff valve control in the OFF position. Drain oil from the oil cooler drain plug.

d. Disconnect the oil supply pipe at the firewall and cap the pipe.

e. Disconnect the oil return pipe at the firewall and cap the pipe.

f. Disconnect the oil vent pipe at the firewall.

g. Disconnect the fuel supply pipe at the firewall, and cap the pipe.

h. Disconnect the carburetor vapor vent return pipe at the firewall.

i. Disconnect the propeller feathering oil pipe at the firewall.

j. Disconnect the vacuum pump suction pipe at the firewall.

k. Disconnect the vacuum pump air pressure pipe at the firewall.

l. Disconnect the hydraulic pressure and suction pipes at the firewall.

m. Disconnect the engine section fire extinguisher distribution pipe at the firewall.

n. Disconnect the direct-reading instrument pipes, including oil pressure, fuel pressure, and manifold pressure, at the firewall.

o. Disconnect the propeller and carburetor alcohol anti-icing fluid pipe at the firewall.

p. Disconnect the ignition, starter, and generator electrical conduits at the firewall.

q. Disconnect two fire detector electrical cables at the firewall.

r. Disconnect the electrical ground cable at the firewall.

s. Disconnect the carburetor air induction door cable at the quick-disconnect pulley at the carburetor.

t. Disconnect the propeller control cables at the quick-disconnect pulleys at the firewall.

u. Disconnect the throttle and mixture control cables at the quick-disconnect pulleys at the bracket mounted on the left side of the engine.

v. Disconnect the generator blast tube shutoff valve control cable at the valve.

w. Disconnect the main electrical conduit running to the engine junction box at the firewall.

x. Attach the hoist sling to the two engine mount hoist eyes and to the propeller shaft.

y. Loosen the nuts on the four engine mount bolts, but do not remove them.

z. Remove the lower engine mount bolts. Make certain that all parts of the engine are disconnected and clear, and that the hoist attachments are tight.

aa. Remove the two upper engine mount bolts and swing the engine forward and up.

5-12. INSTALLATION OF DEMOUNTABLE POWER PLANT. Reverse the removal procedure.

Note

When installing the engine mount attach bolts at the firewall, torque each of the four bolts to 1620 to 1860 inch-pounds before installing the safety wire. Before starting the engine, accomplish depreservation procedures as described in applicable technical orders.

5-13. ANTI-DRAG RING AND COWL FLAP BOW-RING SUPPORT. The bow-ring for each engine is an Alclad hat section assembly built in two sections and provides support for the anti-drag ring and cowl flaps. The bow-ring is supported by four stainless steel support beams and four A-frame supports. The four stainless steel support beams attach with screws to the inside of the bow-ring, and extend aft and bolt to the forward side of the inner ring. At the two top supports, the load is transferred through fittings on the inner ring to a tube assembly, which extends aft and is bolted to the aft end of the engine mount assembly. The four A-frame supports bolt to the stainless steel support beams at the bow-ring and to lugs on the engine mount ring that extend through the inner ring. Each bow-ring incorporates 12 cowl flap hinge assemblies and 11 laminated phenolic pads. The hinge assemblies, two for each movable cowl flap, include a shock-mount for the flaps. The phenolic pads provide rubstrips for the anti-drag ring when attached. The anti-drag and cowl flaps are grounded to the engine mount assembly with bonding strips, or jumpers which are attached to the bow-ring hat section flange.

5-14. REMOVAL OF ANTI-DRAG RING AND COWL FLAP BOW-RING SUPPORT. The following procedure removes the cowling bow-ring support with the cowl flaps attached. For removal of an individual flap, see paragraph 5-223.

a. Place the cowl flaps selector switch for each engine in the OPEN position. The switch is located on the left overhead electrical panel in the flight compartment.

b. Remove each engine section anti-drag ring (see paragraph 5-8).

c. Remove one bolt connecting each cowl flap jackscrew assembly jackshaft end fitting to its flap.

Note

Unless the jackscrew assemblies are to be removed, do not turn the shaft after disconnecting all the jackshaft end fittings. Loop a wire through the end fitting mounting holes and secure it to the adjacent structure. This will insure correct setting of the shafts at installation.

d. Loosen the Camloc fasteners and remove the lower fixed cowl flaps (one on each side of each engine section).

e. Remove the screws that attach the oil cooler airscoop to the cowling bow-ring support.

f. At the bow-ring attachment on each engine, remove the screws attaching each of the four bow-ring supports between the bow-ring and inner ring. The four A-frame assemblies which are attached to the engine mount ring at these points do not have to be disconnected.

g. Remove the cowling bow-ring support assembly, lifting it forward over the engine cylinders. If the propellers are installed, remove the screws at the top and bottom splices of the bow-ring and remove it in two halves.

5-15. INSTALLATION OF ANTI-DRAG RING AND COWL FLAP BOW-RING SUPPORT. Reverse the removal procedure.

5-16. INNER RING AND SEAL-RING DIAPHRAGM. The inner ring and seal-ring diaphragm are stainless steel assemblies on each engine, which combine to form a tight fire seal between the engine and the engine accessory section. The inner ring is bolted to the engine mount assembly in five places and supports or assists in supporting the following: the carburetor airscoop elbow, the leading edge of the accessory cowling, the oil cooler supports, the oil cooler airscoop, the fixed cowl flaps, the cowl flaps actuator power unit and jackscrew assemblies, the generator and engine mount blast tube shutoff valve assembly, the engine section drip pan, the engine accessory section electrical junction box, the engine section fire detectors and some of the accessory section fire detectors, two of the propeller governor cable pulleys and mounting brackets, and supports for electrical conduit and plumbing pipes. Since the inner ring is a spot-welded, one-piece assembly, it is not removable except when the engine has been removed from its mounting. The seal-ring diaphragm is mounted forward of the engine mounting ring that supports the engine flexible mounts and is held in place between the shock mounts and the mounting bosses on the engine by the nine bolts that attach the shock mounts and links to the engine. The perimeter of the seal-ring incorporates an asbestos tape and is designed so that upon engine installation approximately 1/8-inch deflection of the perimeter will occur upon its contact with the inner ring, thus forming a tight fire seal. The seal-ring is built in five segments and is held together by bolts and nuts at each joint. The seal-ring is interchangeable between left and right engines. The inner ring is made interchangeable by changing the exhaust stack well covers to the inboard side.

5-17. REMOVAL OF INNER RING. The following procedure removes the inner ring with the anti-drag ring and cowl flaps bow-ring support, cowl flaps, and the bow-ring supports as one assembly. For removal of the anti-drag ring and bow-ring support, see paragraph 5-14.

a. Remove the engine section accessory cowling (see paragraph 5-5).

b. Remove the engine section anti-drag ring (see paragraph 5-8).

c. Remove the engine section carburetor airscoop (see paragraph 5-189).

d. Remove the engine section oil cooler airscoop.

e. Remove the engine from its mount assembly (see paragraph 5-11).

f. Remove two bolts at the five attachment points of the inner ring formers to the engine mount tubes in the engine accessory section.

g. Disconnect the two upper anti-drag ring and cowl flap bow-ring support tube assemblies at the aft side of the inner ring.

h. Disconnect the four anti-drag ring and cowl flap bow-ring support A-frames where they attach to lugs on the engine mount ring at the forward side of the inner ring.

i. Lift the inner ring assembly free.

5-18. INSTALLATION OF INNER RING. Reverse the removal procedure.

5-19. REMOVAL OF SEAL-RING DIAPHRAGM. Removal of the seal-ring diaphragm is not possible without first removing the engine from its mount assembly.

a. Perform steps a through e of paragraph 5-17.

b. Remove the bolts attaching the engine shock mounts to the engine bosses, and remove the shock mounts and seal-ring diaphragm.

5-20. INSTALLATION OF SEAL-RING DIAPHRAGM. Reverse the removal procedure.

5-21. ENGINE MOUNT. (*See figures 5-4 and 5-5.*) The engine mount is a welded chrome-molybdenum steel structure, consisting of a forward ring to which eight tubular truss members are welded. The forward ring incorporates forged lugs, in which rubber shock mount assemblies are installed. The four fittings which attach the mount to the firewall on the aft end of the engine mount are welded to the truss members.

522 REMOVAL OF ENGINE MOUNT. (*See figure 5-5.*) The demountable power plant is removed from the engine mount as follows:

a. Install the demountable power plant on an engine stand and secure the clamp on the propeller shaft.

b. Remove all accessories, clamps, cables and rods supported by the engine mount.

c. Attach the hoist to the tubular engine mount.

d. Remove the eight large mounting bolts from the engine bosses and the shock mounts.

e. Remove the bolts that lock the balance bars to the shock mount lugs.

f. Rotate the shock mount lugs away from the engine bosses and fire seal.

g. Remove the engine mount from the engine and fire seal.

5-23. MINOR REPAIR AND REPLACEMENT OF ENGINE MOUNT.

a. Inspect the rubber shock mounts on the forward ring of the engine mount for failure of the rubber bond or for excessive disintegration of the rubber due to contact with fuel or oil. Replace the shock mounts when necessary, as shown in figure 5-4.

b. Line maintenance repairs of the engine mount structure are not considered feasible because the engine mount must be installed in a jig before structural repairs can be made (see the applicable handbook of structural repair).

5-24. INSTALLATION OF ENGINE MOUNT.

a. Disassemble the shock mount balance bars and install the flexible mounts on the tubular engine mount ring, torquing the mounting bolts to 325 to 400 inch-pounds.

CAUTION

When installing the flexible cores, be sure that the proper side of the core is facing forward, as indicated by the word FRONT molded in the rubber.

b. Rotate the mounting bracket approximately 90 degrees, in order to clear the engine parts when the engine is installed on the mount.

c. Install the fire seal baffle on the engine (see paragraph 5-17, step g).

d. Move the engine mount assembly into position behind the engine and rotate the shock mount lugs back into position.

e. Install the shock mount balance bars with eight large mounting bolts (AN9-64), each with a spherical washer through the engine bosses.

Note

Tighten the engine mount bolts and balance bar bolts to the following prescribed torque value: engine mount nut (AN355-9) 750 to 850 inch-pounds; balance bar nut (AN365-428) 60 to 80 inch-pounds.

f. Lock the balance bar bolts (AN4-11A) to lock the balance bars to the shock mount lugs.

CAUTION

When tightening the main attach bolts at the engine pads, be sure the mounting bracket is seated against the shoulder of the core assembly. Check fire seal and diaphragm fit for gaps and tightness.

5-25. POWER PLANT BOLT TORQUE VALUES. Unless otherwise noted, see paragraph 5-263 for power plant bolt torque values.

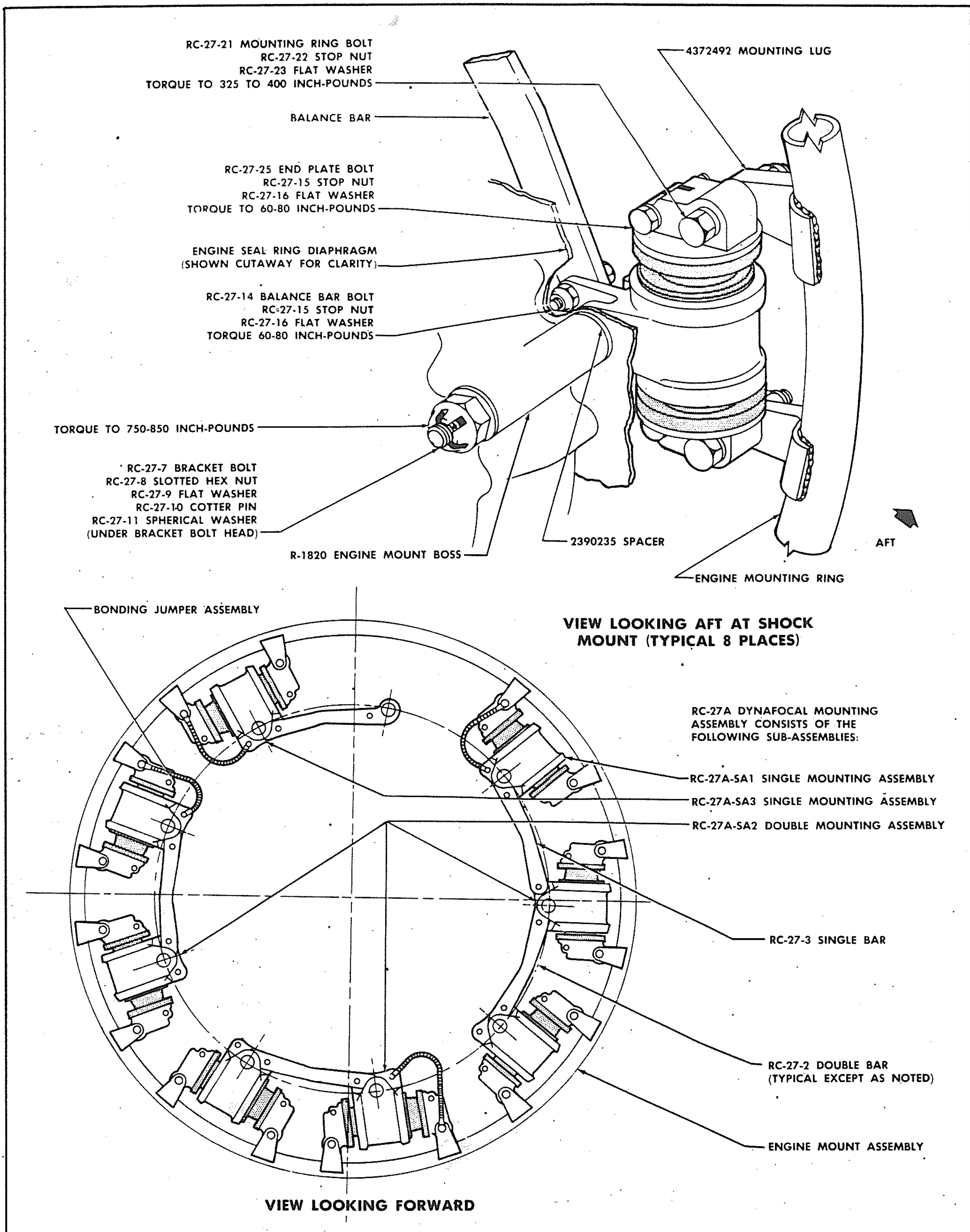


Figure 5-4. Engine Shock Mounts

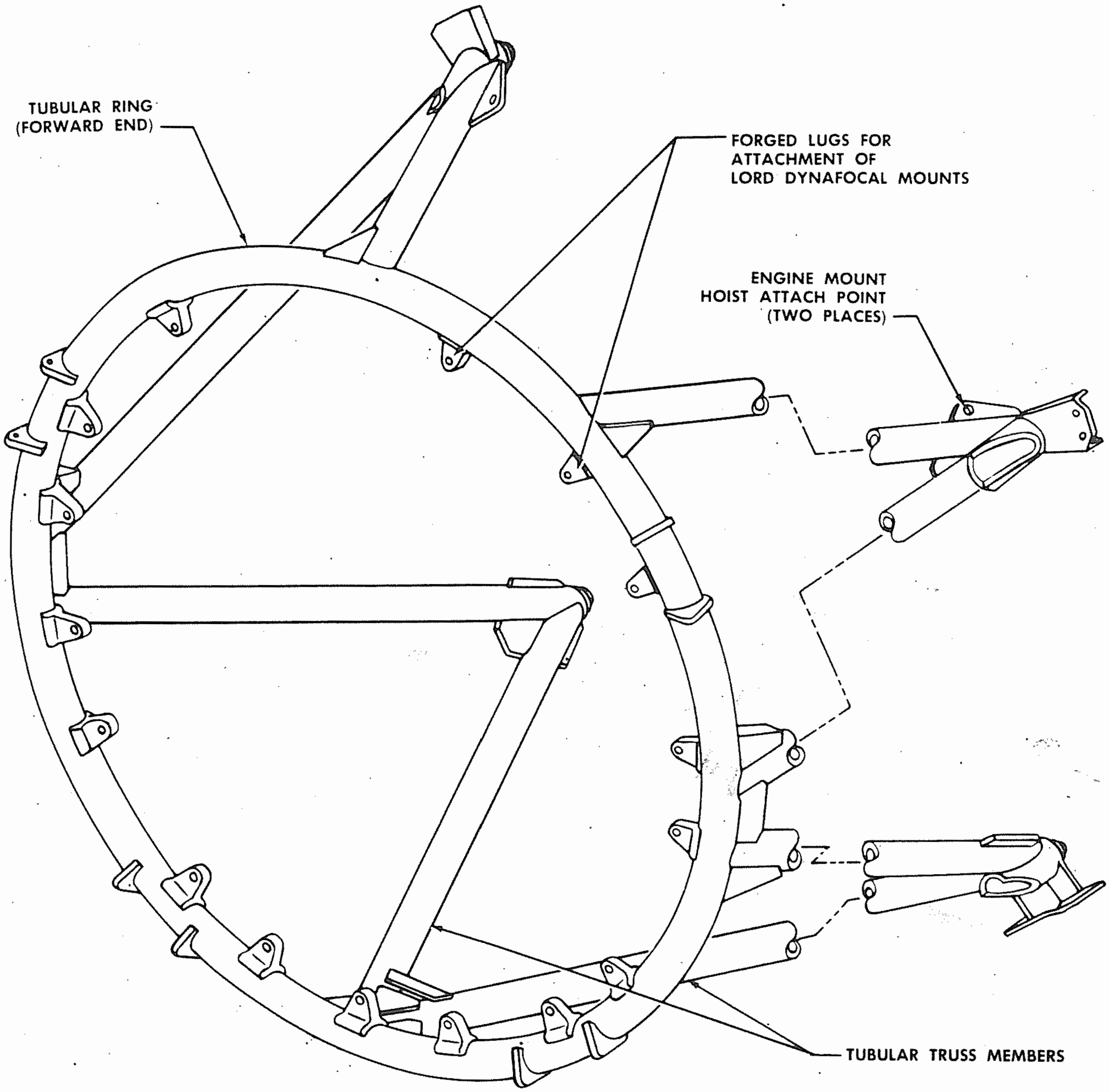


Figure 5-5. Engine Mount

5-26. FUEL SYSTEM.

5-27. DESCRIPTION. *(See figure 5-6.)* Fuel is supplied to the two engines from four fuel tanks mounted in the wing center section and two sets of bladder-type fuel cells located in the outer wings. For fuel tank capacities, see paragraph 1-18. Fuel quantity for each of the four center wing fuel tanks is measured by a Liquidometer unit located in each fuel tank. The outer wing fuel tank quantity is measured by two Liquidometer units for each set of fuel cells. Three fuel quantity indicating gages are mounted on the instrument panel. The four center wing fuel tank Liquidometers are wired to a selector-type fuel gage that includes a switch to select the tank for which the fuel quantity is to be determined. When the selector switch is turned to the fuel tank desired, a dial on the fuel quantity gage calibrated for that particular fuel tank will automatically come into view. The left and right outer wing tanks each have a separate fuel quantity gage, which indicates the fuel quantity of each set of fuel cells at all times when the ground power or aircraft's batteries are connected and the fuel quantity circuit breaker is closed. Under normal operating conditions, the left engine draws fuel from the left outer wing tank and the left center wing tanks, and the right engine draws fuel from the right outer wing tank and the right center wing tanks. Fuel can be used from any one tank for either or both engines, by using the selector valves located on the control pedestal. For best fuel usage, fuel tanks should be filled in the following sequence: front tanks; rear tanks; outer wing tanks. A fuel strainer is provided in the filler neck of each center wing and outer wing fuel tank. In addition to the strainers in the filler necks, strainers are also incorporated in the fuel system pipes between the selector valves and the left and right fuel booster pumps. The fuel flows from any one tank through the selector valve, through the strainer, through the booster pump, through the emergency firewall shutoff valve to the engine-driven fuel pump, and then to the carburetor. A vapor overflow pipe connects from the top chamber of the carburetor to the No. 2 and No. 3 front center wing tanks. A fuel pressure pipe from the back of each engine carburetor connects to the fuel pressure gage on the pilot's instrument panel. Fuel is supplied to the engine by the fuel booster pumps at 20.5 to 21.5 psi and by the engine-driven pumps at 19 to 21 psi. A hose assembly from the fuel pressure tee fitting on the forward side of each firewall connects to an oil dilution solenoid valve, which releases fuel into the engine oil system. The engine primer solenoid valve is mounted on each carburetor and allows priming of the engines through external pipes to two points near the carburetor bottom flange. Each of the four center wing tanks is vented overboard, and the vent pipes are equipped with siphon breaker pipes. The outer wing fuel tanks are vented overboard through a vent chamber located in the top of the wing aft of the front spar, between wing stations 152 and 179, and then through a vent pipe from the venturi chamber to the bottom skin of the wing aft of the rear spar, at wing station 216.

5-28. TROUBLE SHOOTING OF FUEL SYSTEM.

Trouble	*Probable Cause*	*Remedy*
a. Lack of fuel pressure.	Fuel tanks empty.	Fill fuel tanks.
	Broken or disconnected fuel pipes.	Repair or replace fuel pipes.
	Fuel tank selector controls not in correct position.	Place controls in correct position. Check adjustment of cable controls. Check condition of selector valves.
	Defective fuel pressure indicator.	Replace fuel pressure indicator.
	Clogging of fuel pressure indicator pipes; clogged fuel pressure restrictor fitting or main fuel pipes.	Remove stoppage.
	Clogged fuel strainer.	Clean strainer.
	Bottom plate on fuel strainer loose.	Install plate in place and safety wing nut.
	Malfunctioning of engine-driven fuel pump.	Operate fuel booster pumps. If fuel pressure is obtained, engine-driven fuel pump is faulty. Replace pump.
b. Low fuel pressure.	Malfunctioning of engine-driven fuel pump.	Check adjustment of pressure relief valve screw. Replace pump if necessary.
	Defective fuel pressure indicator.	Replace fuel pressure indicator.
	Clogged fuel pressure indicator pipes; clogged fuel pressure restrictor fitting or main fuel pipes.	Remove stoppage.
	Broken or disconnected fuel pipes.	Repair or replace fuel lines.
	Clogged fuel strainer.	Clean strainer.
	Bottom plate on fuel strainer loose.	Install plate in place and safety wing nut.
	Ruptured diaphragm in engine-driven fuel pump.	Replace fuel pump.
c. High fuel pressure.	Adjustment screw for pressure relief valve on engine-driven fuel pump set incorrectly.	Correct setting of adjustment screw.
	Defective fuel pressure indicator.	Replace fuel pressure indicator.

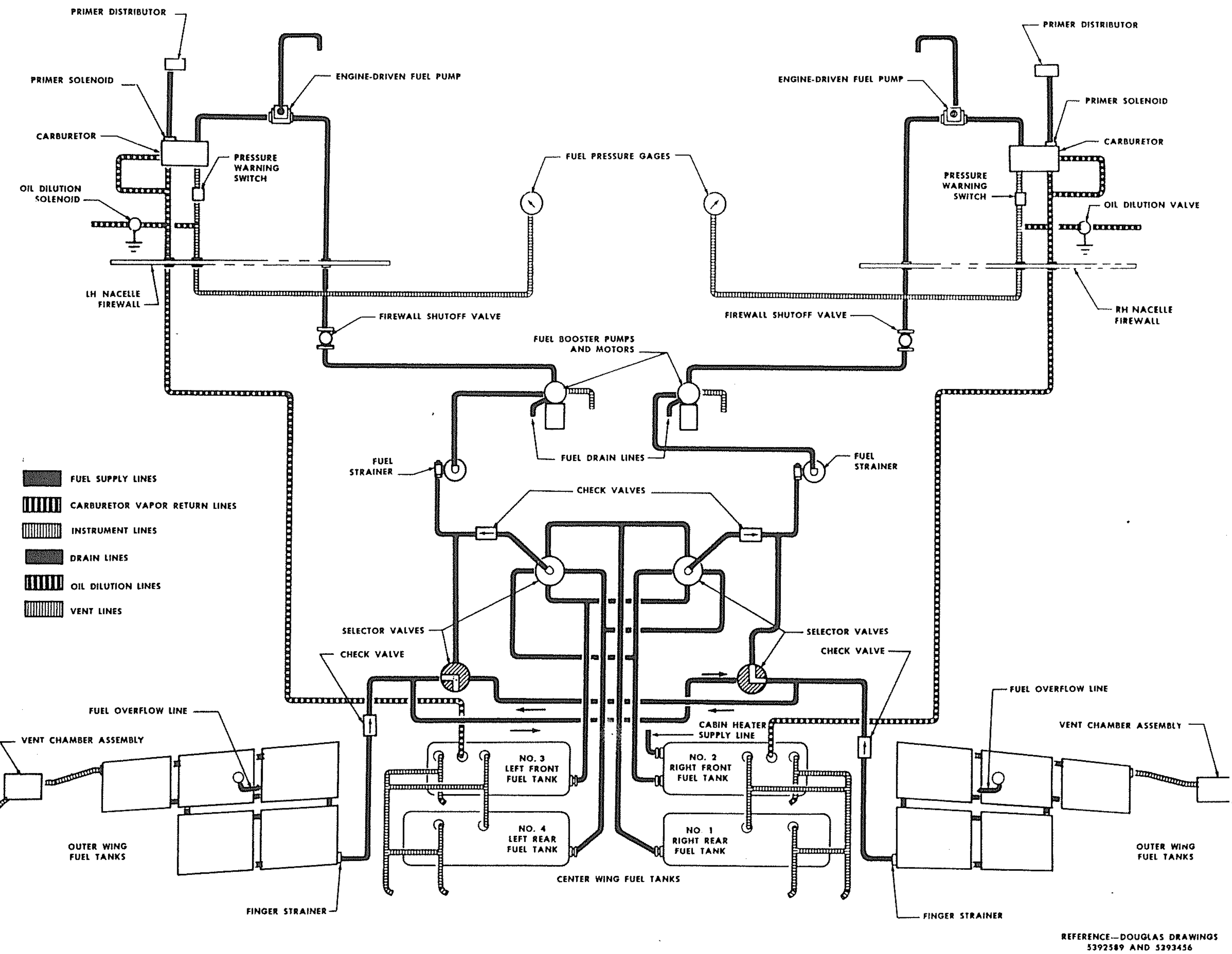


Figure 5-6. Fuel System Schematic

Trouble	Probable Cause	Remedy
c. *(Continued)*	Failure of pressure relief valve on engine-driven fuel pump (indicated by normal fuel pressure indications at idling speed, with an increase to extremely high pressure as rpm increases).	Replace engine-driven fuel pump.
d. Fluctuating fuel pressure.	Defective fuel pressure indicator.	Replace fuel pressure indicator.
	Malfunctioning of engine-driven fuel pump.	Replace pump.
	Air in fuel system caused by the following:	
	Brief operation on empty tank.	Select tank with fuel and operate booster pump to force air from system.
	Loose pipes or fittings between fuel tank and engine-driven fuel pump.	Tighten loose pipes and fittings; operate booster pump to force air from system.
	Hot fuel or high-altitude flying, resulting in vaporized fuel, which forms air bubbles in system.	Use booster pump to pressurize system, thereby preventing fuel vaporization.
	Malfunctioning of carburetor.	Bleed or replace carburetor.
	Clogged carburetor vapor vent return pipe.	Remove stoppage.
e. Abnormal increase or decrease in fuel pressure during variations in altitude.	Clogged fuel pressure restrictor fitting.	Remove stoppage.
	Clogged balance pipe on engine-driven fuel pump.	Remove stoppage.
	Clogged carburetor vapor vent return pipe.	Remove stoppage.
f. Fuel booster pump pressure is incorrect.	Loose connections or high resistance.	Tighten and clean connections.
	Low voltage.	Check voltage across pump.
	Malfunctioning of pump.	Replace pump.
g. Repeated opening of fuel booster pump circuit breaker.	Excessive current demand.	Check wiring; replace pump if necessary.
h. Fuel booster pump noisy.	Malfunctioning of pump.	Replace pump.
i. Fuel booster pump discharge pressure surges.	Variation in supply voltage.	Check and correct voltage.
	Loose connections.	Clean and tighten connections.
	Fuel in fuel booster pump motor.	Replace pump.
	Fuel in tank too low.	Add fuel.
j. External fuel leaks on lower surface of wing.	Faulty sump drain cock.	Replace drain cock.
	Loose or broken fuel pipe or connection.	Tighten or replace fuel pipe or hose connection.
	Faulty fuel strainer drain cock or leaky strainer.	Replace drain cock and/or strainer.
	Leaky fuel tanks.	Remove tanks, repair and reinstall.
k. Primer solenoid valve fails to operate.	Ground, short, or open wiring.	Check wiring *(see figure 10-21)*.
	Faulty priming switch.	Remove and replace switch.
	Faulty selector switch.	Remove and replace switch.
	Faulty priming solenoid valve.	Remove and replace solenoid valve.

5-29. GENERAL SAFETY PRECAUTIONS.

5-30. FIRE PREVENTION. Make every possible effort to prevent fire or explosion by observing the following safety precautions before and during any fuel system maintenance:

a. Remove all possible fire hazards, such as drained fuel and gasoline or oil-soaked rags, which may be in or about the aircraft.

b. Disconnect the battery cart and aircraft batteries from the aircraft.

c. Electrically ground the aircraft.

d. Clear the fuel tanks of fuel vapor with an air blast prior to beginning any work on them.

5-31. FUEL SYSTEM OPERATIONAL CHECK. It is *mandatory* to perform a thorough operational check of the fuel system after completing any fuel system maintenance and before any flight.

a. Start and run each engine on its respective fuel tanks.

b. Operate each selector valve control to obtain the various tank-to-engine fuel flow combinations. Check to be sure that each selector valve is rigged to open and close correctly.

5-32. DRAINING FUEL TANKS. *(See figures 5-7 and 5-8.)* For information on draining the fuel tanks, see paragraph 1-88.

5-33. FILLING FUEL TANKS. For detailed instructions on filling the center wing and outer wing fuel tanks, see paragraph 1-87.

5-34. CENTER WING FUEL TANKS. *(See figure 5-9.)* The center wing fuel tanks are constructed by welding together an upper and lower shell of 3-S half-hard aluminum sheets. Each tank is equipped with a water-collecting sump and a drain cock located on the bottom of the tank *(see figure 5-10)*. An electric Liquidometer unit in the top of each tank connects to an indicator on the instrument panel. The 2 forward tanks, each with a capacity of 202 US. (168.19 Imp.) gallons, are mounted in the center wing of the aircraft between the front and center wing spars on each side of the aircraft center line *(see figure 5-9)*. The 2 aft tanks, with a capacity of 200 US. (166.52 Imp.) gallons each, are mounted in the center wing of the aircraft between the center and rear spars on each side of the center line of the aircraft. The tanks are supported by a padded cradle assembly, held in place by shock-mounted straps.

5-35. REMOVAL OF CENTER WING FUEL TANKS. The same removal procedure is used for all four center wing fuel tanks.

a. Drain the fuel tanks.

b. Remove the top plate from around the filler neck by removing the attach screws. Allow the rubber seal to remain with the bank.

c. Disconnect the vent pipes, the carburetor vapor return pipe, and Liquidometer tank unit electrical wiring, working through the inspection holes near the filler neck.

d. Remove the bottom plating from the wing center section under the tank by removing the line of screws on the spars at the front and rear ends of the plate and unfastening the bolts in the flanges at the sides of the plate.

Note

At least two men will be required to remove the bottom plates. It will be found helpful if, when removing the screws from the forward and aft edges of the cover plate, two screws are left in near the center of the plate at each edge. Then, with one man supporting the plate, these screws may be removed by the second man.

e. Disconnect the electrical bonding braids from the tank bay. Do not disconnect the bonding braids from the tank and do not fasten the bonding braids together, as there is danger of damaging the soldered tank connection or of pulling the bonding braids loose.

f. Remove the cover from the tunnel between the left and right tanks by removing the screws at the forward and aft ends of the tunnel cover. This exposes the main fuel pipes.

g. Disconnect the main fuel pipe at the tank fitting and at the tee on the pipe leading to the selector valves. Remove this section of fuel pipe.

h. Remove the fuel pipe fitting from the tank.

i. Unscrew the turnbuckles in the tank support straps and lower the tank downward. Two men will be required.

5-36. MINOR REPAIR AND REPLACEMENT OF CENTER WING FUEL TANKS. Repairs on fuel tanks requiring the application of heat should not be made while the tanks are installed in the aircraft. Electrically ground the tanks removed from the aircraft prior to starting the repair work, and keep them grounded at all times while the work is progressing. Use only explosionproof lights when repairing fuel tanks. A fuel tank requiring repairs by welding or by the use of an open flame must be drained, removed from the aircraft, and thoroughly cleaned and prepared for such work.

WARNING

To eliminate any possibility of igniting explosive gases when repairing nonself-sealing aircraft fuel tanks that have contained fuel, the following precautions must be taken.

a. Flush the tank with hot water admitted at the bottom of the tank and allowed to overflow at the top. This operation removes deposits of fuel adhering to the inside surfaces of the tank.

b. After flushing with water, clean with live steam. Pass the steam through the tank for a minimum period of three hours. Mount the tank so that an opening is located at the top and one is located at the bottom, at the lowest point of the tank. Feed the live steam in at the top of the opening and allow the steam to escape through the bottom opening, closing all other openings. Steam vapor cleaners, such as the Kerrick or Hy-Pressure Jenny cleaners, should not be used in lieu of live steam cleaning, as the temperature of the vapor from such cleaners is not high enough to positively remove the danger of explosion. They may, however, be used as a substitute for the hot water flushing.

c. If facilities for cleaning the tank with live steam are not available, clean the tank as specified by hot

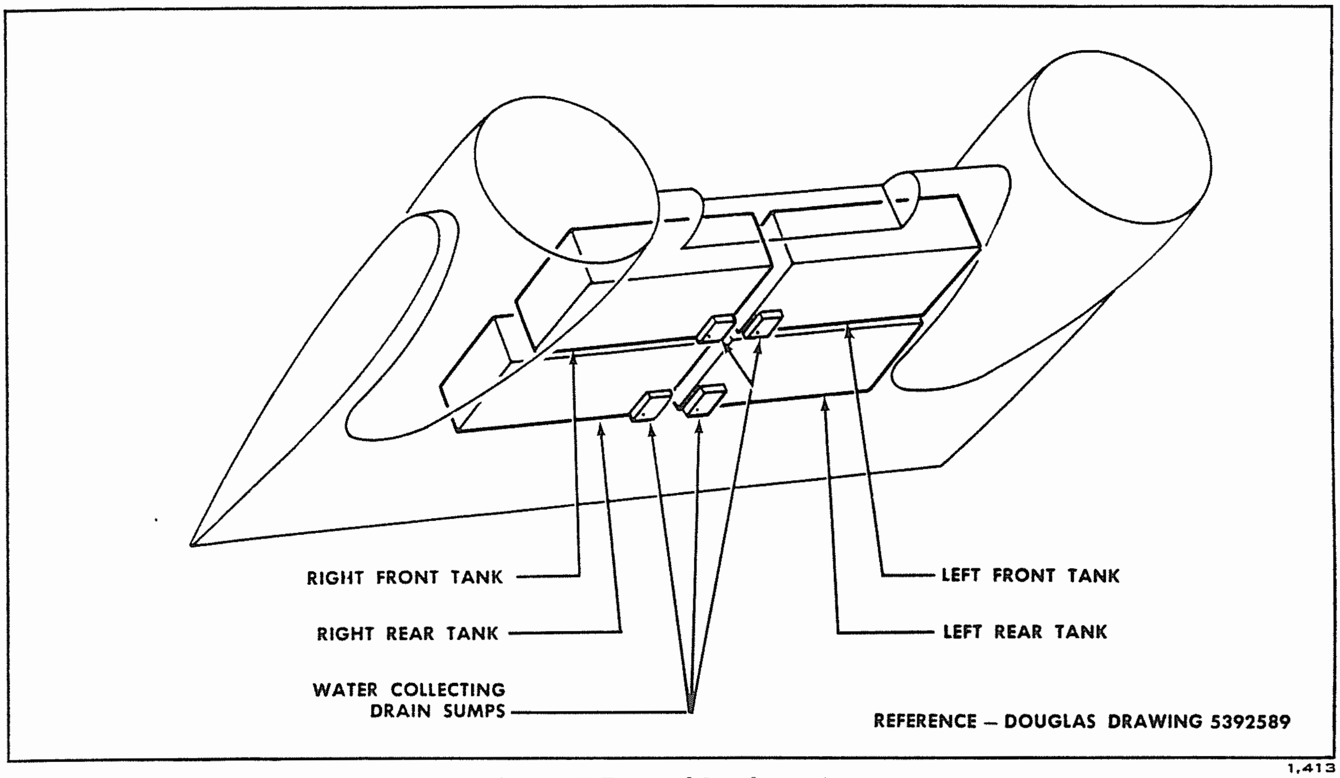


Figure 5-7. Fuel Tank Drains

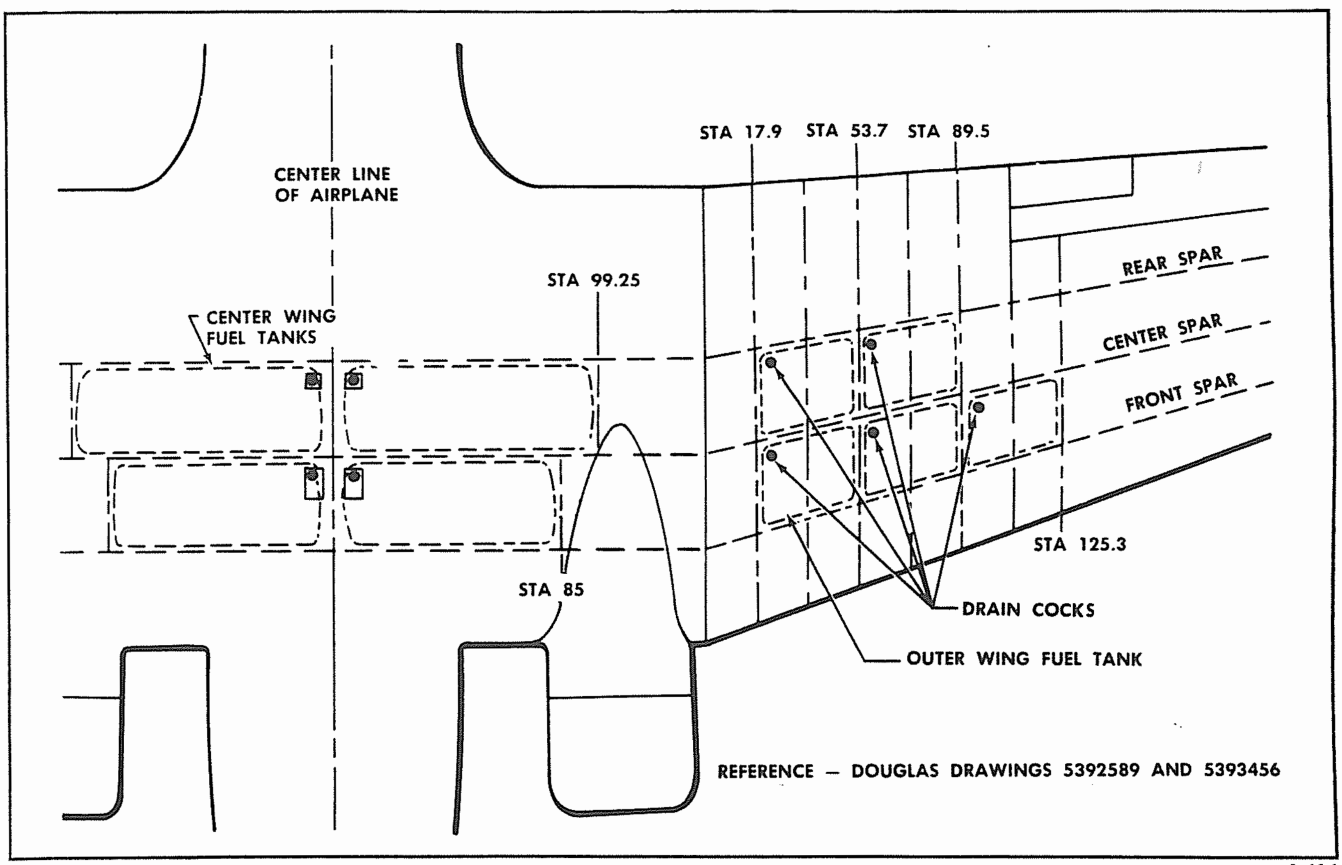


Figure 5-8. Fuel Tank Drain Points

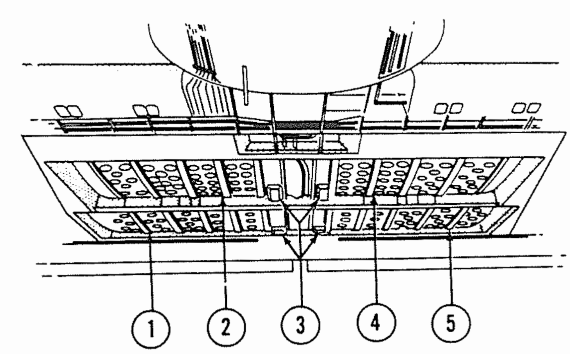
water flushing; then insert dry ice (CO_2) in the tank in the proportion of 1 pound to each 25 US. (20.81 Imp.) gallon capacity of the tank. One-half of the dry ice should be crushed in small pieces (1/4 to 1/2 inch in size) to obtain quick evaporation and the remainder should be in pieces of 1 to 2 inches in size, to obtain a slower rate of evaporation.

d. After the dry ice has been in the tank approximately 15 minutes, or until the gas can be observed flowing out of the top of the tank (similar to visible heat waves rising from a smokeless fire or heated surface), it is safe to begin the tank repair. If the repair is extensive, that is, if operations will be continued longer than 1/2 hour, it will be necessary to add dry ice in 1- to 2-inch pieces in the proportion of 1/2 pound to each 25 US. (20.81 Imp.) gallon capacity of the tank for every half hour of work operations to be continued on the tank. This method operates on the principle of reducing the oxygen content of the container to a point where combustion is impossible, in the same manner that combustion is prevented by the use of CO_2 fire extinguishers. It is important that all openings at the bottom of the tank be tightly closed, as the CO_2 gas is heavier than air and will escape if the lower openings are not tightly closed. It is also important, because the refrigerating effect of the dry ice might cause cracking of the metal ahead of the weld, to conduct all welding operations as far away as possible from the supply of unmelted dry ice in the container. For example, if the bottom of the tank is to be repaired, the tank should be inverted so that the bottom will become the upper surface.

e. If steam cleaning facilities are not available and dry ice cannot be obtained, continue the flushing of the tank with hot water for a minimum period of one hour; then thoroughly dry the interior of the tank with compressed air. This is not a positive preparation method and should not be used unless absolutely necessary.

f. When the exterior of the tank is to be cleaned with paint remover or other combustible solvents, do this cleaning prior to the flushing, steaming, or inserting the dry ice into the interior of the tank. Do the repair work as soon as possible after the tank has been cleaned and dried, except in the case of the use of dry ice. Take every precaution, regardless of the safety methods used, to remove all combustible material and fumes from the tank prior to the application of heat, to accomplish the repair.

g. When tank repairs can be accomplished by soldering with a soldering iron, not flame, it will not be necessary to steam clean the tank, provided that all fuel or oil has been drained from the tank and caution is used not to overheat the soldering irons. Do not heat the irons to a degree where they will cause particles of dust to become white hot, as this will be sufficient heat to ignite the explosive mixture in the tank.



1. Right Rear Tank
2. Right Front Tank
3. Water Collecting Sumps
4. Left Front Tank
5. Left Rear Tank

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Figure 5-9. Center Wing Fuel Tanks

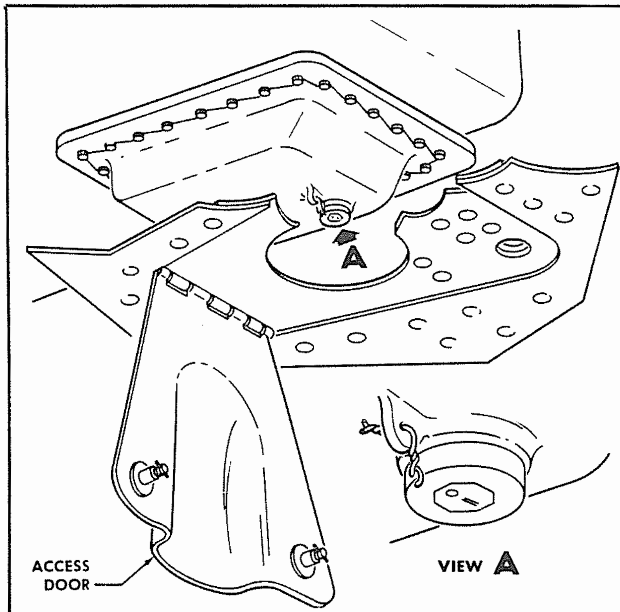

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Figure 5-10. Center Wing Fuel Tank Drain Cock

h. Deep scratches and dents that can not be removed by burnishing should be repaired by welding. Do not weld tanks that have contained fuel near any combustible materials nor in any building containing such materials. After any repair to a fuel tank, make a pressure check in a testing jig, applying 3- to 3½-psi pressure. Cover the outside of the repaired area with

soapsuds and watch for leaks. If the pressure drops, test the entire surface of the tank.

CAUTION

Three and one-half psi pressure is enough to cause some bulging of the tank. At the time the tank is to be tested at this pressure, the tank should be placed in a wooden jig. The jig will support the surfaces of the tank while it is being used.

i. Corrosion is caused by impurities and water in the bottom of the tank. These impurities may come from several sources, such as storage containers, screens, tank truck fittings, supply pipes, etc. The following procedure should be performed when corrosion has taken place:

Remove.

Clean with steam.

Clean with acid (nitric acid solution).

Clean with sodium and chromate (hot steam solution of sodium chromate, 30 to 35 per cent by weight).

Keep the tank at least 1/3 full for 1 hour.

Flush with fuel and allow to dry.

When corrosion has taken place to such an extent that the tank bottom becomes weak or pitted, install a new bottom of 3S½H aluminum-alloy sheet treated with No. 205 Alumalite process.

Apply potassium chromate process to the entire tank to remove all traces of corrosion.

Replace any damaged tank strap rubber bushings.

5-37. INSTALLATION OF CENTER WING FUEL TANKS. Reverse the removal procedure, with the following additional steps:

a. Check all seams in the tank bay and corrugation, and, if any of the black sealing compound is cracked or loose, reseal with du Pont 5115 Fairprene calking compound or equivalent.

b. When installing the fuel tanks, a little soap on the strap pads will make the tanks slide in place more easily.

c. Put the inboard end of each tank in place first. Have one man on top of the wing to connect the carburetor vapor return pipe to the tank fitting. Raise the entire tank into the tank bay; then drop the inboard end of the tank slightly and push the carburetor vapor return pipe down to meet the fitting and connect before the tank is in place.

Note

It is important that the carburetor vapor return pipe be connected before the tank is in place, as it will be difficult to make this connection after the tank is completely in place.

5-38. OUTER WING FUEL TANKS. *(See figure 5-11.)* The outer wing fuel tanks are constructed of rubber-impregnated fabric and are of the bladder-type cell. Five of these cells are located in each outer wing, three between the front and center spars between wing stations 17.9 and 125.3 and two between the center and rear spars between wing stations 17.9 and 89.5. All five cells in each outer wing are connected together and have a total fuel capacity in excess of 400 gallons for each wing. The fuel cells are not interchangeable, due to the cross-section and taper of the wing and due to the fitting locations in the cells. Each one of the cells is equipped with a sump drain cock on the bottom aft inboard end of the cell, which is accessible through the bottom wing skin. Two electric Liquidometer units are installed in the top of each set of five cells; one unit is installed in the top of the rear inboard cell and one unit in the top of the front outboard cell. Each outer wing set of Liquidometers is connected to a separate indicator located on the instrument panel. A filler neck and fuel strainer are installed in the top of the front center cell, with an overflow pipe from the filler neck routed between the front center and inboard cells and overboard through the bottom wing skin. The cells are held in place in the outer wing by the various fittings and male latch-type hangers on the bladders, which are attached to female latch-type hangers on the wing structure. Vent interconnectors are installed between each cell and the adjoining cell. A single vent pipe is routed from the upper outboard side of the front outboard cell to a vent chamber located in the top of the wing aft of the front spar between wing station 152 and station 179. The vent pipe is then routed overboard from the vent chamber through another pipe terminating at the bottom wing skin aft of the rear spar at wing station 216.

5-39. MINOR REPAIR AND REPLACEMENT OF OUTER WING FUEL TANKS.

5-40. OUTER SURFACE REPAIR OF OUTER WING FUEL TANKS. *(See figure 5-12.)* Do not attempt to repair a cell if the radius of the cell or the fitting area has been damaged; the cell should be returned to the factory for repair. If a cell has a damaged area which is larger than two inches, or if a cell is blistered or the plies have been torn apart, the cell should be returned to the factory. Fuel cells should be worked on in a dry place, under low humidity con-

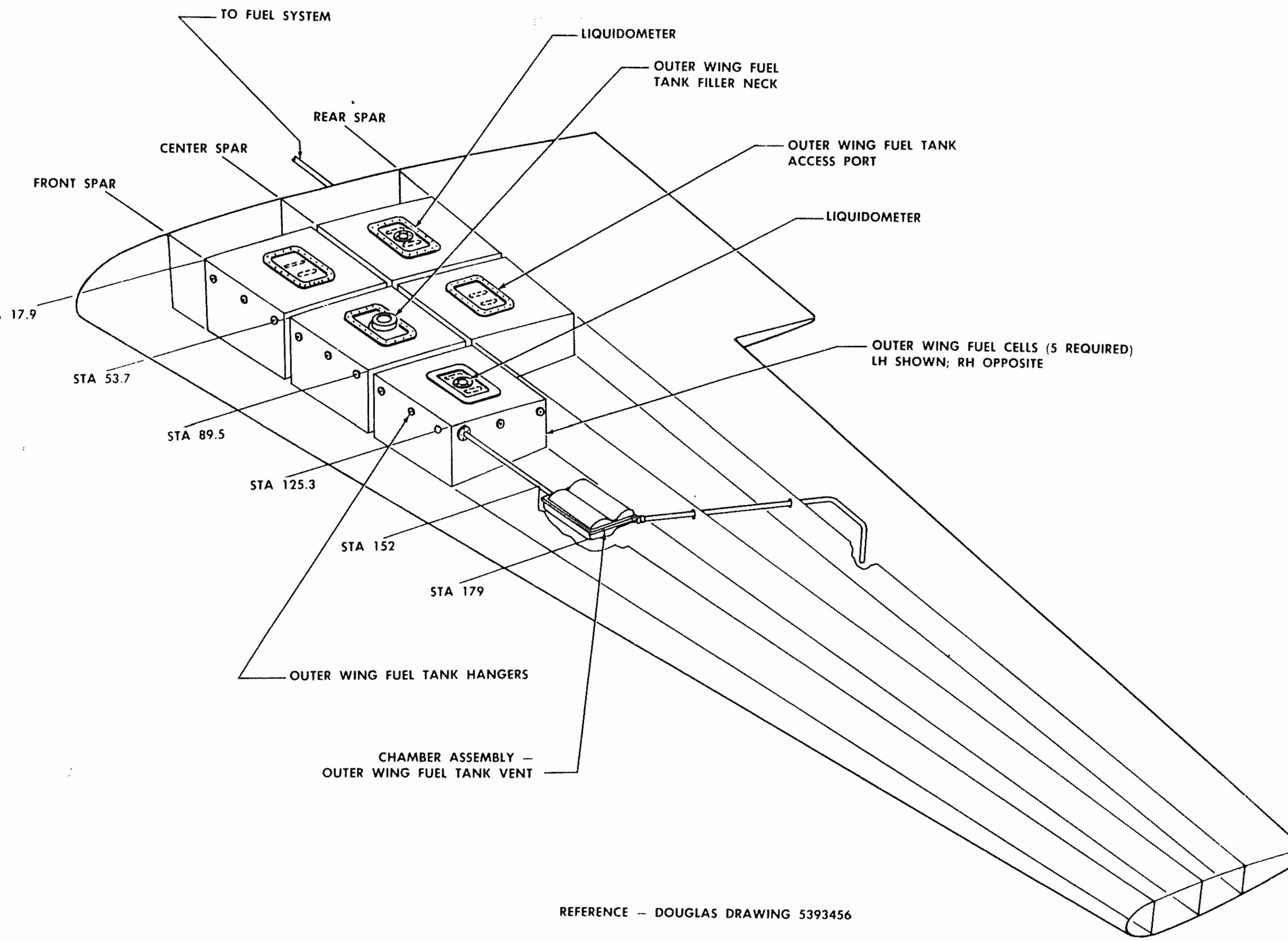

Figure 5-11. Outer Wing Bladder-Type Fuel Tanks

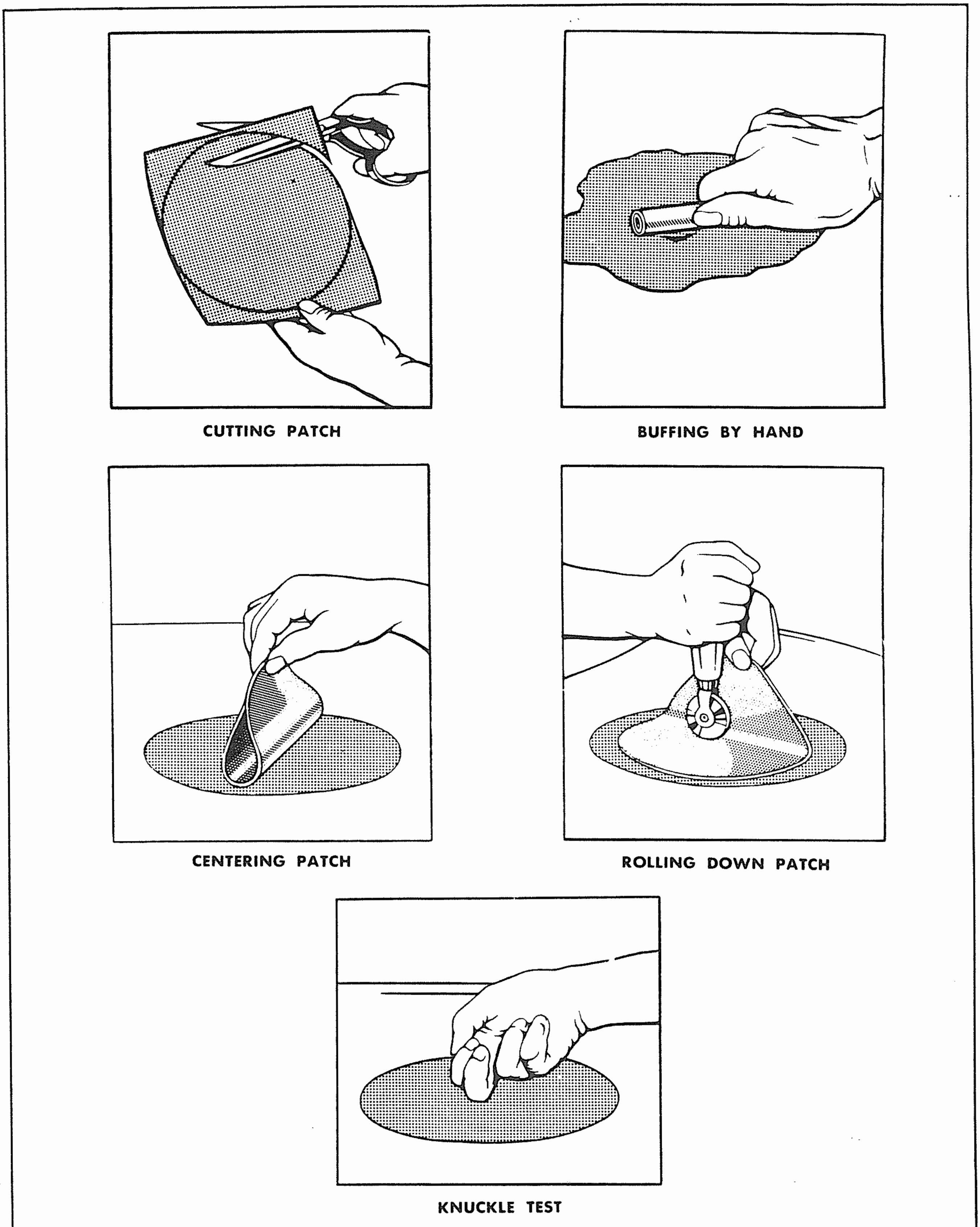

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Figure 5-12. Repairing Bladder-Type Fuel Cells

ditions. High humidity combined with a low temperature results in water condensation in cemented areas.

CAUTION

Before inspection or repair of any cell-type tank, clear cell of fuel vapor by an air blast. DO NOT attempt to work in the cell until it is absolutely dry. Use only vaporproof lights during inspection and repair. Before accomplishing any work in a fuel cell, empty pockets of all loose items and brush all metal filings from clothing.

a. Cut a piece of synthetic rubber-coated fabric (U S. Rubber 3136 outside repair material) large enough to cover the damage for at least two inches from the cut in any direction.

b. Hand-buff this material lightly and thoroughly with No. 0 sandpaper, and then wash it with U. S. Rubber 3339 methyl ethyl ketone solution to remove all of the buffing dust.

c. Apply a coat of U S. Rubber 3230 black rubber cement to the buffed side of the patch and allow it to dry for 10 to 15 minutes. Then apply a second coat, which also must be allowed to dry 10 to 15 minutes.

d. Hand-buff the damaged area of the cell lightly with No. 0 fine sandpaper, and then wash it with methyl ethyl ketone solution to remove the buffing dust *(see figure 5-12)*.

e. Apply two coats of black rubber cement to the buffed cell area, allowing each coat to dry for 10 to 15 minutes. Check for tackiness of the cemented area by tapping lightly with the knuckles. Freshen cemented patch and the cemented area of the cell with methyl ethyl ketone solution.

f. While both the cell and the patch are still tacky, place the patch on the cemented area of the cell, as shown in figure 5-12.

g. Holding part of the patch off of the cemented area, roll or press the patch down ½ to 1 inch across at a time *(see figure 5-12)*. This is necessary to prevent air pockets between the patch and the cell surface. If a U S. Rubber repair kit is used, make certain that the roll and not the stitcher is used to roll down the patch.

h. Seal the edge of the patch with one coat of black rubber cement; the seal should be ½-inch strip all around. Allow the patch to remain undisturbed for six hours.

5-41. INNER SURFACE REPAIR OF OUTER WING FUEL TANKS. *(See figure 5-12.)* Do not attempt to repair a cell if the radius of the cell or the fitting area has been damaged. The cell should be returned to the factory for repair. If the cell has a damaged area longer than two inches, or if a cell is blistered or the plies have been torn apart, the cell should be returned to the factory.

CAUTION

Before inspection or repair of any cell-type tank, clear cell of fuel vapor by an air blast. DO NOT attempt to work in the cell until it is absolutely dry. Use only vaporproof lights during inspection and repair. Before working in a fuel cell, empty pockets of all loose items and brush all metal filings from clothing.

a. With No. 0 sandpaper, lightly and thoroughly hand-buff a piece of U S. Rubber 5063 Buna N nylon sandwich material large enough to cover the damaged area for at least two inches from the cut in any direction. Wash off the buffing dust with methyl ethyl ketone solution.

b. Apply two coats of black rubber cement to the side of the patch opposite the red fabric.

c. With No. 0 fine sandpaper, lightly and thoroughly hand-buff the cell area to be patched, and then wash off the buffing dust with methyl ethyl ketone solution *(see figure 5-12)*.

d. Apply two coats of black rubber cement to the buffed cell area and allow each coat to dry to a tacky stage, approximately 10 to 15 minutes. Freshen the cemented area of the patch and the cemented area of the cell with methyl ethyl ketone solution.

e. While the cement is still tacky, apply the patch to the cemented area of the cell, centering the patch over the cut in the cell, as shown in figure 5-12.

f. To avoid trapping air between the patch and the cell surface, hold part of the patch off of the cemented area of the cell and roll it down ½ to 1 inch across at a time *(see figure 5-12)*. Remove the red fabric from the patch, moistening it with methyl ethyl ketone solution.

g. Apply two additional coats of black rubber cement ½ inch around the edge of the patch. Allow the first coat to dry for 15 minutes before applying the second coat. Permit the second coat to dry for 12 hours or more, so that the patched area of the cell will not stick to other areas of the cell when the cell is returned to its original position.

5-42. INSPECTION AND TEST OF OUTER WING FUEL TANKS. If visual inspection of the fuel cell does not disclose a leak, make the following pressure test:

a. Plug all openings with tight-fitting plugs.

b. Spray a solution of Turco LAC 598 bubble fluid (one part fluid to six parts water, by volume) on the exterior of the cell and on all fittings.

c. With a manometer attached to the tank, apply 1/4 psi and allow the tank to settle for approximately 10 minutes. Then reapply 1/4 psi of dry air in the tank. If a test cage is available, place the cell in the test cage, and apply one psi air pressure to the cell and check for bubbles. Keep cell from direct rays of the sun and protect from wind while testing.

d. Allow a settling interval of 10 minutes, constantly maintaining pressure. If no leaks are observed, close the air valve and disconnect the air supply pipe to the manometer. The cell must hold the pressure for 15 minutes. If a pressure drop is observed, check for leaks.

5-43. REMOVAL OF AFT INBOARD OUTER WING FUEL CELL.

CAUTION

Prior to removal, the cell must be drained and sprayed or wiped with oil, preferably at least 24 hours prior to removal.

a. Drain all cells by opening the drain cocks in the bottom skin of the wing forward of the rear and center spars.

b. Remove the access plate from the wing top skin over the cell.

c. Remove the electrical disconnect plug from the Liquidometer unit in the cell cover plate.

d. Remove the cell cover assembly and Liquidometer unit by removing the 36 bolts around the edge of the cover.

e. Remove the access door on the bottom wing skin forward of the rear spar and inboard of wing station 17.9.

f. Disconnect the fuel pipe from the aft inboard fuel cell fitting and strainer assembly on the inboard side of rib station 17.9.

g. Remove the finger strainer assembly and seal from the inboard side of rib station 17.9 by removing the attach bolt and washers.

h. Remove the drain cock and four attaching bolts in the wing skin and bottom of the cell.

i. Remove the six vent interconnector bolts, washers and the seal from the inside wall of the cell.

j. Remove the access plate in the top wing skin over the aft outboard cell.

k. Remove the aft outboard cell cover plate by removing the 36 bolts around the edge of the cover.

l. Remove the cell interconnector bolts from the inboard side of the aft outboard cell.

m. Unlatch the male latch-type bladder hangers from the female latch-type bladder hangers on the center spar, rear spar, wing rib station 53.7 and wing rib station 17.9.

n. Remove the four supporting bolts from the corners of the aft inboard cell top opening and the four clips on the bottom side of the upper wing panel. Collapse the cell to the bottom and fold.

o. Remove the cell through the access hole in the top of the wing.

Note

Do not use force or pry the cells through the access holes.

5-44. INSTALLATION OF AFT INBOARD OUTER WING FUEL CELL. For information on the installation of the aft inboard outer wing fuel cell, see paragraph 5-53.

5-45. REMOVAL OF AFT OUTBOARD OUTER WING FUEL CELL.

CAUTION

Prior to removal, the cell must be drained and sprayed or wiped with oil, preferably at least 24 hours prior to removal.

a. Drain the fuel from all cells by opening the drain cocks in the bottom wing skin forward of the center and rear spars.

b. Remove the access plate from the wing top skin over the cell.

c. Remove the cell cover assembly from the top of the fuel cell by removing the 36 bolts around the edge of the cover.

d. Remove the access plate from the wing top skin over the inboard aft bladder-type fuel cell.

e. Remove the electrical disconnect plug from the Liquidometer unit in the inboard aft fuel cell cover assembly.

f. Remove the inboard aft fuel cell cover assembly with the Liquidometer unit by removing the 36 bolts around the edge of the cover assembly.

g. Remove the six vent interconnector bolts, washers and the seal from the inside wall of the aft inboard fuel cell.

h. Remove the forward center fuel cell filler neck and strainer assembly by removing the access cover over the fuel cell, disconnecting the drain hose connector, and removing the bolts and washers around the outside edge of the fuel cell and filler neck assembly.

i. Remove the six vent interconnector bolts, washers and the seal from the aft side wall of the forward center fuel cell.

j. Remove the cell interconnector bolts from the bottom inboard side of the aft outboard fuel cell.

k. Remove the drain cock and four attaching bolts in the wing skin and bottom of the fuel cell, outboard of wing station 53.7 and forward of the rear spar.

l. Remove the four supporting bolts from the corners of the aft outboard cell top opening and remove the clips on the bottom of the upper wing panel.

m. Unlatch the male latch-type bladder hangers from the female latch-type bladder hangers on the center spar, rear spar outboard side of wing rib station 53.7 and inboard side of wing rib station 89.5.

n. Collapse and fold the cell to the bottom of the wing to remove the cell through the access opening in the top of the wing.

CAUTION

Do not use force when removing the fuel cell through the top of the wing. Serious damage to the cell can result by chafing and scratching against the wing structure if force is used.

5-46. INSTALLATION OF AFT OUTBOARD OUTER WING FUEL CELL. For information on the installation of the aft outboard outer wing fuel cell, see paragraph 5-53.

5-47. REMOVAL OF FORWARD OUTBOARD OUTER WING FUEL CELL.

CAUTION

Prior to removal, the cell must be drained and sprayed or wiped with oil, preferably at least 24 hours prior to removal.

a. Drain the fuel from all cells by opening the drain cocks in the bottom of the wing skin forward of the center and rear spars.

b. Remove the access plate from the wing top skin over the forward outboard cell.

c. Remove the electrical disconnect plug from the Liquidometer unit on the cell cover of the forward outboard fuel cell.

d. Remove the forward outboard fuel cell cover assembly with the Liquidometer unit by removing the 36 bolts around the edge of the cover assembly.

e. Remove the access cover on the wing top skin between the front and center spars and between wing station 125.3 and station 152.

f. Disconnect the vent pipe from the vent connect fitting on the top outboard side of wing rib station 125.3 aft of the front spar.

g. Remove the vent connect fitting from the outboard side of wing rib station 125.3 aft of the front spar by removing six bolts, six washers and a seal.

h. Remove the six vent interconnect bolts, washers and seal from the top of the inboard wall of the forward outboard fuel cell.

i. Remove the 12 cell interconnect bolts, washers, and the seal from the bottom inboard wall of the forward outboard fuel cell.

j. Remove the drain cock and four attaching bolts in the wing skin on the bottom of the fuel cell outboard of wing station 89.5 and forward of the center spar.

k. Remove the four supporting bolts from the corners of the forward outboard fuel cell opening and remove the clips on the bottom of the upper wing assembly.

l. Unlatch the male latch-type bladder hangers from the female latch-type bladder hangers on the center spar, front spar, and wing rib station 125.3 and station 89.5.

m. Collapse and fold the cell to the bottom of the wing to remove the cell from the access opening in the top of the wing.

CAUTION

Do not use force when removing the cell through the top opening of the wing. Serious damage to the cell can result by chafing and scratching against the wing structure if force is used.

5-48. INSTALLATION OF FORWARD OUTBOARD OUTER WING FUEL CELL. For information on the installation of the forward outboard outer wing fuel tank, see paragraph 5-53.

5-49. REMOVAL OF FORWARD CENTER FUEL CELL.

CAUTION

Prior to removal, the cell must be drained and sprayed or wiped with oil, preferably at least 24 hours prior to removal.

a. Drain the fuel from all cells by opening the drain cocks in the bottom of the wing skin forward of the center and rear spars.

b. Remove the forward center fuel cell filler neck and strainer assembly by removing the access cover over the fuel cell, disconnecting the drain hose connector, and removing the bolts and washers around the outside edge of the fuel cell cover and filler neck assembly.

c. Remove the forward outboard fuel cell access cover.

d. Disconnect the electrical disconnect plug from the Liquidometer unit on the top of the forward outboard fuel cell.

e. Remove the forward outboard fuel cell cover assembly and Liquidometer unit by removing the 36 bolts and washers around the edge of the cover assembly.

f. Remove the six vent interconnect bolts, washers, and the seal from the top inboard wall of the forward outboard fuel cell.

g. Remove the 12 fuel cell interconnect bolts, washers, and the seal from the bottom inboard wall of the forward outboard fuel cell.

h. Remove the six vent interconnect bolts, washers and the seal from the top of the aft wall of the forward center fuel cell.

i. Remove six vent interconnect bolts, washers, and the seal from the top inboard wall of the forward center fuel cell.

j. Remove the 12 fuel cell interconnect bolts, washers, and the seal from the bottom of the inboard wall of the forward center fuel cell.

k. Remove the drain cock and four attaching bolts in the bottom wing skin forward of the center spar and outboard of wing station 53.7.

l. Remove the four supporting bolts from the corners of the forward center fuel cell top opening and remove the clips on the bottom of the upper wing panel.

m. Unlatch the male latch-type bladder hangers from the female latch-type bladder hangers on the center spar, front spar, wing rib station 53.7 and wing rib station 89.5.

n. Collapse and fold the cell to the bottom of the wing to remove the cell through the access opening in the top of the wing.

CAUTION

Do not use force when removing the fuel cell through the top of the wing. Serious damage to the cell can result by chafing and scratching against the wing structure if force is used.

o. To remove the filler neck overflow pipe from the inboard side of rib station 53.7, the forward center and forward inboard fuel cells must be collapsed, as outlined in step n, preceding, to gain access to the two clips on the inboard side of wing rib station 53.7 and also to the clip on the bottom of the top wing panel between the front and center spars outboard of wing rib station 53.7.

5-50. INSTALLATION OF FORWARD CENTER FUEL CELL. Reverse the removal procedure.

5-51. REMOVAL OF FORWARD INBOARD OUTER WING FUEL CELL.

CAUTION

Prior to removal, the cell must be drained and sprayed or wiped with oil, preferably at least 24 hours prior to removal.

a. Drain the fuel from all cells by opening the drain cocks in the bottom of the wing skin forward of the center and rear spars.

b. Remove the access plate from the wing top skin over the forward inboard fuel cell.

c. Remove the forward inboard fuel cell cover assembly by removing the 36 bolts and washers around the edge of the cover assembly.

d. Remove the forward center fuel cell filler neck and strainer assembly by removing the access cover over the fuel cell, disconnecting the drain hose connector, and removing the bolts and washers around the outside edge of the fuel cell cover and filler neck assembly.

e. Remove the six vent interconnect bolts, washers, and the seal from the top inboard wall of the forward center fuel cell.

f. Remove the 12 fuel cell interconnect bolts, washers and the seal from the bottom inboard wall of the forward center fuel cell.

g. Remove the aft inboard fuel cell access cover in the top wing panel.

h. Disconnect the electrical disconnect plug from the Liquidometer unit on the top of the aft inboard fuel cell.

i. Remove the aft inboard fuel cell cover assembly and Liquidometer unit by removing the 36 bolts and washers around the edge of the cover assembly.

j. Remove the 12 fuel cell interconnect bolts, washers and the seal from the bottom forward wall of the aft inboard fuel cell.

k. Remove the drain cock and four attaching bolts in the bottom wing skin forward of the center spar and outboard of wing station 17.9.

l. Remove the four supporting bolts from the corners of the forward inboard fuel cell top opening and remove the clips on the bottom of the upper wing panel.

m. Unlatch the male latch-type bladder hangers from the female latch-type bladder hangers on the center spar, front spar, wing rib station 17.9 and wing rib station 53.7.

n. Collapse and fold the fuel cell to the bottom of the wing to remove the cell through the access opening in the top of the wing.

CAUTION

Do not use force when removing the fuel cell through the top of the wing. Serious damage to the cell can result by chafing and scratching against the wing structure if force is used.

5-52. INSTALLATION OF FORWARD INBOARD OUTER WING FUEL CELL. For information on the installation of the forward inboard outer wing fuel cell, see paragraph 5-53.

5-53. INSTALLATION OF OUTER WING FUEL CELLS. Reverse the removal procedure of the applicable fuel cell being installed and incorporate the following installation instructions.

a. Check all rivet heads and seams in the wing fuel cell area to be sure that $\frac{1}{32}$-inch-thick felt covers the rivet heads and the seams.

b. Dust powdered talc on the outside of the fuel cell before installing the cell in the wing.

c. When reinstalling the seals on the vent interconnects and fuel interconnects, make certain that the seal is not damaged.

d. Apply a film of RL-3700 compound, not to exceed a thickness of approximately 0.010 inch on both sides of the seals, in the filler neck, drain fittings and pipe connections.

e. Torque all vent and fuel interconnect bolts to 30 to 40 inch-pounds. Recheck the torque in 30 minutes.

f. Torque all cell cover assembly bolts to 40 to 50 inch-pounds. Check the torque after 30 minutes.

5-54. FUEL TANK SELECTOR VALVES. *(See figures 5-13, 5-14, and 5-15.)* Four cable-operated fuel tank selector valves are located near the bottom of the fuselage, between fuselage station 191 and the front spar. The two fuel selector valves located on the forward face of the front spar are four-way valves and are used to select fuel from the center wing fuel tanks. The four-way center wing fuel tank selector valves are controlled by means of handles and dial controls on the top of the control pedestal in the cockpit. The left fuel selector valve supplies fuel to the left engine and the right fuel selector valve supplies fuel to the right engine. Fuel may be supplied from any of the four center wing fuel tanks to either engine by use of these valves. When fuel is not drawn from the center wing fuel tanks, the valves can be placed in the OFF position. The two fuel selector valves located in the center wing section between station 191 and 203 are three-position fuel selector valves and are used to select fuel from the outer wing fuel tanks. The three-position outer wing fuel tank selector valves are controlled by handles and dial controls located on the aft side of the control pedestal in the flight compartment. By means of these two valves, fuel can be selected from either the left or right outer wing fuel tanks for either engine. When fuel is not required from the wing fuel tanks, the valves should be placed in the OFF position. The left wing tank selector valve ordinarily supplies fuel to the left engine and the right wing tank fuel selector valve supplies fuel to the right engine.

5-55. REMOVAL OF FUEL TANK SELECTOR VALVES.

a. Remove the access doors located near the front of the center wing.

b. Remove the safety wire and disconnect the tank selector valve control cables at the turnbuckles.

c. Disconnect and cap all the fuel pipes at the selector valves.

d. Disconnect the attaching bolts that hold the valve to the support bracket and remove the selector valve.

5-56. MINOR REPAIR AND REPLACEMENT OF FUEL TANK SELECTOR VALVES.

a. If a selector valve is leaking, check for worn seals around the valve ports and around the cover plate. If parts are readily available, seals may be replaced. If parts are not readily available, the entire unit should be replaced with a serviceable unit.

b. Check the control cables for wear and replace if necessary.

c. Examine pulleys, pulley brackets, and selector valve support bracket for damage: replace if necessary.

5-57. INSTALLATION OF FUEL TANK SELECTOR VALVES. Reverse the removal procedure, then tension the cables in accordance with the Cable Rigging Tension Chart, figure 2-7, and safety the turnbuckles.

5-58. FUEL TANK SELECTOR VALVE CONTROLS. *(See figure 5-15.)* The fuel tank selector valve controls consist of two-way cable systems which extend from the selector valve controls on the pedestal, aft through the fuselage to the center wing tank selector valves located on the forward face of the center wing front spar, to the outer wing tank selector valves located at the leading edge of the center wing. Operation of the controls on the pedestal determines the flow of fuel from any of the center wing or outer wing tanks to the engines.

5-59. FUEL TANK SELECTOR VALVE CONTROL CABLES. *(See figure 5-15.)* The fuel tank selector valve control cables extend from the drums in the control pedestal, aft through the lower part of the fuselage, to the center wing selector valve drums located on the forward face of the center wing front spar, and to the outer wing tank selector valve drums located at the leading edge of the center wing.

5-60. REMOVAL OF FUEL TANK SELECTOR VALVE CONTROL CABLES.

a. Remove the necessary access doors on the bottom of the fuselage and on each side of the control pedestal, and the necessary floor panels in the aircraft to gain access to the fuel tank selector valve control cables for removal.

b. Disconnect the fuel tank selector valve cables at the turnbuckles at fuselage station 183. Thread the cables for removal.

c. Release the set screws on the two fuel tank selector drums in the control pedestal and the two selector drums under the pedestal.

d. Remove the necessary fairlead grommets and pulley guard pins.

e. Pull one end of each forward selector valve control cable at the terminal ends at station 183, unwinding them from the drums in the control pedestal. Pull the cables slowly to prevent chipping of the micarta pulleys. Use two men to perform this operation; one to pull the cables and the other to guide the cable ends through the pulley brackets and fairleads. Remove the cables from the aircraft.

f. Release the set screws on the four fuel tank selector valve control drums in the center wing.

g. Remove the necessary pulley guard pins.

h. Pull one end of each aft selector valve control cable at the terminal ends at station 183, unwinding them from the drums on the selector valves, and remove the cables from the aircraft.

(Continued on Page 369)

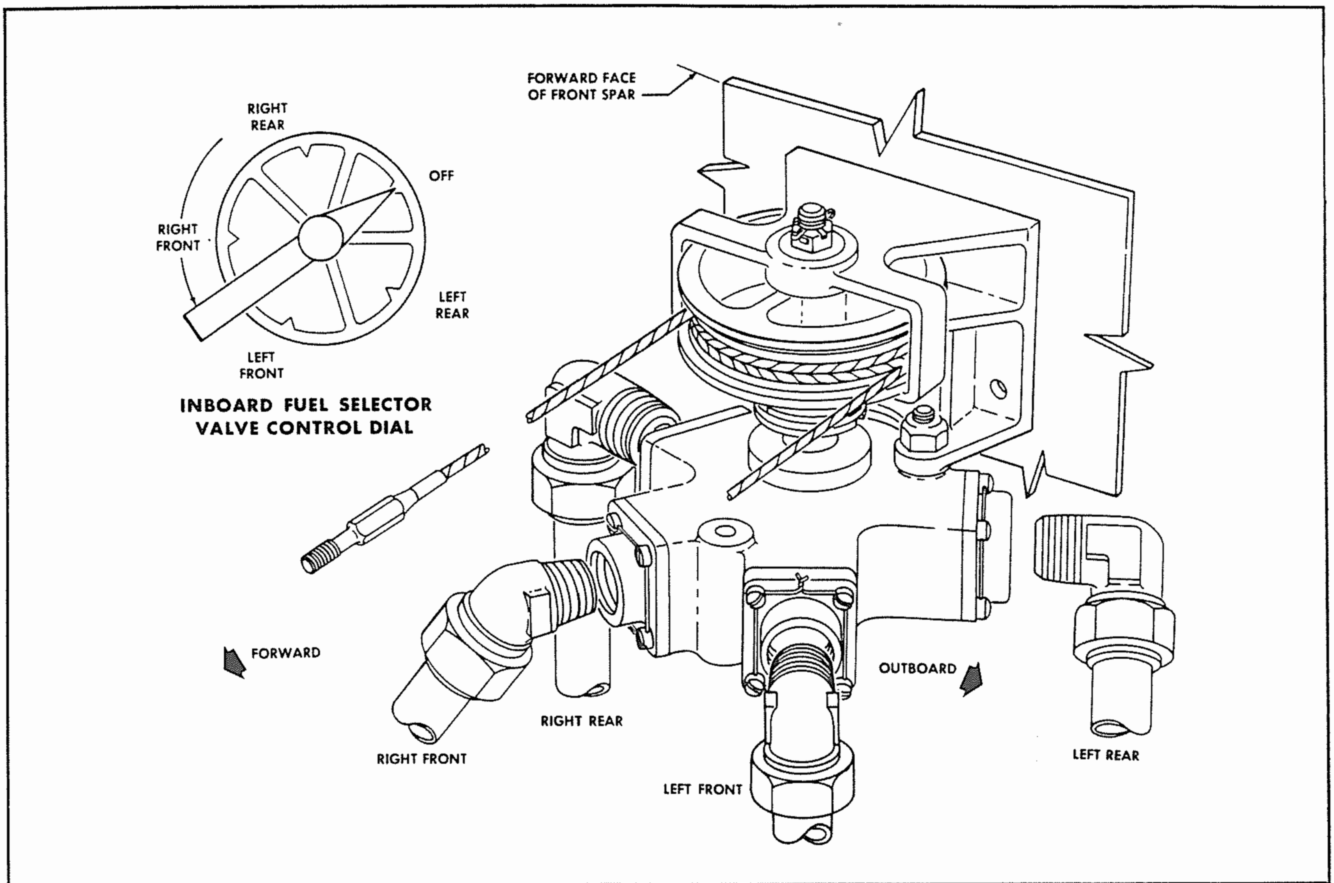

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Figure 5-13. Center Wing Fuel Tank Selector Valve

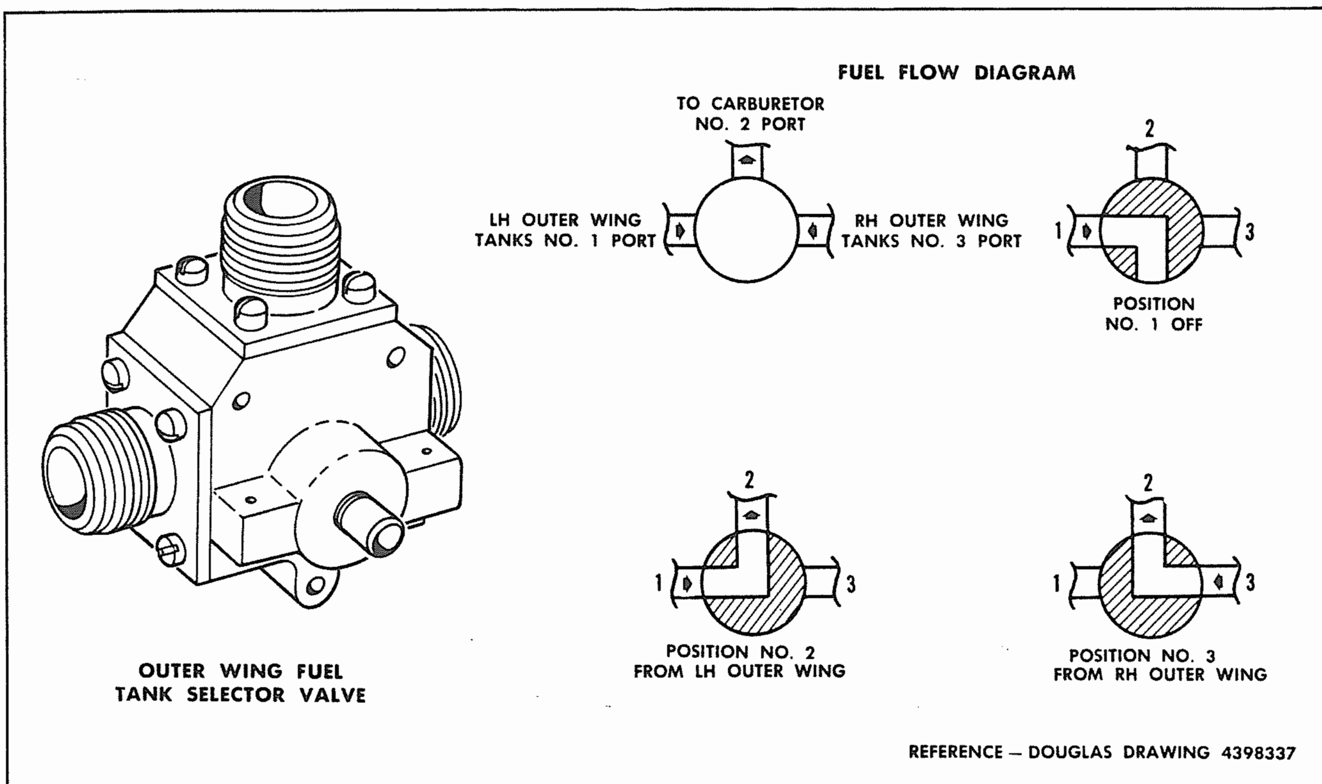

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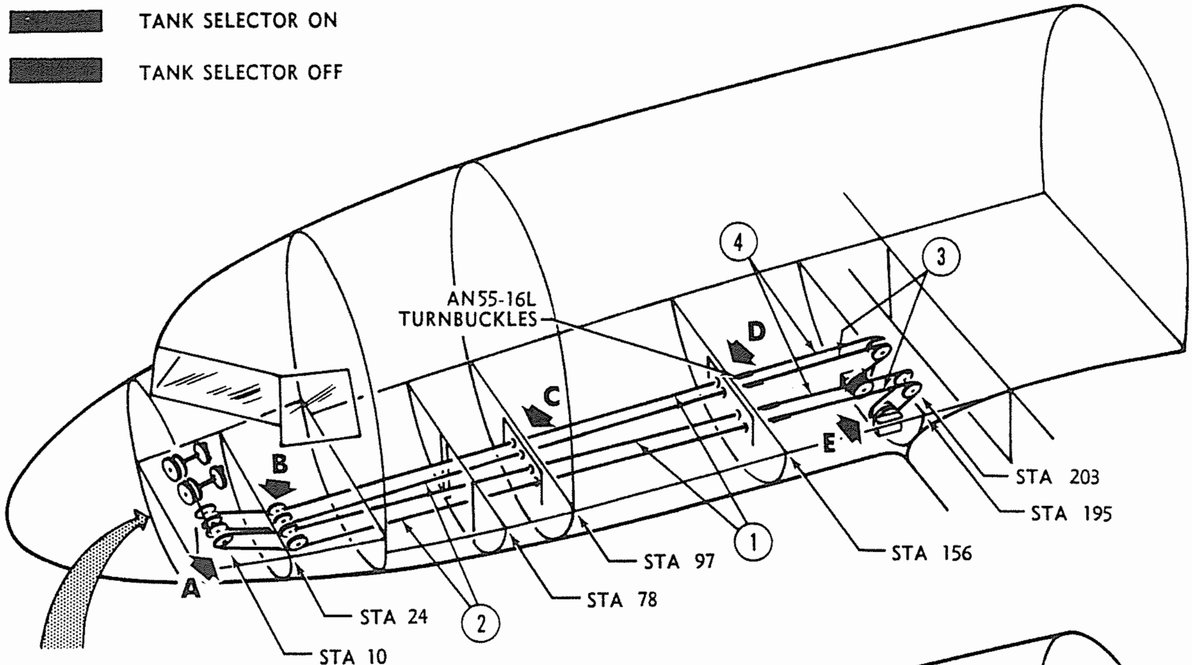
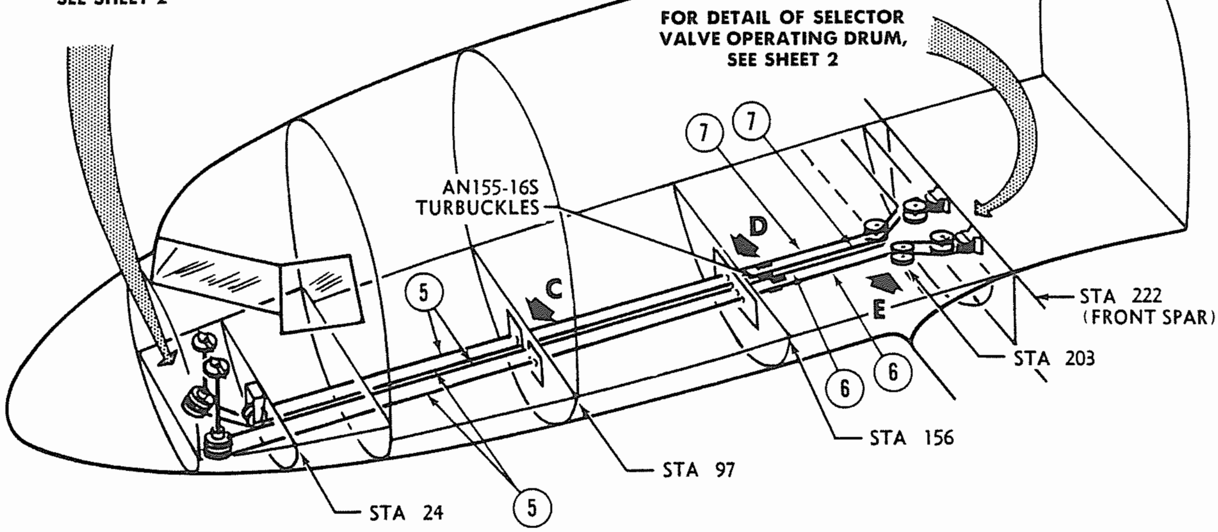
Figure 5-14. Outer Wing Fuel Tank Selector Valve



OUTER WING TANK SELECTORS



CENTER WING TANK SELECTORS

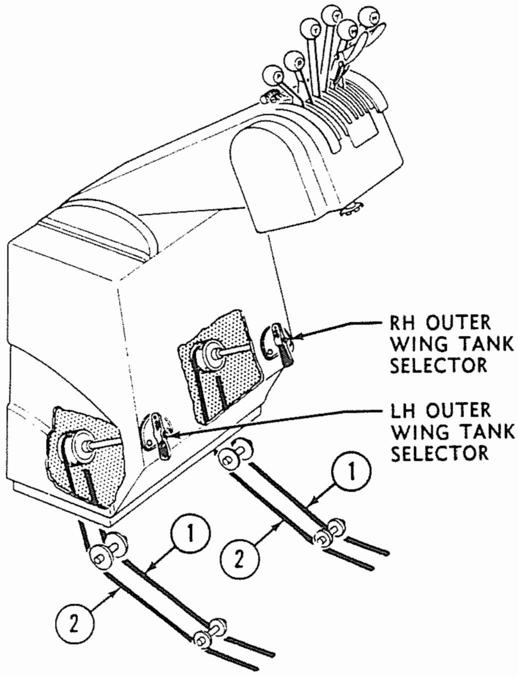
Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 7.
2. For cable assembly details, see Sheet 7.
3. For rigging and adjustment diagram, see Sheet 5.
4. Tension cables in accordance with the Cable Rigging Tension Chart, Figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart, paragraph 5-264.
6. Reference—Douglas drawings 5372517, 5392589, 5389252, and 5398557.

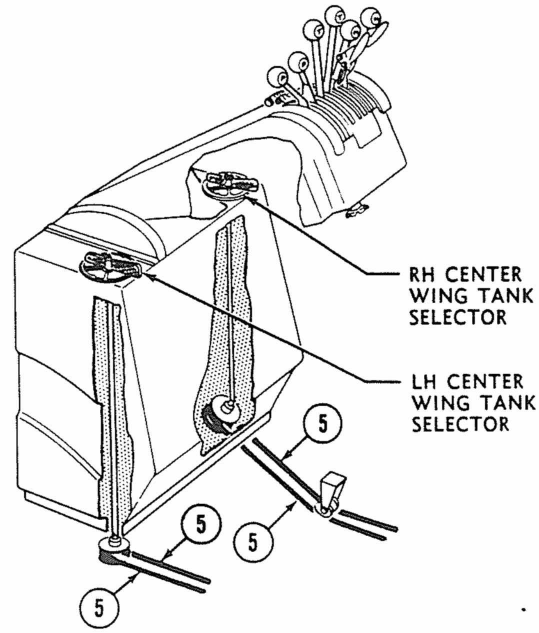
A—Pulley Brackets—Station 10, see Sheet 3
B—Pulley Brackets—Station 26, see Sheet 3
C—Fairleads—Station 97, see Sheet 3
D—Fairleads—Station 156, see Sheet 4
E—Pulley Brackets—Station 203, see Sheet 4

1,467

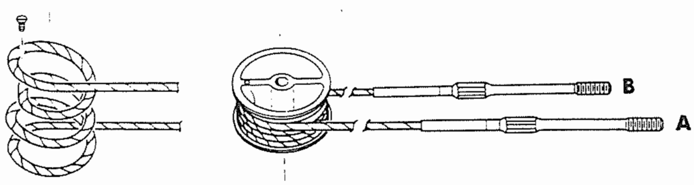
Figure 5-15 (Sheet 1 of 7 Sheets). Fuel Tank Selector Valve Control System — Key Drawing



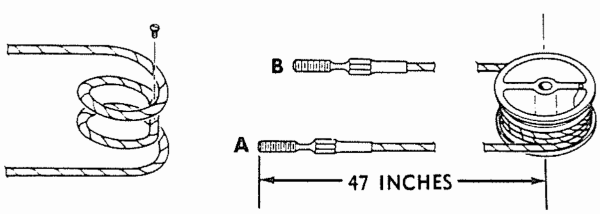
VIEW OF OUTER WING TANK CONTROLS IN PEDESTAL



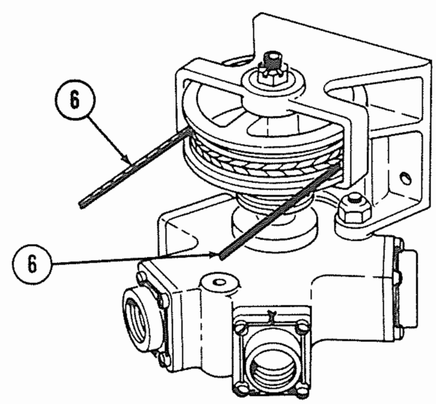
VIEW OF CENTER WING TANK CONTROLS IN PEDESTAL



CENTER WING TANK SELECTOR DRUM IN CONTROL PEDESTAL (TYPICAL)



CENTER WING TANK SELECTOR DRUM AT STATION 222 (TYPICAL)



VIEW OF LH CENTER WING TANK CABLE DRUM AND SELECTOR VALVE AT STATION 222 , FRONT SPAR (RH SIMILAR)

Notes:

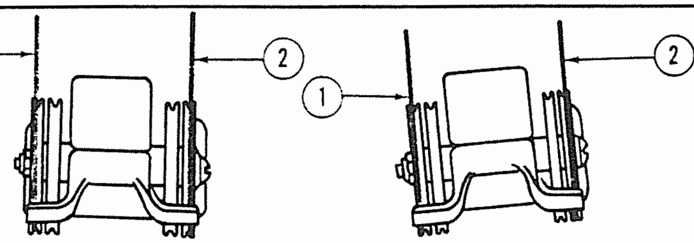
1. Wrap cable on drums from each end to center with equal number of turns on each side.
2. Wrap drum in control pedestal so that cable (A) is 18 inches longer than cable (B).
3. Wrap drum at station 222 so that cable (A) is 47 inches in length as indicated above.

TANK SELECTOR ON

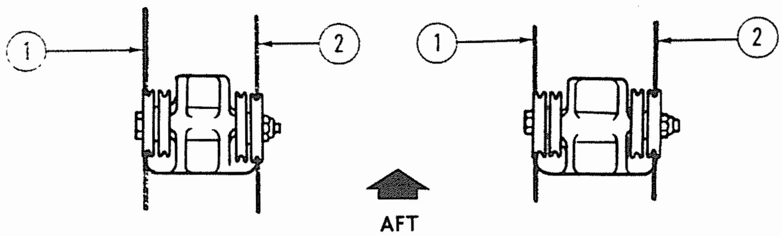
TANK SELECTOR OFF

1,468

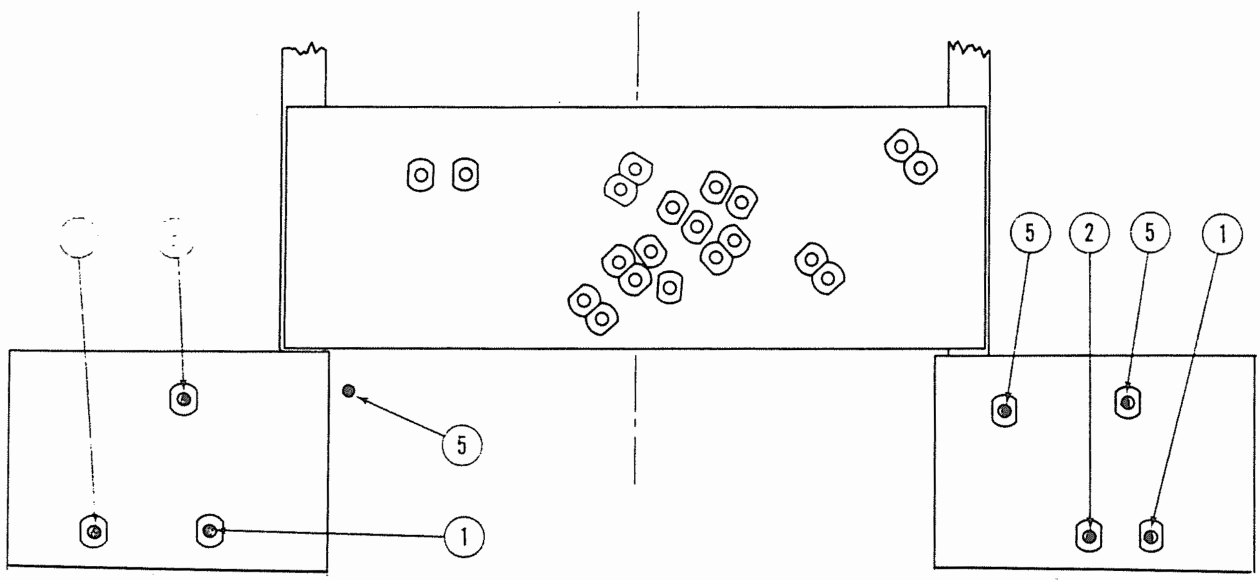
Figure 5-15 (Sheet 2 of 7 Sheets). Fuel Tank Selector Valve Control System — Control Pedestal and Center Wing Tank Valve



VIEW A — PULLEY BRACKETS
AT STATION 10
(LOOKING AFT)



VIEW B — PULLEY BRACKETS
AT STATION 24
(LOOKING DOWN)



VIEW C — FAIRLEADS
AT STATION 97
(LOOKING FORWARD)

TANK SELECTOR ON

TANK SELECTOR OFF

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Figure 5-15 (Sheet 3 of 7 Sheets). Fuel Tank Selector Valve Control System — Pulley Brackets, Stations 10 and 24; Fairleads, Station 97

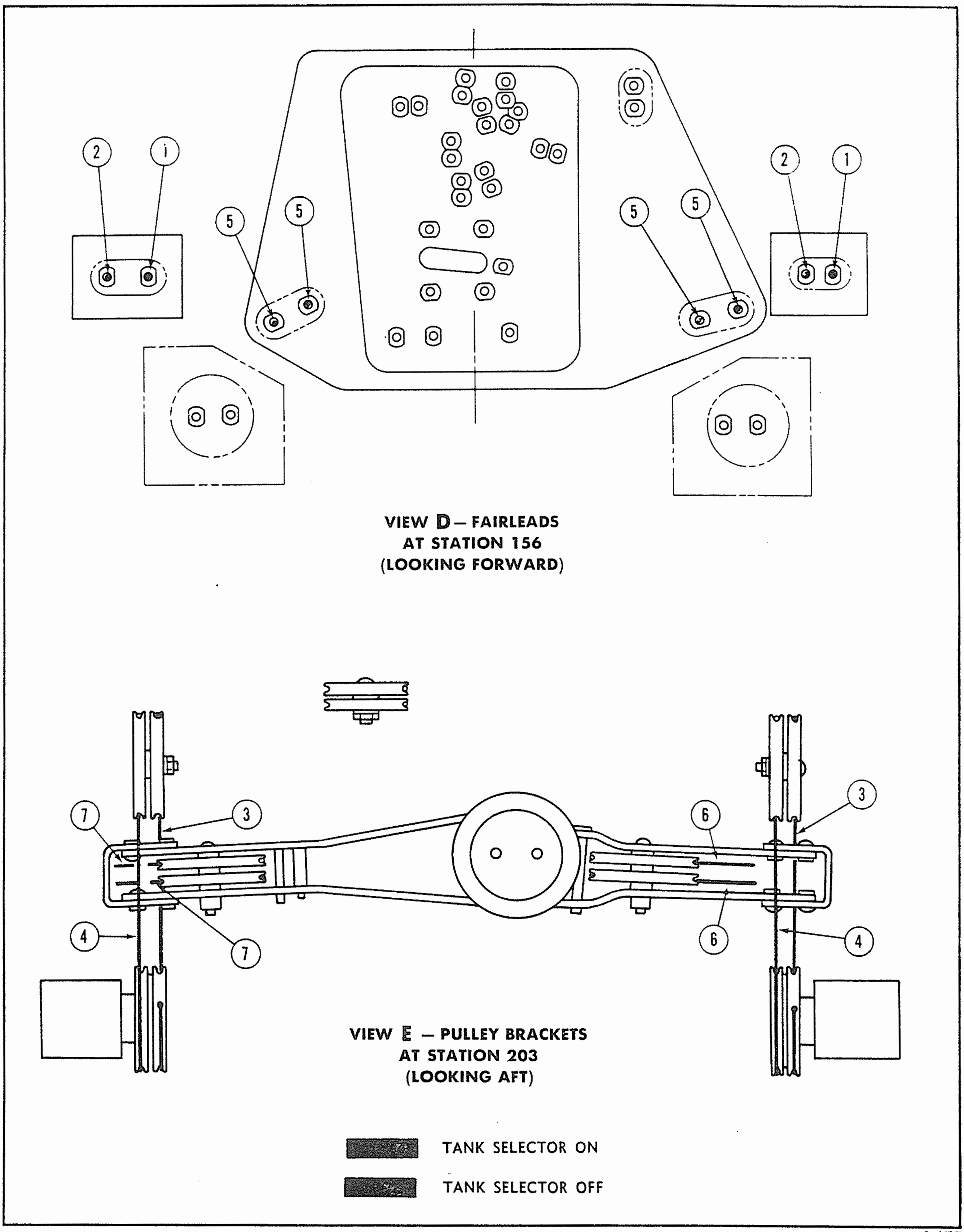
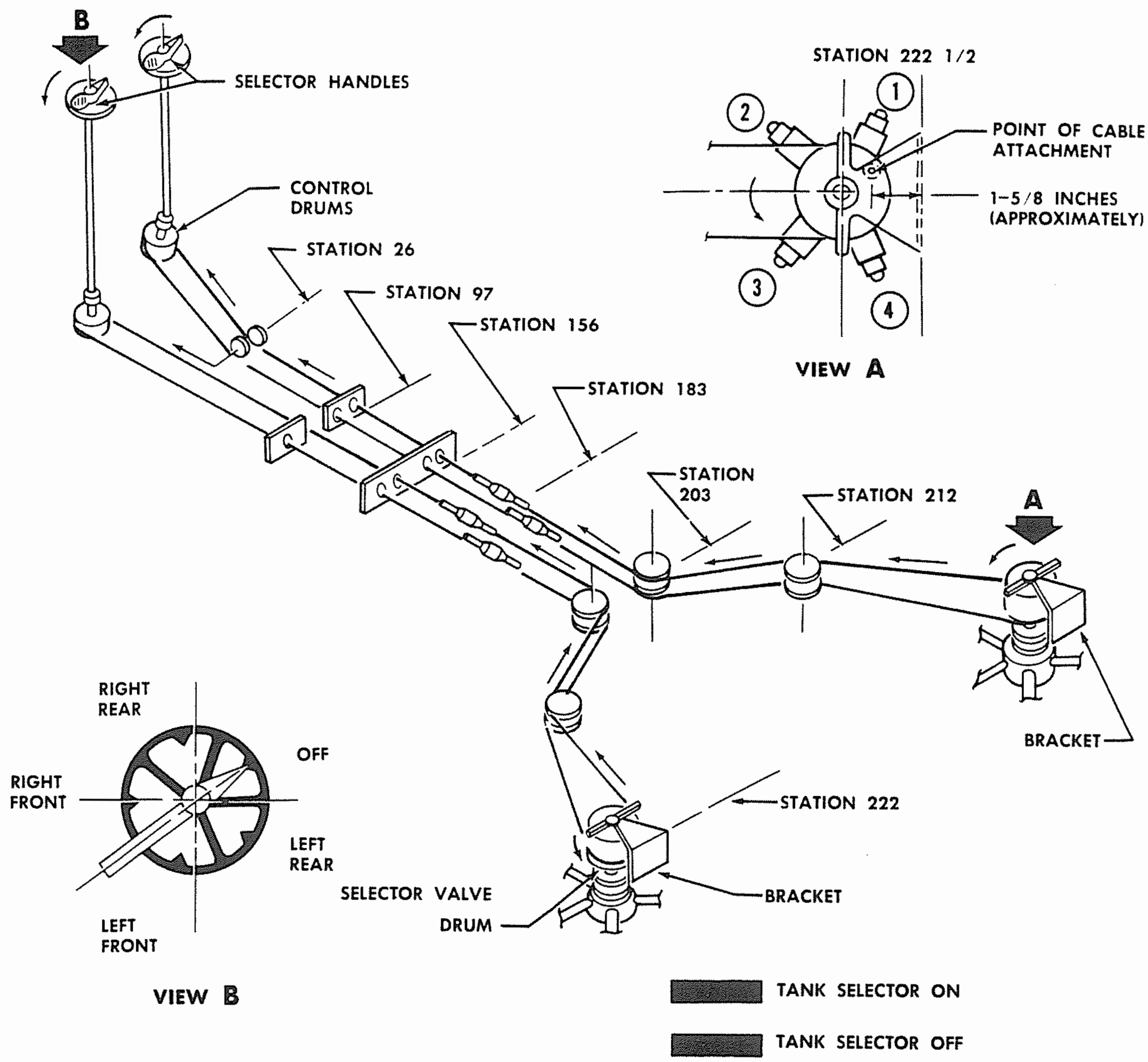


Figure 5-15 (Sheet 4 of 7 Sheets). Fuel Tank Selector Valve Control System — Fairleads, Station 156; Pulley Brackets, Station 203



CENTER WING TANK SELECTOR ADJUSTMENT PROCEDURE

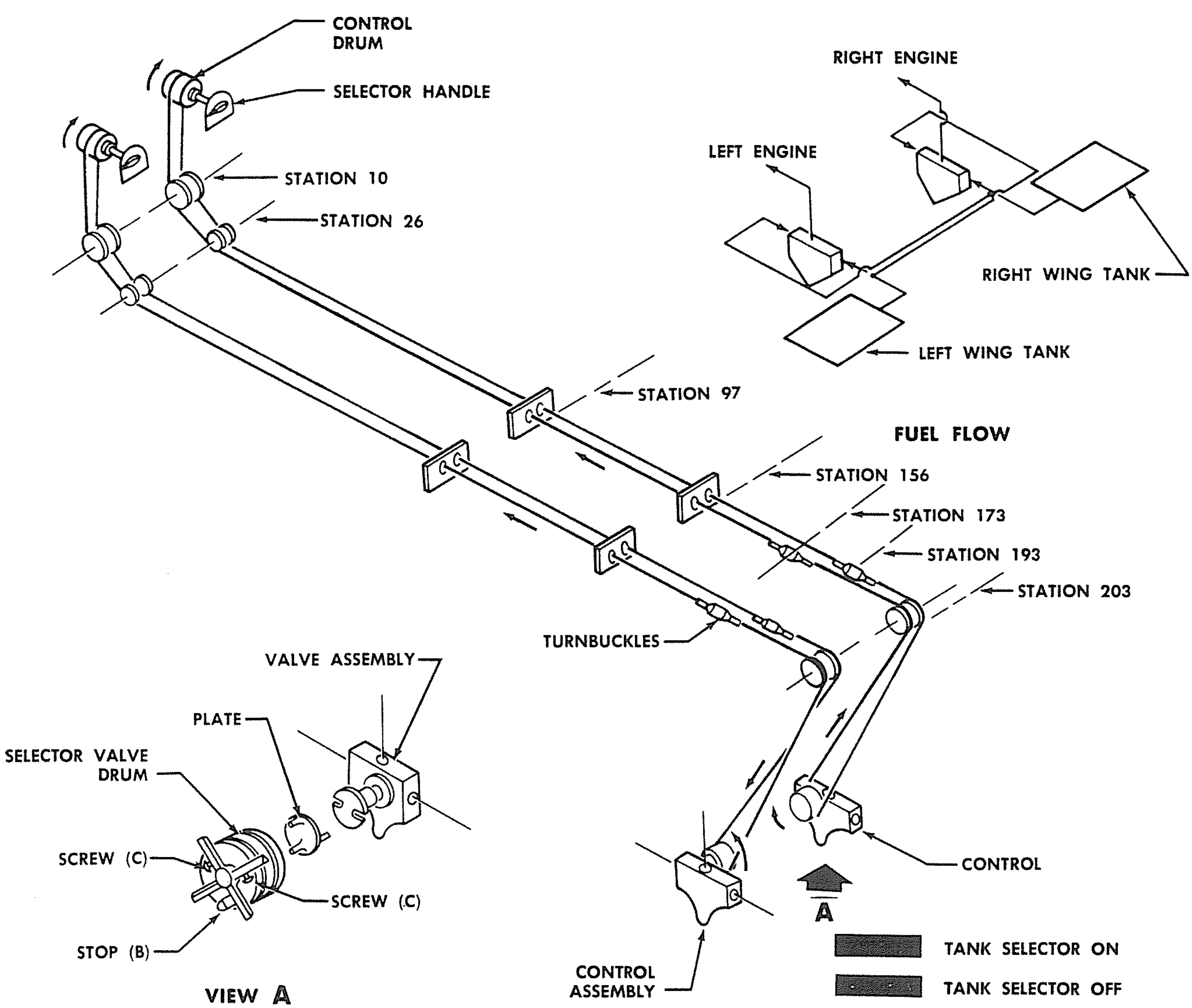
1. Set the drum on the bracket assembly as shown in View A. Valve ports 1 through 4 should be closed.

   Note: 1 5/8-inch dimension shown is approximate. Exact setting should be determined by the integral detent in valve.

2. Set the selector handle in the cockpit in the OFF position as shown in View B and tension the cables with the turnbuckles at Station 183 in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
3. Turn the selector handle counterclockwise from OFF position through RIGHT REAR, RIGHT FRONT, and LEFT FRONT, to LEFT REAR. Valve ports 1 through 4 respectively should open.
4. Position of each valve port opening should be clearly defined by the integral detent in valve.
5. Safety the turnbuckles.

8225

Figure 5-15 (Sheet 5 of 7 Sheets). Fuel Tank Selector Valve Control System — Adjustment Diagram, Center Wing Tank



OUTER WING TANK SELECTOR ADJUSTMENT PROCEDURE

1. Set the valve assembly so that the slots in the driver plate are horizontal, with the wide slot facing aft on the right side of the airplane and forward on the left side, as shown in View A. Note: This setting should be defined by the integral detent in valve.
2. Check the control assemblies for 90 degrees right and left rotation of drum. Install the plate and control assemblies.
3. Rotate the control drum full 90 degrees right and left from the initial setting. The valve position should be defined by the detent when Stop B is approximately 1/16 inch from screws C.
4. Set the control drum and the valve assembly in the OFF position (rotated full counterclockwise) and set the selector handle in the cockpit to OFF.
5. Tension the cables with the turnbuckles at fuselage stations 173 and 193, in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
6. Turn the selector handle to the OFF, LEFT, and RIGHT positions. The valve control drum should rotate from full counterclockwise to full clockwise position and each position should be defined by the integral detent in valve.
7. Safety the turnbuckles.

8226

Figure 5-15 (Sheet 6 of 7 Sheets). Fuel Tank Selector Valve Control System — Adjustment Diagram, Outer Wing Tank

OUTER WING TANK SELECTOR VALVE CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS			
						(1)	(2)	(3)	(4)
1	2398395-503	2	A	174 1/2	3/32 dia 7x7 flex	RA2487-3	AN669L3 RH		
2	2398395-505	2	B	190 1/2	3/32 dia 7x7 flex	RA2487-3	AN669L3 LH		
3	2398395-1	2	B	48	3/32 dia 7x7 flex	RA2487-3	AN669L3 LH		
4	2398395-501	2	A	32	3/32 dia 7x7 flex	RA2487-3	AN669L3 RH		

CENTER WING TANK SELECTOR VALVE CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS			
						(1)	(2)	(3)	(4)
5	3391412-575	2	C	370	3/32 dia 7x7 flex	AN669S3 LH	AN669S3 LH		
6	4392014-513	1	D	108	3/32 dia 7x7 flex	AN669S3 RH	AN669S3 RH		
7	4392014-515	1	D	104 1/2	3/32 dia 7x7 flex	AN669S3 RH	AN669S3 RH		

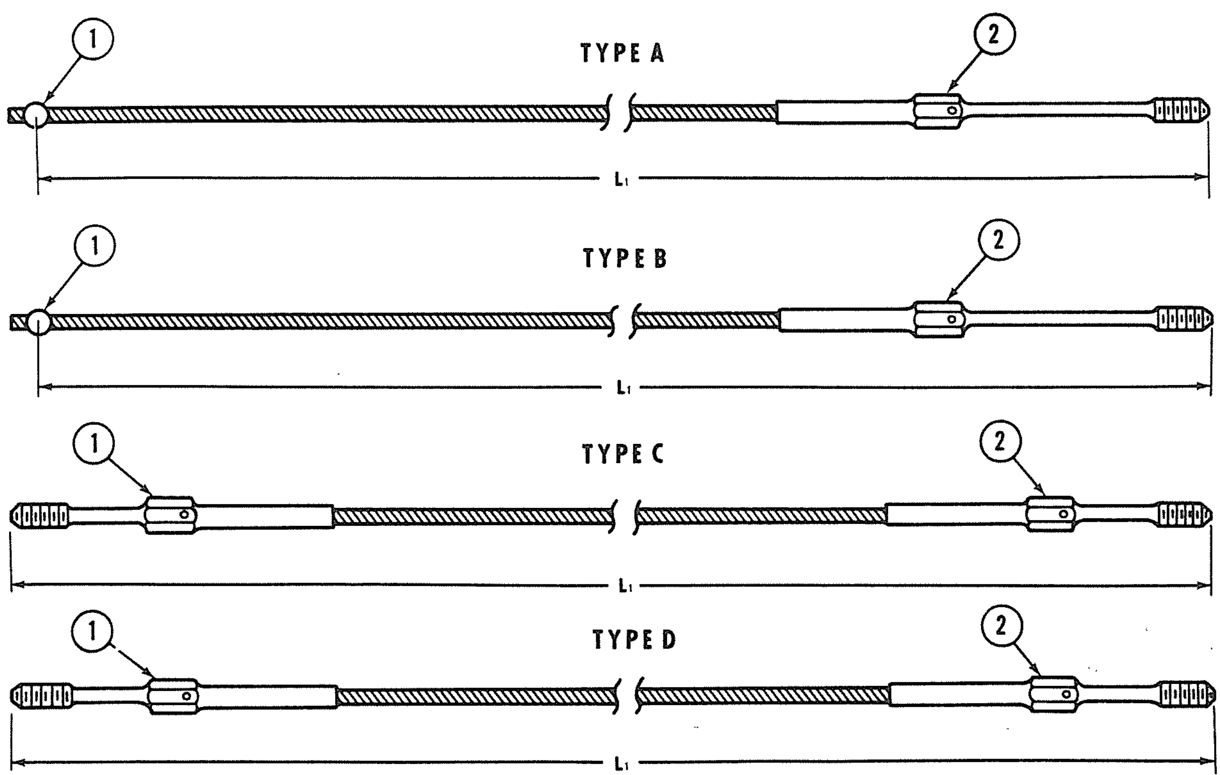


Figure 5-15 (Sheet 7 of 7 Sheets). Fuel Tank Selector Valve Control System — Cable Charts and Cable Assemblies

(Continued from Page 360)

5-61. MINOR REPAIR AND REPLACEMENT OF FUEL TANK SELECTOR VALVE CONTROL CABLES.

a. Inspect all pulleys and fairleads for general condition. Replace any badly worn fairleads and damaged pulleys.

b. Check all cables and replace any cable that shows more than six broken wires in any one-inch length (see paragraph 2-51).

5-62. INSTALLATION OF FUEL TANK SELECTOR VALVE CONTROL CABLES.

a. Wind the aft outer wing and center wing tank selector valve control cables on the selector valve control drums. Route the cables forward through the aircraft, replacing pulley guard pins.

b. Tighten the set screws on the drums.

c. Wind the forward selector valve control cables on the two fuel tank selector drums in the control pedestal and the two drums under the pedestal. Route the cables aft through the fuselage, installing necessary guard pins and fairlead grommets.

d. Tighten the set screws on the drums and connect the cables with turnbuckles at station 183.

e. Adjust the selector valve controls (see paragraph 5-63).

f. Tension the cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

g. Safety all turnbuckles.

h. Replace the access doors and floor panels in the aircraft.

5-63. OPERATIONAL TEST AND ADJUSTMENT OF FUEL TANK SELECTOR VALVE CONTROLS. *(See figure 5-15.)*

a. After installation of the fuel tank selector valve, locate the small notch of the yoke adjacent to the reference mark on the top of the control shaft of the valve. This notch will be adjacent to the valve port that is open.

b. Place the tank selector valve dial indicator in the cockpit in the position corresponding to the valve. That is, if the notch shows that the right rear tank port is open, set the dial indicator to RIGHT REAR. Connect the control cables, being certain that the dial indicator remains in the same position. If the dial indicator is slightly off center, adjust to center by adjusting the cable turnbuckles.

c. Tension the cables between the turnbuckles and the fairlead at station 156, in accordance with the Cable Rigging Tension Chart, figure 2-9.

d. Safety the turnbuckles.

5-64. FUEL STRAINERS. *(See figure 5-16.)* The two fuel pipe strainers, one for each engine fuel system, are located in the center wing on a rib forward of the front spar. The strainers collect sediment from the fuel before it reaches the engines. Drains at the base of these fuel strainers should be serviced daily. A finger strainer is installed in each outer wing fuel tank fuel pipe connector. These strainers are accessible through small access doors on the bottom wing panel aft of the center spar inboard of station 17.9.

5-65. SERVICING FUEL PIPE STRAINERS. A visual inspection of the fuel pipe strainer screen should be made after each 25 hours of flying time. Remove the fuel pipe strainer screen as follows:

a. Place the fuel tank selector dial indicator in the cockpit in the OFF position.

b. Cut safety wires from the strainer and thumbscrew.

c. Drain fuel from the strainer through the drain valve. Remove the plug near the top to let in air.

d. Turn the thumbscrew to the left. The bell-shaped cover will come loose immediately.

e. Remove the screen and cover in one unit.

f. Remove the screen from the cover by pulling straight out. Use a little force if necessary.

g. Clean the screen thoroughly and wipe out the strainer barrel.

h. Check the drain valve to see that it is not clogged.

i. Reinstall the screen in the strainer with the pointed end of the cone-shaped screen in the up position.

j. Before inserting the cover, hold the bell-shaped portion and turn the thumbscrew to the left several turns. Insert the cover and turn to the right. Turn the wing nut to the right until the cover tightens firmly, and safety-wire the wing nut to the housing.

5-66. REMOVAL OF OUTER WING FUEL TANK FINGER STRAINERS.

Note

The outer wing fuel tank finger strainers should be removed and inspected every 1000 hours of flying time.

a. Drain all fuel from the outer wing fuel cells.

b. Remove the access cover on the bottom wing panel aft of the center spar and inboard of wing station 17.9.

c. Remove the two bolts, nuts and washers on the fuel cell fitting to remove the plate and seal.

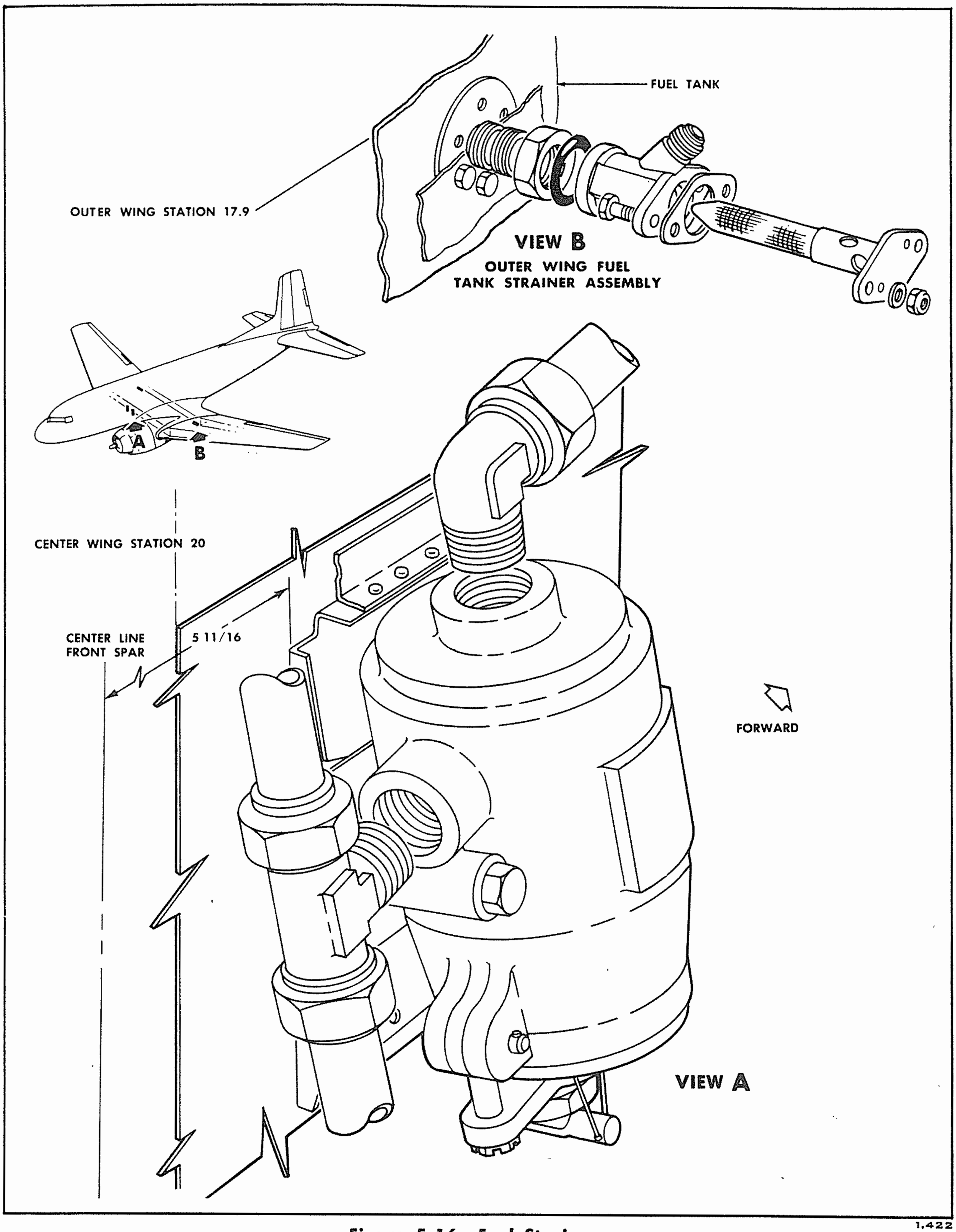


Figure 5-16. Fuel Strainers

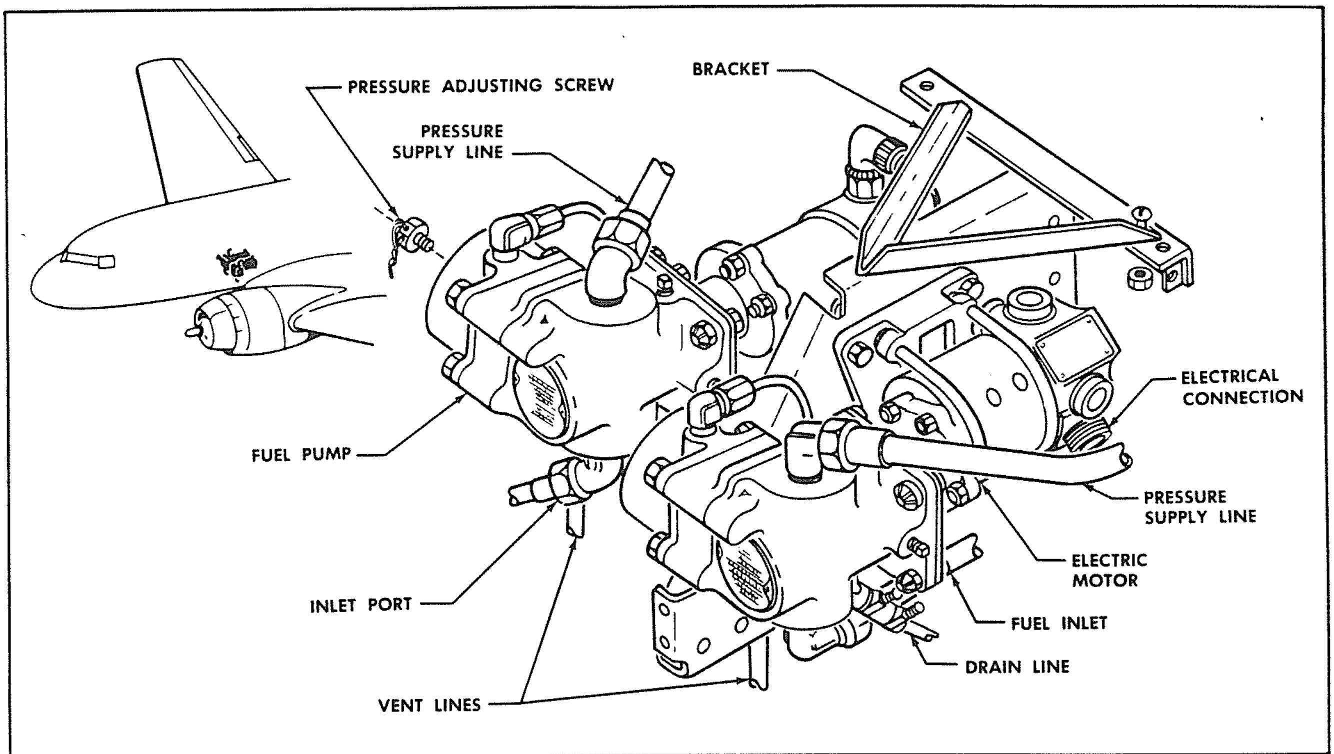


Figure 5-17. Electric Fuel Booster Pumps

d. Using a finger, pull the strainer out of the bottom opening.

5-67. INSTALLATION OF OUTER WING FUEL TANK FINGER STRAINERS. Reverse the removal procedure.

Note

Check the condition of the seal before reinstalling the cover, and replace with a new seal if necessary.

5-68. REMOVAL OF FUEL STRAINER. Ordinarily, it is unnecessary to remove the fuel pipe strainers. However, if the unit is to be replaced, proceed as follows:

a. Remove the access door.

b. Disconnect the fuel pipe.

c. Remove the two bolts that hold the strainer unit to its support and remove the strainer.

5-69. INSTALLATION OF FUEL PIPE STRAINER. Reverse the removal procedure.

5-70. TESTING FUEL STRAINERS AFTER INSTALLATION. After the fuel strainers are installed, turn on the tank selector valves and operate the fuel booster pumps to remove any air from the fuel system. Check each fuel strainer for any leakage of fuel.

5-71. FUEL BOOSTER PUMPS. *(See figure 5-17.)* The two electrically driven fuel booster pumps, one for each engine fuel system, are located in the center wing section forward of station 203. Each 4-blade, rotary vane-type fuel booster pump is powered by a 24-volt, d-c, 1/4-horsepower electric motor, which operates at approximately 2000 rpm. There is a bypass around the booster pump through which the fuel flows when the booster pump is not in use. The booster pump is used for engine starting, for hot weather and high altitude operation, when selecting a new fuel supply, or in case of failure of the engine-driven fuel pump.

5-72. REMOVAL OF FUEL BOOSTER PUMPS.

a. Place the fuel tank selector valve dial indicator in the cockpit in the OFF position.

b. Disconnect the electric plug.

c. Disconnect and cap pipes.

d. Disconnect the bolts attaching the pump to the support bracket and remove the pump.

5-73. INSTALLATION OF FUEL BOOSTER PUMPS. Reverse the removal procedure.

5-74. ADJUSTMENT AND TEST OF FUEL BOOSTER PUMPS AFTER INSTALLATION.

a. With the engine not running and the fuel tank selector valve dial indicators placed in the FRONT TANKS position, test the booster pumps by running them for at least five minutes. At the end of this time, the fuel pressure indicator should read 20.5 to 21.5 psi, at 28 ($\pm$0.1) bus voltage, and there should be no leakage of fuel.

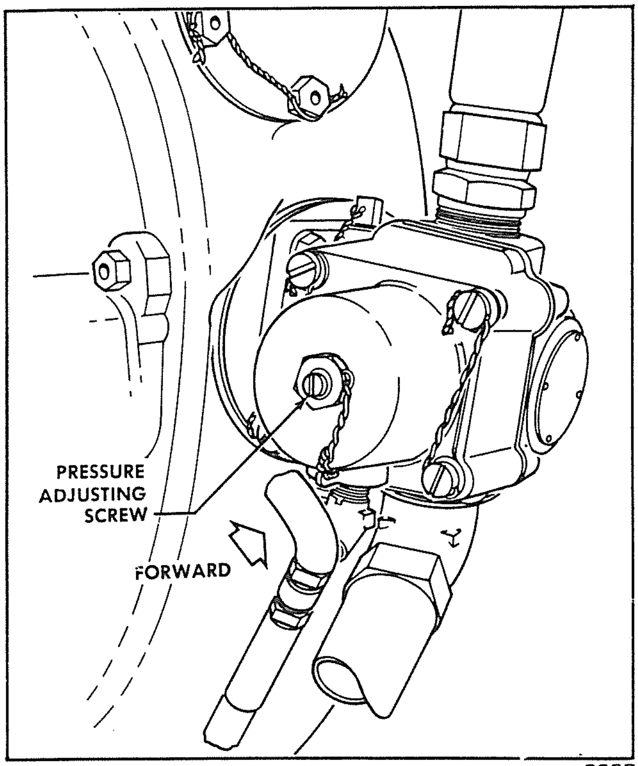


Figure 5-18. Engine-Driven Fuel Pump, Showing Pressure-Adjusting Screw

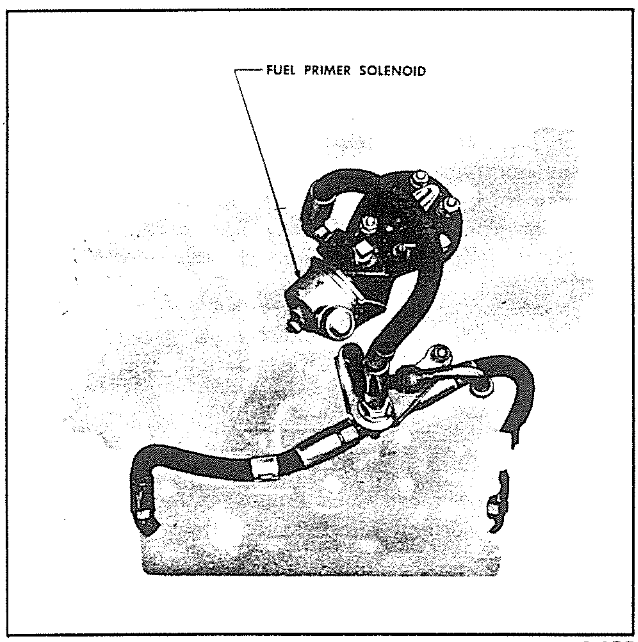


Figure 5-19. Fuel Primer Solenoid

b. If the fuel pressure indicator shows more or less than 20.5 to 21.5 psi, adjust the pressure regulator screw on the pressure relief valve housing of the fuel booster pump by loosening the lock nut and turning the adjusting screw clockwise to increase the fuel pressure or counterclockwise to decrease the fuel pressure. Tighten the lock nut, and replace the cover.

Note

Check the fuel booster pump vent pipe and the pressure relief valve housing for any fuel discharge or leakage. Continued discharge or leakage is sufficient cause for replacing the pump.

5-75. ENGINE-DRIVEN FUEL PUMPS. *(See figure 5-18.)* An engine-driven fuel pump, mounted on the rear case of each engine, is operated by means of a drive shaft coupled with the engine drive gear. The four-blade rotary pump is equipped with an adjustable pressure relief valve which maintains a constant fuel pressure at varying engine speeds. The pressure relief valve is spring-loaded and opens when the pump is operating at high speed, which allows excess fuel to flow back into the intake side. At low operating speeds, the pressure relief valve closes and allows the full flow of fuel to pass through the outlet port. A vent pipe is connected to the diaphragm vent boss and a drain pipe is attached to a drain fitting near the pad to allow drainage of any small amount of fuel that might leak past the seal at the mounting pad. A bypass valve in the pump allows fuel to flow through when the engine-driven pump is inoperative.

5-76. REMOVAL OF ENGINE-DRIVEN FUEL PUMPS.

a. Disconnect and cap all pipes at the engine-driven fuel pump.

b. Remove the four screws securing the pump to the mounting pad.

c. Pull the pump straight out.

5-77. INSTALLATION OF ENGINE-DRIVEN FUEL PUMPS. Reverse the removal procedure.

5-78. ADJUSTMENT AND TEST OF ENGINE-DRIVEN FUEL PUMPS AFTER INSTALLATION. With the engine running at 1500 rpm, the fuel pressure indicator should read 19 to 21 psi. If the fuel pressure indicator registers more or less than 19 to 21 psi, proceed as follows:

a. Adjust the pressure regulator screw on the bottom of the pressure relief valve housing on the engine-driven fuel pump, by loosening the lock nut and turning the adjusting screw clockwise to increase fuel pressure. Turn the adjusting screw counterclockwise to decrease fuel pressure.

b. Check the drain pipe from each engine-driven fuel pump for any fuel discharge. Continued leakage from the drain pipe is sufficient cause for replacement of the pump.

c. Idle the engine and watch the fuel pressure indicator for change in fuel pressure. There should be a pressure drop of approximately two psi. If the pressure drops to 14 psi or lower when the engine is turning at idle speed, the pump should be replaced.

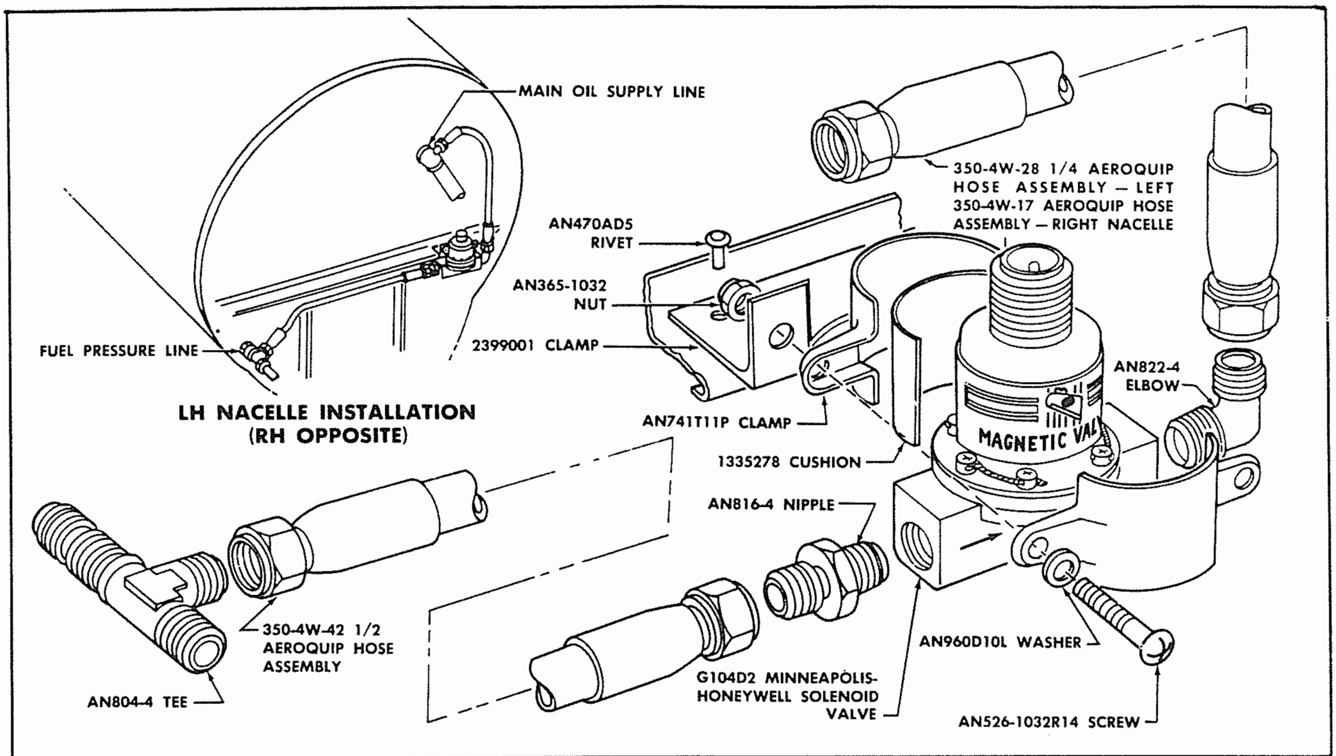


Figure 5-20. Oil Dilution Solenoid

Note

Some fluctuation in fuel pressure may be caused by air in the fuel pipes. Operate the booster pump and then recheck the fuel pressure indicator.

5-79. ENGINE FUEL PRIMER SOLENOID VALVE. *(See figure 5-19.)* An electrically operated, solenoid-type primer valve is mounted on each carburetor. When the valve is operated by an electric switch on the pilot's electrical control panel, fuel flows through internal passages in the carburetor or through the primer valve and then through external piping to two points near the carburetor bottom flange.

5-80. REMOVAL OF ENGINE FUEL PRIMER SOLENOID VALVE.

a. Disconnect and cap the fuel pipe at the valve.

b. Disconnect the electric plug.

c. Remove the attaching screws and remove the valve.

5-81. INSTALLATION OF ENGINE FUEL PRIMER SOLENOID VALVE. Reverse the removal procedure.

5-82. TEST OF ENGINE FUEL PRIMER SOLENOID VALVE AFTER INSTALLATION.

a. Disconnect the pipe on the outlet side of the solenoid valve.

b. Turn on the booster pump and check the valve for leakage. Leakage should not exceed 10 drops of fuel per minute.

5-83. OIL DILUTION SOLENOIDS. *(See figure 5-20.)* An oil dilution solenoid is located on the forward outboard side of each firewall and is connected by means of a flexible hose assembly fitting in the fuel pressure pipe to provide dilution of engine oil. The solenoids are controlled by a three-position toggle-type selector switch located on the co-pilot's electrical panel. When the switch is placed in either the RH or LH position, and the fuel booster or engine-driven pump is ON for the respective engine, fuel passes through the solenoid into a pipe connecting with the engine oil system.

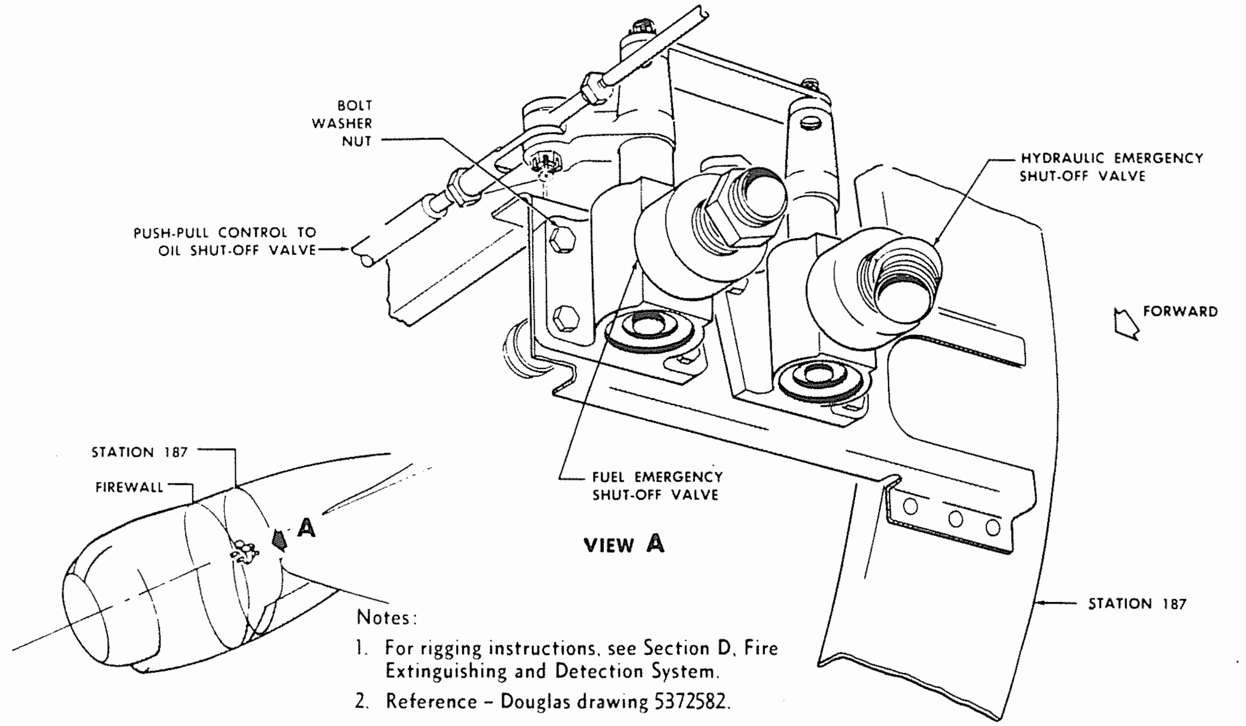
5-84. DRAINING OIL DILUTION SOLENOID. The drain plug on the bottom of the solenoid should be removed periodically to drain any residue or foreign substance that may be present. In case of failure of the solenoid, the unit should be replaced with a serviceable unit.

5-85. REMOVAL OF OIL DILUTION SOLENOID.

a. Remove the electrical disconnect plug.

b. Remove the fuel pipes from the bottom of the solenoid.

c. Loosen the two nuts and clamp and remove the solenoid from the firewall.



Notes:

1. For rigging instructions, see Section D, Fire Extinguishing and Detection System.
2. Reference – Douglas drawing 5372582.

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Figure 5-21. Emergency Shutoff Valves

5-86. INSTALLATION OF OIL DILUTION SOLENOID. Reverse the removal procedure.

Note

When installing the oil dilution solenoid, the arrow indicating the direction of flow on the bottom of the solenoid should point outboard.

5-87. FUEL EMERGENCY SHUTOFF VALVES. *(See figure 5-21.)* A fuel emergency shutoff valve is installed in the main fuel supply pipe to each engine at a point just aft of the firewall. The valves are cable-operated from handles located under the access door on the floor between the pilot's and co-pilot's seats. In case of fire in the engine, the corresponding handle is pulled, which shuts off the fuel, hydraulic, and oil systems to that engine.

5-88. REMOVAL OF FUEL EMERGENCY SHUTOFF VALVES.

a. At the fuel valve end, disconnect the valve actuating rod, the Arens control to the oil shutoff valve, and the link interconnecting the fuel and hydraulic shutoff valves.

b. Disconnect and cap the fuel pipes to the valve ports.

c. Disconnect the attaching bolts and remove the valve.

5-89. INSTALLATION OF FUEL EMERGENCY SHUTOFF VALVES. Reverse the removal procedure.

5-90. ADJUSTMENT OF FUEL EMERGENCY SHUTOFF VALVE CONTROLS. For rigging of the emergency shutoff valves, see paragraph 4-227. After installation of the fuel emergency shutoff valves, check the operation of the fuel emergency shutoff valve controls. With the control handles in the open position, the valves should be open. Operate the control handles by pulling up on them. This should close the shutoff valves.

5-91. FUEL PRESSURE INDICATING SYSTEM. The fuel pressure indicating system consists of two direct-reading fuel pressure indicators installed on the right side of the pilots' instrument panel and two fuel pressure pipes, which run from each instrument to the carburetor and are connected to the carburetor by means of a restrictor fitting. A fuel pressure warning switch, which tees into each fuel pressure pipe, operates a fuel pressure warning light for each engine fuel system *(see figure 5-6)*. The switches are set at 18 psi (maximum) on increasing fuel pressure, and 16 psi (minimum) on decreasing fuel pressure. See paragraph 6-18 for information on the fuel pressure indicating system.

5-92. FUEL QUANTITY INDICATING SYSTEM. The center wing tanks fuel quantity indicating system consists of a multiple-type fuel quantity indicator installed on the right section of the main instrument panel and an electrically-operated Liquidometer-type transmitter installed in the top of each tank. The tank unit consists of a transmitter and support assembly, on which a float arm pivots. An adjustable resistance assembly with screws for adjusting the stroke and end position is an integral part of the tank unit.

5-93. CARBURETOR. Each injection-type carburetor discharges fuel under positive pressure from a discharge bar in the carburetor base near the lower attaching flange. A carburetor preheat system is incorporated in the installation.

5-94. REMOVAL OF CARBURETOR.

a. Remove the upper section of the accessory cowling on the outboard side.

b. Disconnect the neoprene sleeve that extends from the airscoop adapter to the air duct.

c. Remove the carburetor airscoop (see paragraph 5-189).

d. Disconnect the throttle and mixture control rods at the carburetor.

e. Disconnect all pipes which are connected to the carburetor.

f. Remove the attaching bolts which secure the carburetor to the blower housing (using the access door in the fire seal if necessary) and lift the carburetor free.

CAUTION

Upon removal of the carburetor, cover the opening to the impeller and blower section of the engine with cloth or a wood panel to prevent foreign matter from entering the engine and causing damage.

5-95. INSTALLATION OF CARBURETOR. Reverse the removal procedure.

CAUTION

Be sure to reinstall the access door in the fire seal if it has been removed.

5-96. FUEL SYSTEM BOLT TORQUE VALUES. Unless otherwise noted, see paragraph 5-263 for fuel system bolt torque values.

5-97. OIL SYSTEM.

5-98. DESCRIPTION. *(See figure 5-22.)* An independent oil system is provided for each engine and consists primarily of an oil supply tank, an oil cooler with an inlet bypass valve, an oil floating control unit, an electric actuator that controls the oil cooler air exit door, an oil dilution valve, an emergency oil shutoff valve, and engine-driven pressure pumps. The oil tank also supplies oil to the propeller feathering system. Oil flows from the supply tank through the inlet pipe to the oil pressure pump, which is part of the engine. From there the oil is forced under pressure to all parts of the engine that require lubrication. A scavenge pump within the engine drains the oil from the engine sump and pumps it to the oil cooler, and then through the floating control unit back to the tank.

5-99. An oil tank, with a usable capacity of 27¾ US. (33.11 Imp.) gallons, is installed in each nacelle aft of the firewall. A cylindrical hopper is installed in each tank. Return oil from the engine is routed through this hopper to isolate it partially from the rest of the oil supply and enable it to recirculate directly to the engine for quick warm-up. The hopper also tends to remove foam and air from the oil going into the engine. Fuel-diluted oil remains in the hopper after operation of the oil dilution system and stopping of the engine. In this way diluted oil is readily available in case of a cold-weather start.

5-100. A standpipe installed in the oil tank sump fitting assures a reserve of oil available only to the propeller feathering pump. The oil for the propeller feathering system is pumped under pressure to the propeller governor mounted on the engine. The pump is installed below and to the side of each oil supply tank.

5-101. The oil cooler airscoop and exit duct are covered by a stainless steel fairing, which forms the bottom section of each engine section cowling. The airscoop provides cooling air for the oil cooler as well as cooling air to the generator, the engine magnetos, and the engine flexible mounts.

5-102. The mechanically bonded aluminum oil cooler consists essentially of an elliptically shaped shell, baffles, tubes, and header plates. The cooler is an airstream heat exchanger, which cools the heated oil returning oil cooler core and takes up the heat transferred from the hot oil that flows over the outside of the tubes. This cooling action lowers the temperature of the oil to a degree suitable for normal engine operation. A flange on the oil cooler provides a means of mounting the inlet bypass valve.

5-103. The oil cooler air exit door, which is mounted at the aft end of the oil cooler duct and controls the flow of air through the cooler, is opened and closed by an electrically operated actuator. This actuator is controlled automatically by the floating control unit or manually by means of a four-position switch on the pilot's switch panel in the flight compartment. When the switch is in the AUTOMATIC position, the oil is thermostatically maintained at a temperature between 77° and 82°C (170° and 180°F).

5-104. Two oil pressure indicators are installed on the main instrument panel. The oil pressure warning system, which is adjusted to operate at 50 (±5) psi, indicates low oil pressure or an absence of oil. The pressure warning switch operates an indicator lamp installed on the main instrument panel.

5-105. The oil temperature indicating system is composed of a resistance-type temperature bulb located in the supply pipe from the oil tank and an indicator located on the main instrument panel. A change in oil temperature causes a corresponding change in the electrical resistance of the circuit and it, in turn, is reflected in the action of the pointer on the indicator.

5-106. An oil emergency shutoff valve is installed below each oil tank. The valves are mechanically operated by handles located under an access door on the floor between the pilot's and co-pilot's seats. In case of fire in either the left or right engine, the corresponding handle is pulled to shut off the oil as well as the fuel and hydraulic systems and generator air blast to that particular engine.

5-107. MINOR REPAIR AND REPLACEMENT OF OIL SYSTEM COMPONENTS. No minor repair of the oil system components is recommended. If careful inspection locates a component of the system that is malfunctioning, the unit should be removed and replaced with a new or overhauled unit.

5-108. TROUBLE SHOOTING OF OIL SYSTEM.

Trouble	*Probable Cause*	*Remedy*
a. Low oil pressure.	Incorrect engine oil pressure relief valve adjustment.	Adjust oil pressure relief valve.
	High oil tempera-ture.	See step b, following.
	Excessive oil foaming, which may be caused by high-altitude operation, excessively hot oil, or low oil supply.	Check oil in tank. If excessive foaming is evident, drain system and fill with new oil.
	Oil leakage, congealed oil, or air in oil pressure indicator pipe.	Inspect pipes and fittings for leaks. Bleed out congealed oil or air. Refill oil pressure pipes, using hydraulic fluid.
	Faulty engine parts such as oil pump, strainer, oil pressure relief valve, or excessive operating clearances.	Remove and replace faulty units.

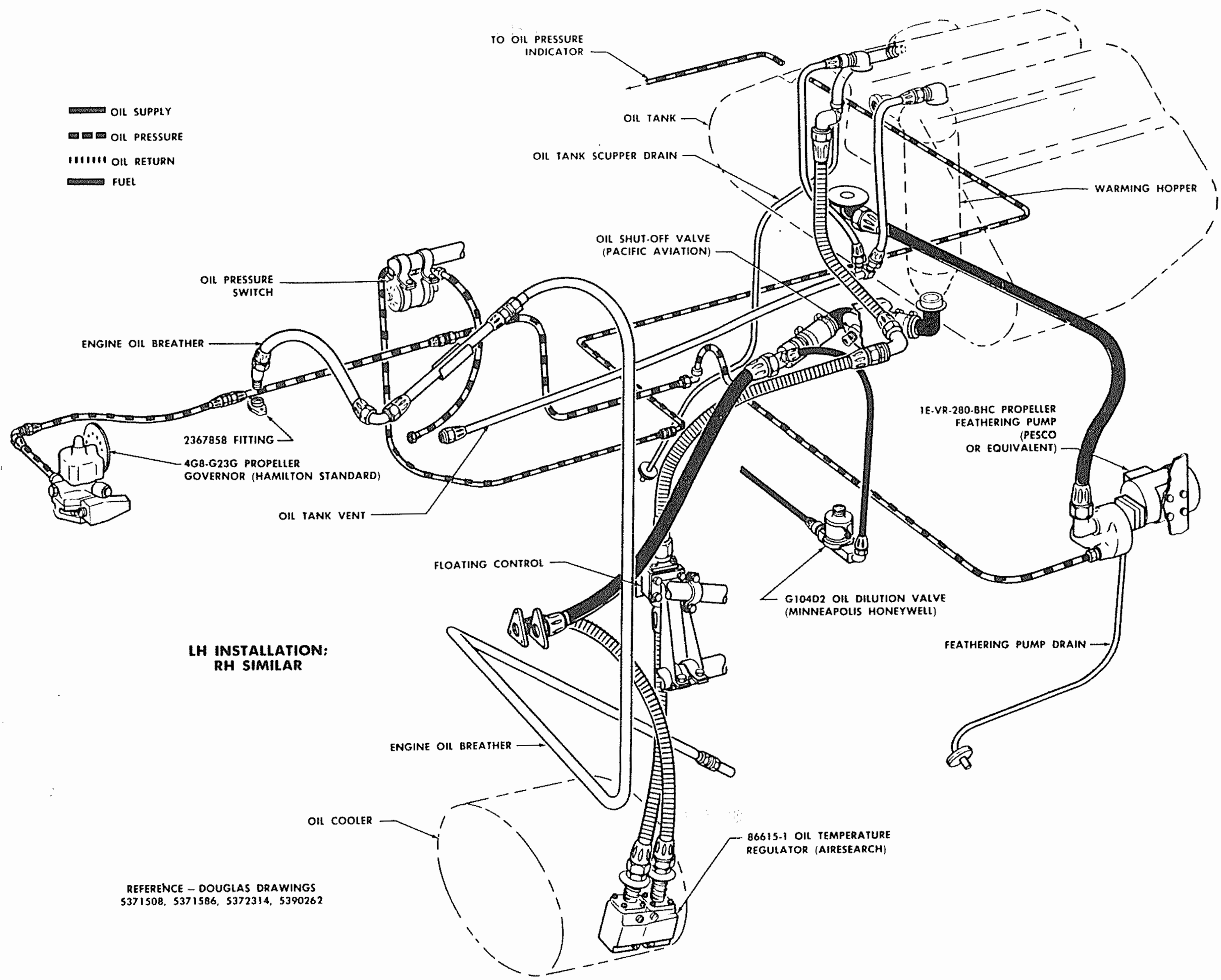


Figure 5-22. Oil System

Trouble	*Probable Cause*	*Remedy*
b. High oil temperature.	Improper setting of oil cooler control switch in cockpit.	Set control switch to AUTO or adjust manually to maintain temperature within limits.
	Insufficient oil supply.	Check oil level in tanks. If supply is very low, check pipes and fittings for leaks.
	Congealed oil in oil cooler.	Warm oil with ground heating unit. *Do not warm oil by using torch or warm water.*
	Diluted or dirty oil.	Check oil in tank. If oil is dirty, drain and refill tank. If oil is diluted, drain and refill tank.
	Damaged or restricted (clogged) oil cooler.	Check cooler and replace if necessary.
	Floating control unit operating improperly.	Check unit and replace if necessary.
	Oil cooler air exit door actuator operating improperly.	Check actuator and replace if necessary.
	Oil cooler inlet bypass valve defective.	Improper spring tension. Check and replace valve if necessary.
c. Fluctuating oil temperatures or inability to maintain required temperature.	Floating control unit; loose contact points or arm binding on pivot.	Replace floating control unit (thermostat).
d. Loss of oil (high oil consumption).	Oil temperature too high.	See step b, preceding.
	Oil pressure too high.	See step d, following.
	Oil leaks in system.	Inspect pipes and fittings for leaks.
	Oil viscosity too low.	Drain and replace oil.
	Faulty engine parts or operation.	Remove and replace faulty units.
	Clogged breather pipes from engine blower case.	Clean out pipes.
e. High oil pressure.	Low air temperature. (Oil too viscous.)	Warm oil supply with ground heat unit. *Do not warm oil by using torch or warm water.*
	Defective oil pressure indicator.	Replace indicator.
	Improper operation of oil pressure relief valve.	Remove and replace oil pressure relief valve.
f. No oil pressure.	Defective oil pressure indicator.	Replace indicator.
	Damaged oil intake pipes or indicator pipes.	Inspect pipes and fittings for damage and replace if necessary.
	Faulty oil pump.	Remove and replace oil pump.
	Air lock in intake pipe.	Pre-oil engine.
g. Fluctuating oil pressure.	Insufficient oil supply.	Check oil level in tanks. If supply is exceedingly low, check pipes and fittings for leaks. If supply is normal, check oil pump for faulty operation.
	Air in oil pressure indicator pipes.	Bleed pipes. Refill pressure pipes with hydraulic fluid.
	Oil viscosity too high.	Warm oil with ground heating unit (*do not warm oil by using torch or warm water*), or replace oil with oil of proper grade.
	Improper operation of oil pressure relief valve.	Remove and replace oil pressure relief valve.
	OIL DILUTION CIRCUIT	
h. Oil-dilution valve fails to operate.	Circuit breaker open.	Reset circuit breaker.
	Short, ground, or open wiring.	Check circuit wiring.
	Faulty control switch.	Remove and replace switch.
	Faulty oil dilution solenoid valve.	Remove and replace oil dilution solenoid valve.

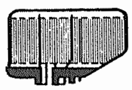
5-109. OIL TANK. *(See figure 5-23.)* A metal oil tank is installed in each nacelle, aft of the firewall. The trunk has a usable capacity of 27¾ US. (33.11 Imp.) gallons. It is saddle-shaped in construction to allow clearance for the alighting gear retracting strut, which it straddles. Baffles within the tank prevent the oil from sloshing back and forth during flight. A filler well is attached to the top of each tank. Overflow oil is drained from this filler well to the outside of the nacelle through a drain tube attached to the well. An oil quantity measuring rod is installed on the filler cap. A sump is provided in the bottom of the tank. This sump is fitted with a drain cock to drain entrapped water and residual oil, including the propeller feathering oil supply.

5-110. DRAINING OIL TANK. For information on draining the oil tank, see paragraph 1-90.

5-111. REMOVAL OF OIL TANK.

a. Drain oil tank (see paragraph 1-90).

b. Disconnect and plug all pipes.

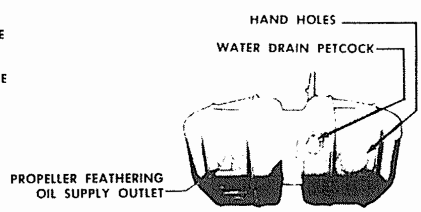


THERMAL EXPANSION AND FOAM SPACE

ENGINE AND PROPELLER FEATHERING USE
PROPELLER FEATHERING USE ONLY

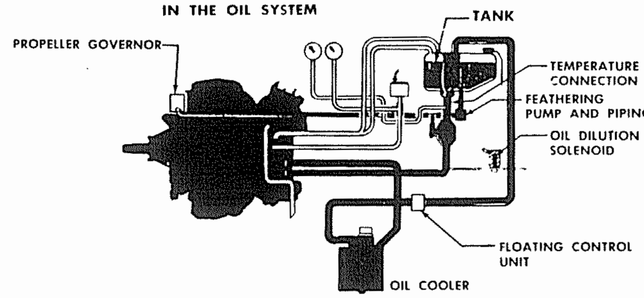
URPOSE OF OIL TANK: To store and introduce lubricating oil into ne oil system. The oil tank provides a temperature accelerating well nopper), a propeller feathering oil supply sump, a quantity meas-ring rod, and connections for vent lines to the engine. It also pro-ides a space, above the 27 3/4-gallon usable capacity level, which llows for foam and 10 per cent oil thermal expansion.

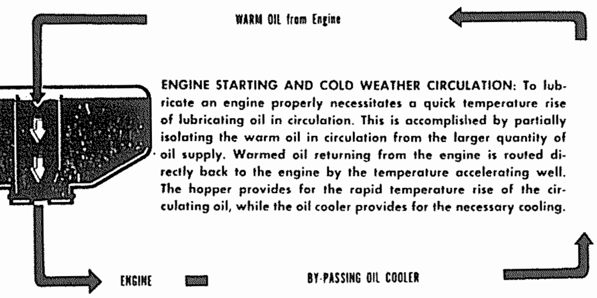
URPOSE OF TEMPERATURE ACCELERATING WELL (HOPPER): To ac-elerate the temperature rise of the circulating oil, by partially sep-rating the oil supply in the tank from the warm used oil returning the hopper enabling the returned oil to recirculate directly to the ngine. The hopper also acts as a means of removing foam and air om the oil going to the engine.



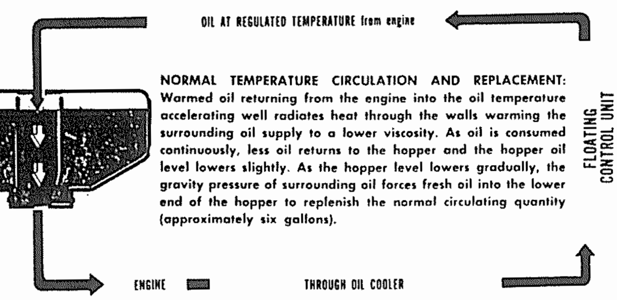
BOTTOM VIEW OF TANK

USE OF SUPPLY TANK AND WARM-UP HOPPER IN THE OIL SYSTEM





ENGINE STARTING AND COLD WEATHER CIRCULATION: To lubricate an engine properly necessitates a quick temperature rise of lubricating oil in circulation. This is accomplished by partially isolating the warm oil in circulation from the larger quantity of oil supply. Warmed oil returning from the engine is routed directly back to the engine by the temperature accelerating well. The hopper provides for the rapid temperature rise of the circulating oil, while the oil cooler provides for the necessary cooling.



NORMAL TEMPERATURE CIRCULATION AND REPLACEMENT: Warmed oil returning from the engine into the oil temperature accelerating well radiates heat through the walls warming the surrounding oil supply to a lower viscosity. As oil is consumed continuously, less oil returns to the hopper and the hopper oil level lowers slightly. As the hopper level lowers gradually, the gravity pressure of surrounding oil forces fresh oil into the lower end of the hopper to replenish the normal circulating quantity (approximately six gallons).

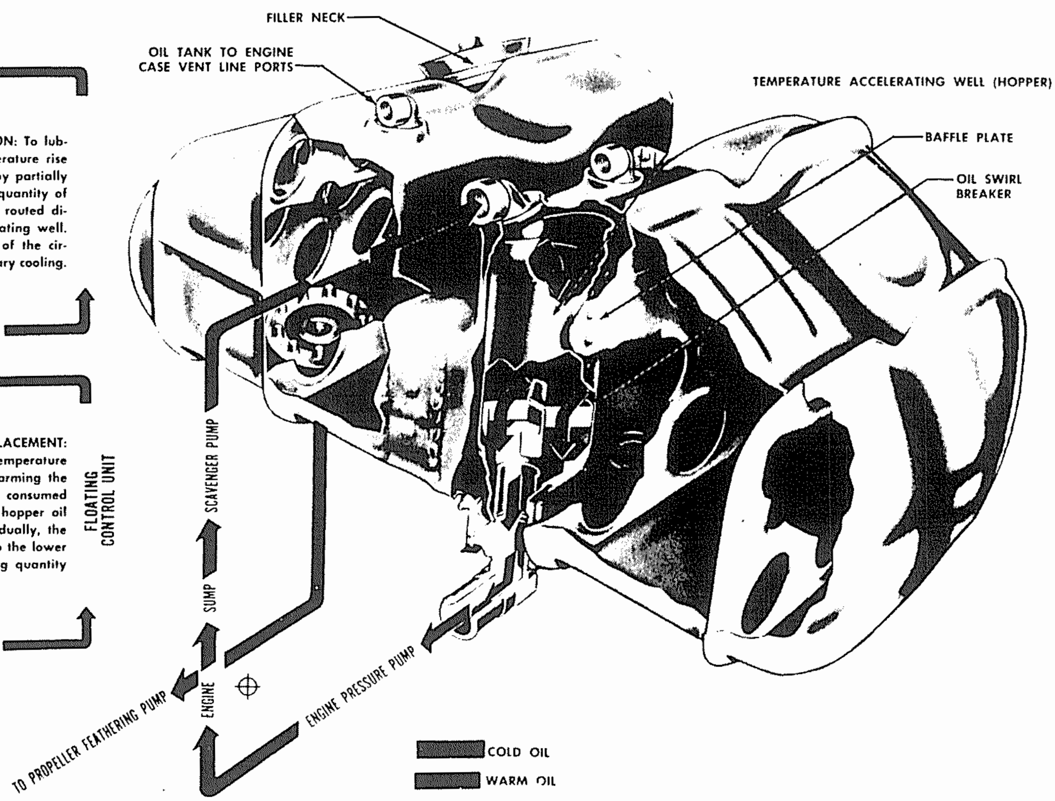


Figure 5-23. Oil Tank Diagram

c. Detach the bonding strips.

d. Remove the tank cross brace above the alighting gear retracting strut.

e. Release and remove the tank retaining straps.

f. Remove the oil tank, lowering it down and out of the nacelle.

5-112. INSTALLATION OF OIL TANK. Reverse the removal procedure.

5-113. FILLING OIL TANK. For information on filling the oil tank, see paragraph 1-89. When filling the oil tank after an engine change, refill the system with oil, Specification MIL-O-6082, Grade 1065 (winter) and 1100 (summer), and proceed as follows:

a. Eliminate all air bubbles from the propeller feathering system pipes as follows: Disconnect the pipe at the propeller governor and hold a vessel beneath the point of disconnection. Push the applicable propeller feathering button IN. This energizes the switch which starts the pump. Watch for bubbles of air at this point. When all air bubbles have disappeared, pull out the propeller feathering button, and reconnect the pipe.

b. Recheck the oil level and fill the tank to the required quantity.

c. Replace the tank filler cap and close the filler cap access door.

5-114. OIL COOLING SYSTEM. *(See figure 5-24.)* The oil cooling system consists essentially of an airscoop, an oil cooler, an air exit door that controls the flow of air through the cooler, an actuator that opens and closes the air exit door, an oil floating control unit that controls operation of the air exit door actuator, and an inlet bypass valve, which is mounted on the cooler and controls the flow of oil through it.

5-115. OIL COOLER AIRSCOOP. The oil cooler airscoop and air exit duct are covered by a stainless-steel fairing, which forms the bottom part of each engine section cowling. The airscoop provides cooling air for the oil cooler and, in addition, supplies cooling air to the generator, the engine magnetos, and the engine flexible mounts. The oil cooler receives ram air directly from the airscoop. The generator, magnetos, and engine flexible mounts receive ram air through a faired-in inlet duct welded to the top of the oil cooler airscoop duct. This inlet duct slips inside a blast tube shutoff valve assembly attached to the bottom aft side of the inner ring, approximately at the engine section centerline. Blast tubes extend from the shutoff valve assembly to the generator and to the rubber engine mounts and the engine magnetos. The tubes are interconnected by hoses and clamps. A drip pan is bolted inside the cowling above the airscoop duct and provides a fire seal between the power section of the engine and the area around the oil cooler and oil cooler entrance duct. The drip pan is connected by a hose to an overboard outlet on the left side of the airscoop. Two triangular-shaped access doors, one on each side and directly above the airscoop inlet, provide access to the lower parts of the engine and the drip pan attachment bolts. The forward end of the airscoop is attached to the anti-drag ring bow-ring support with screws. The aft end of the scoop assembly attaches to the aft oil cooler fairing and inner ring with screws. The right and left upper edges are held down when the fixed cowl flaps are attached. Additional attachment is provided at each side of the forward oil cooler cradle where the scoop is bolted to the aft oil cooler fairing. (There are two bolts on each side, accessible through the oil cooler access door when the cooler is removed.)

5-116. REMOVAL OF OIL COOLER AIRSCOOP.

a. Place the cowl flaps selector switch, located on the left overhead electrical panel in the flight compartment, in OPEN position to extend the flap to the full open position.

b. Remove the two lower segments of the anti-drag ring (see paragraph 5-8).

c. Loosen the Camloc fasteners and remove the lower fixed cowl flaps, one on each side of the engine section.

d. Remove the oil cooler (see paragraph 5-119).

e. Remove two bolts from each side of the forward oil cooler cradle, which attach the oil cooler airscoop and the oil cooler aft fairing.

f. Remove the two triangular-shaped access doors, one on each side directly above the airscoop inlet. Working through these doors and the space from which the fixed cowl flaps were removed, remove the bolts attaching the drip pan, and remove the drip pan.

g. Remove the screws attaching the airscoop to the anti-drag ring bow-ring support at the forward end, and the screws attaching the aft end to the aft oil cooler fairing and inner ring.

h. Remove the oil cooler airscoop.

5-117. INSTALLATION OF OIL COOLER AIRSCOOP. Reverse the removal procedure.

5-118. OIL COOLER. *(See figure 5-24.)* The mechanically bonded aluminum oil cooler consists essentially of an elliptically shaped shell, baffles, tubes, and header plates. The cooler is an airstream heat exchanger, which cools the heated oil returning from the engine. The inlet bypass valve is a four-port, spring-loaded valve designed to operate in conjunction with the oil cooler in directing oil flow according to cooling requirements and pressure in the system. Increased viscosity, due to low temperatures, causes a pressure drop across the cooler core and the bypass passages and thus increases the pressure in the inlet chamber. The high pressure

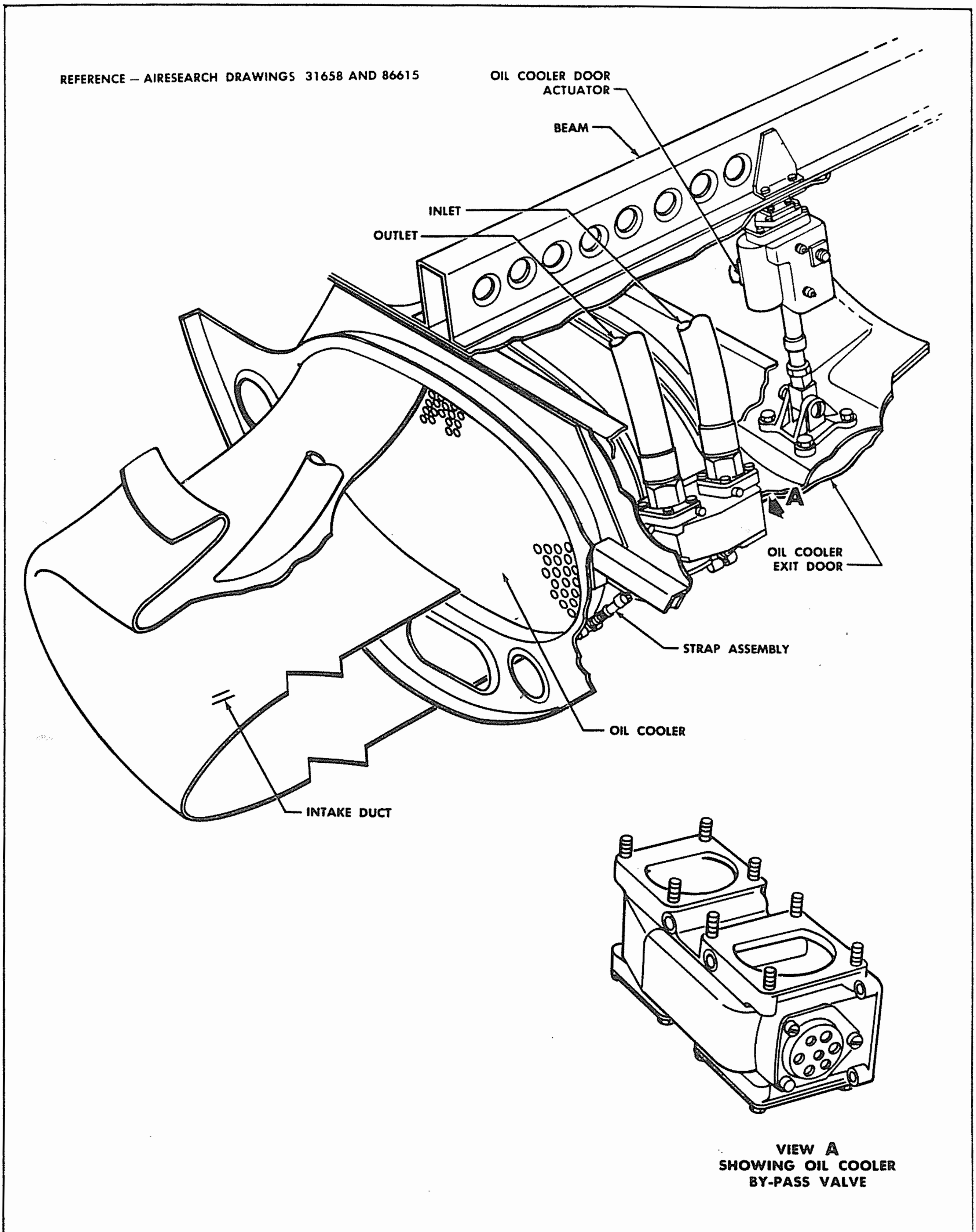


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Figure 5-24. Oil Cooler

that builds up opens the surge valve partially, allowing the flow to bypass to the cooler outlet. When the pressure is sufficiently high, the surge valve completely closes the entrance to the cooler core, bypassing the entire flow through the valve to the outlet. With no cooling, the oil in the pipes and in the tank hopper will warm up rapidly, reducing the pressures and allowing the surge valve to close. This permits flow to the cooler core entrance. With the oil in the core still in a congealed condition, the flow takes the line of least resistance and opens the bypass relief valve, which permits flow to the bypass passages around the cooler core, warming the oil in the core. The bypass passages lead back to the core outlet from the cooler, where the oil flows past the core check valve and into the outlet chamber of the valve. As the temperature of the oil rises, the pressure drop across the cooler is reduced. With the pressure reduction, the bypass relief valve closes, and the entire flow is directed through the cooler core and out the core check valve to the outlet chamber. The core is protected against pipe surges during normal operation by the bypass relief valve.

5-119. REMOVAL OF OIL COOLER.

a. Remove the oil cooler access door.

b. Drain the oil cooler by opening the plug on the bottom of the unit.

c. Disconnect and plug the two pipes at the oil inlet and outlet ports.

d. Remove the straps holding the cooler and remove the cooler.

5-120. INSTALLATION OF OIL COOLER. Reverse the removal procedure.

5-121. OIL COOLER INLET BYPASS VALVE. *(See figure 5-24.)* The inlet bypass valve, installed directly on the oil cooler, is a four-port spring-loaded valve designed to operate in conjunction with the oil cooler in directing oil flow according to cooling requirements and pressure in the system. The interior of the valve housing contains three valve assemblies: the surge protection valve, the bypass relief valve, and the core check valve. The inlet by-pass valve operates under the three following primary conditions: during a cold start, when high oil pressure exists; at low oil temperature; and at high oil temperatures.

5-122. REMOVAL OF OIL COOLER INLET BYPASS VALVE.

a. Remove the oil cooler according to the procedure outlined in paragraph 5-119.

b. Remove the inlet bypass valve from the oil cooler assembly by removing the attaching bolts.

5-123. INSTALLATION OF OIL COOLER INLET BYPASS VALVE. Reverse the removal procedure.

5-124. OIL COOLER SUPPORT AND AFT FAIRING ASSEMBLY. The oil cooler support and aft fairing assembly is an integral unit. The forward end of the assembly is attached to the inner ring and the aft end is attached to mounting plates that are a part of the engine mount.

5-125. REMOVAL OF OIL COOLER SUPPORT AND AFT FAIRING ASSEMBLY.

a. Disconnect the oil pipes attached to the bypass valve on the oil cooler.

b. Disconnect the actuator electrical connection.

c. Remove the oil cooler airscoop (see paragraph 5-116).

d. Support the fairing and remove the eight bolts that attach the aft end of the fairing assembly to the engine mount at the firewall.

e. Remove the bolts holding the forward end of the fairing assembly to the inner ring.

f. Remove the unit.

5-126. INSTALLATION OF OIL COOLER SUPPORT AND AFT FAIRING ASSEMBLY. Reverse the removal procedure.

5-127. OIL COOLER AIR EXIT DOOR. *(See figure 5-24.)* The oil cooler air exit door, formed of aluminum-alloy sheet, is mounted at the aft end of the oil cooler duct. A panel on each side, perpendicular to the door, increases the efficiency of the oil cooler air exit. Two cast hinge brackets are mounted on the forward side of the door and a cast actuator hinge bracket is mounted near the trailing edge of the door. The door is opened and closed by an electrically operated actuator, which receives signals from the floating control unit.

5-128. REMOVAL OF OIL COOLER AIR EXIT DOOR.

a. Open the oil cooler air exit door by holding the control switch in the flight compartment in the OPEN position.

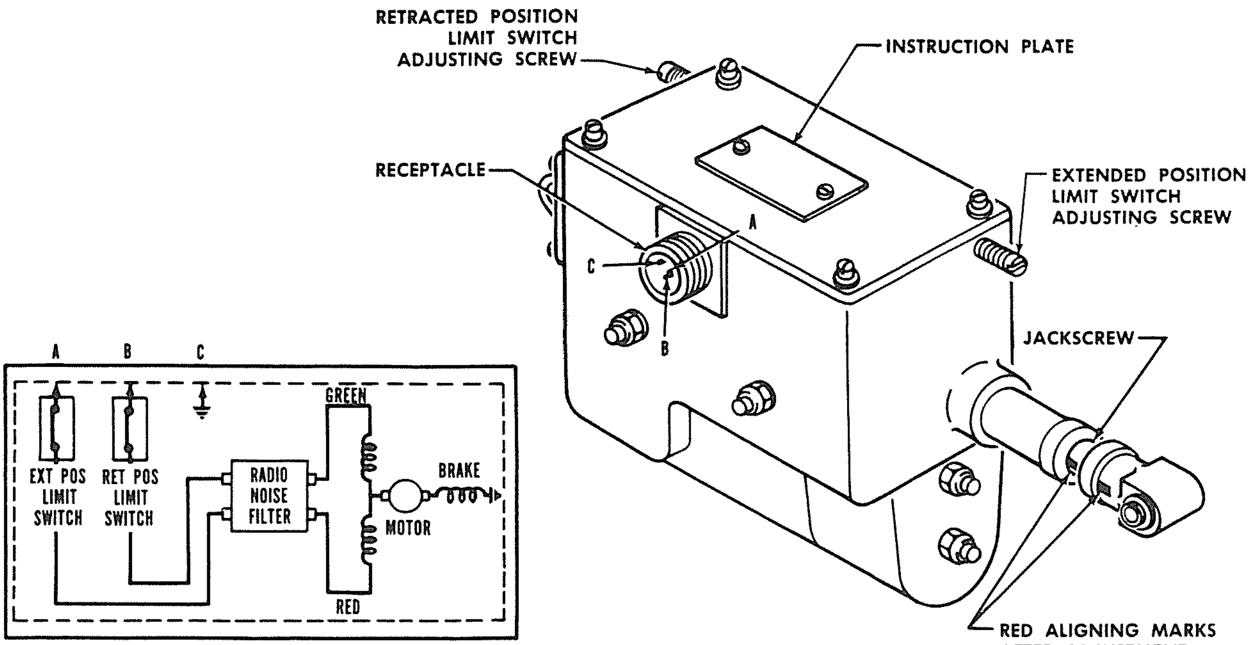
b. Remove the bolt attaching the cooler air exit door to the actuator jackscrew end fitting.

c Remove the two bolts attaching the cooler air exit door to the two hinge brackets and remove the door.

5-129. INSTALLATION OF OIL COOLER AIR EXIT DOOR. Reverse the removal procedure.

5-130. ADJUSTMENT OF OIL COOLER AIR EXIT DOOR.

a. With the door disconnected at the jackscrew shaft, turn the control switch on the overhead electrical panel in the flight compartment to CLOSE until the limit switch opens the motor circuit at the full retracted position.



Notes:

1. Current at rated load 3 amps.
2. Duty cycle: 3 min on—17 min off.
3. Time of stroke (normal 4.00) 9 seconds maximum.
4. Radio noise free per spec AAF32466A.
5. Unit equipped with non-jamming stops.

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Figure 5-25. Air Exit Door Actuator

b. With the actuator in the fully retracted position, manually screw the jackshaft in until it bottoms. Reverse the direction of rotation and back the jackshaft out a few complete turns to prevent the shaft from bottoming.

c. Turn the control switch to OPEN until the limit switch opens the motor circuit at the fully extended position.

d. Counting the turns, manually screw the jackshaft out a few turns to see that it is not bottomed at the limit switch extended position. If the shaft does bottom, limit switch adjustment may be necessary. Otherwise, reverse the direction of rotation to obtain the original position.

e. Connect the door to the jackscrew shaft and energize the power unit slowly to the door full closed position by moving the switch from CLOSE to OFF several times so as to prevent damage if the jackscrew shaft is too short.

f. If adjustment is not correct, return the door to the open position and shorten or lengthen the jackshaft by rotating the shaft in the proper direction. Repeat step d, preceding. If the jackshaft has moved farther than the previous check, check for bottoming. The door trailing edge should fit the mating surface with a gap of $1/16$ ($\pm 1/16$) inches (no preload if touching), when the correct closed adjustment is obtained.

g. If the closed adjustment is correct, energize the power unit through a complete cycle to be sure that the door fully opens.

CAUTION

Power unit limit switches are set for the correct number of revolutions of the power unit that will furnish the normal stroke for the jackscrew shaft. Unless it is absolutely necessary to obtain the correct adjustment, do not change this setting or attempt to adjust the two limit switches.

5-131. OIL COOLER AIR EXIT DOOR ACTUATOR. *(See figures 5-24 and 5-25.)* The electric actuator, which is controlled automatically by the floating control unit or manually by a switch on the overhead electrical switch panel in the flight compartment, opens and closes the oil cooler air exit door. The actuator is is hinge-mounted at both ends in order to maintain correct relationship with the door while it opens and closes. The reversible electric power unit drives a reduction gear assembly, which actuates the jackscrew shaft. Fully extended and retracted positions of the

door are controlled by limit switches. The motor also contains an electromagnetic brake for exact positioning without coast of the actuator.

5-132. REMOVAL OF OIL COOLER AIR EXIT DOOR ACTUATOR.

a. Open the oil cooler air exit door by holding the control switch on the overhead electrical panel in the flight compartment in the OPEN position.

b. Disconnect the electrical connection at the actuator.

c. Remove the bolt attaching the cooler air exit door to the actuator jackscrew end fitting.

d. Remove the bolt attaching the upper end of the actuator to the structure within the air exit duct.

5-133. INSTALLATION OF OIL COOLER AIR EXIT DOOR ACTUATOR. Reverse the removal procedure using the adjustment procedures outlined in paragraph 5-130.

5-134. FLOATING CONTROL UNIT. *(See figure 5-26.)* The oil floating control unit, also known as the thermostat, installed in the oil return pipe above the oil cooler in each nacelle, consists of a bimetallic immersion type coil assembly enclosed in an aluminum alloy housing, a support assembly, seals, arms assembly, cover assembly, and a continuous running constant speed 27-volt d-c motor. The motor operates with a maximum current consumption of 0.25 ampere. A capacitor and inductance type filter is installed in the drive motor circuit to eliminate radio noise. The bimetallic element is secured to the support casting at one end. The other end is fastened to a shaft which extends into the base housing and rotates in ball bearings. A contact arm with a double-faced contact point is mounted on one end of the serrated shaft and is held in place by a nut and washer. Floating contact arms are located on each side of the double-faced contact arm and are oscillated by a continuously rotating cam connected to the motor through a reduction gear box assembly. The temperature regulating range of the unit is set at 77° to 82° C (170° to 180° F). The bimetallic element is in direct contact with the temperature variations of the oil. A change in oil temperature causes the bimetallic element to contract (wind) or expand (unwind), rotating the shaft. This rotation is transmitted to the center arm with the double-faced contact point which connects with one of the floating contact arms. Contact with one floating arm sends electrical impulses through the pin (A) thereby increasing the opening of the air exit door. Contact with the other arm sends electrical impulses through pin (C) thereby decreasing the opening of the air exit door. The proximity of the center contact arm to a floating arm contact point determines the duration of the current impulse to the actuator which operates the air exit door.

5-135. REMOVAL OF FLOATING CONTROL UNIT.

a. Disconnect and cap the oil hose assemblies at the floating control unit.

b. Disconnect the electrical wiring.

c. Remove the bolts attaching the unit to its support and remove the unit.

5-136. INSTALLATION OF FLOATING CONTROL UNIT. Reverse the removal procedure.

5-137. OPERATIONAL CHECK OF FLOATING CONTROL UNIT. Check the operation of the unit by manually running the air exit door to the open position. Set the manual override switch to automatic. The door should close with a cold engine.

Note

The unit is calibrated at the factory for the proper temperature regulating range and no adjustment is to be made while the unit is installed in the aircraft.

5-138. OIL PRESSURE INDICATING SYSTEM. *(See figure 5-22.)* The oil pressure indicating system consists of two oil pressure indicators, installed on the main instrument panel in the flight compartment, and pipes leading from the instruments to the oil pressure fittings on the engines. The indicators are of the direct-reading type.

5-139. OIL PRESSURE WARNING SYSTEM. *(See figure 5-22.)* The oil pressure warning system indicates low pressure or absence of oil pressure in either engine. The system consists essentially of a pressure warning switch located on the engine mount on the right side of each engine and an indicator lamp for each engine, installed on the main instrument panel in the flight compartment. The warning switch is connected in series with an indicator lamp. A decrease in oil pressure below a safe level causes the pressure switch to close and the warning lamp to illuminate. The pressure switch is adjusted to operate at 50 (±5) psi.

5-140. OIL TEMPERATURE INDICATING SYSTEM. *(See figure 5-22.)* The oil temperature indicating system consists of an electrical resistance-type temperature bulb located in the main engine oil supply pipe in each nacelle and two oil temperature indicators located on the main instrument panel in the flight compartment. A change in temperature of the bulb causes a corresponding change in its electrical resistance and results in an unbalance in the indicator circuit. The unbalance causes the pointer to swing up or down the scale according to the temperature change.

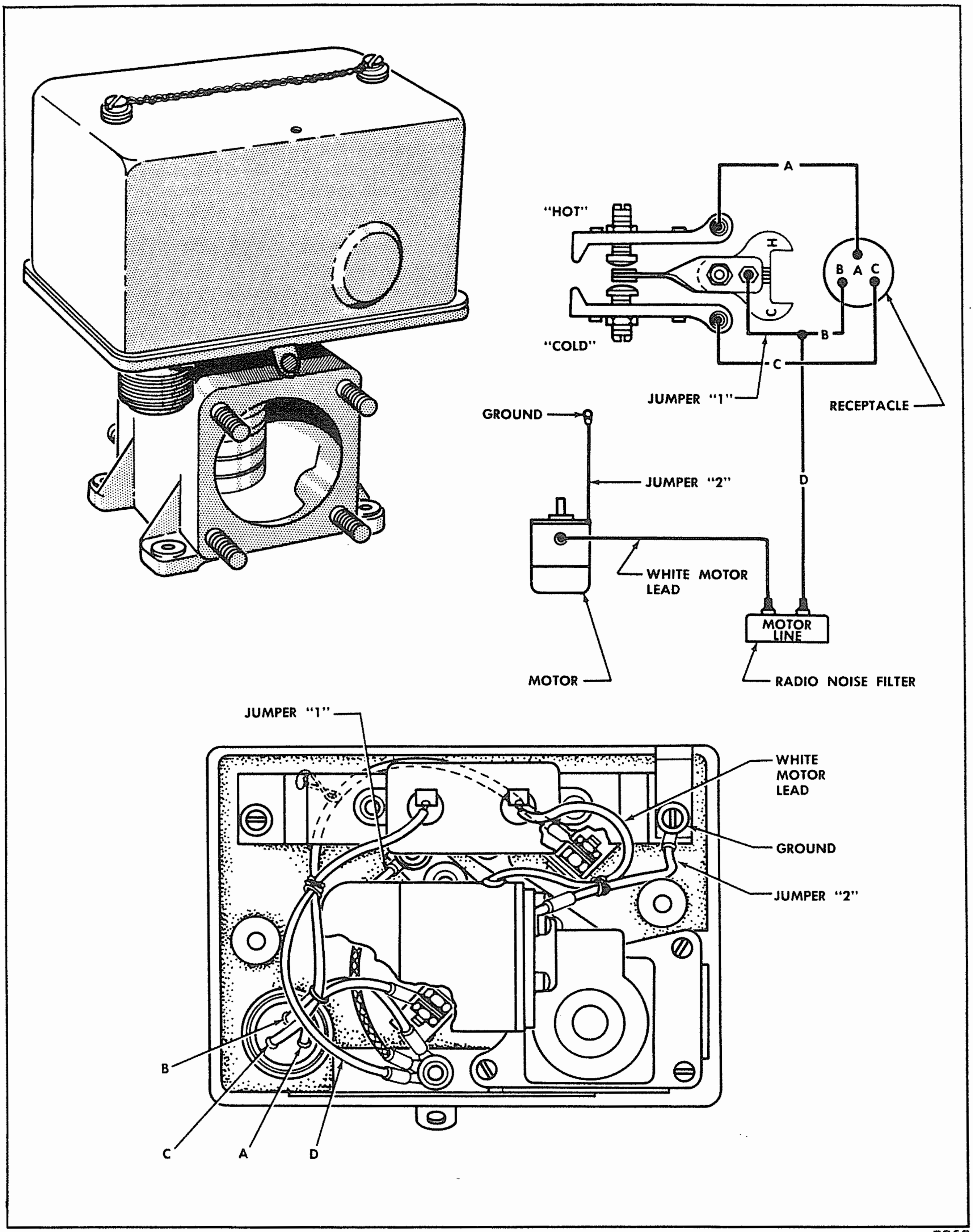

Figure 5-26. Floating Control Thermostat Assembly

5-141. OIL EMERGENCY SHUTOFF VALVE. *(See figures 5-22 and 4-35.)* An oil emergency shutoff valve is installed in each engine oil supply pipe, below each oil tank. The valves are mechanically operated by handles located under the access door on the floor between the pilot's and co-pilot's seat. In case of fire in either the left or right engine, the corresponding handle is pulled, shutting off the flow of oil, fuel, and hydraulic fluid to the engine, as well as the air blast to the generator, magnetos, and engine mounts.

5-142. REMOVAL OF OIL EMERGENCY SHUTOFF VALVE.

a. Drain the oil tank (see paragraph 1-90).

b. Disconnect the control attached to the valve arm.

c. Loosen the hose clamps on the ends of the hoses adjacent to the valve.

d. Disconnect the bolts attaching the valve to the support and remove the valve.

e. Plug the pipes at the hoses adjacent to the valve.

5-143. INSTALLATION OF OIL EMERGENCY SHUTOFF VALVE. Reverse the removal procedure.

5-144. ADJUSTMENT OF OIL EMERGENCY SHUTOFF VALVE CONTROLS. After installation of the oil emergency shutoff valves, check the operation of the controls. With the control handles, located under the access door in the floor of the flight compartment, in the up position, the valves should be closed.

5-145. OIL DILUTION SYSTEM. *(See figure 5-22.)* An electrically operated oil dilution solenoid valve mounted on the forward side of the firewall allows fuel to flow from the fuel pressure pipe into the engine oil supply pipe when the oil dilution switch is energized. This system is used before the engines are stopped when a cold-weather start is anticipated.

5-146. PROPELLER FEATHERING OIL SYSTEM. *(See figure 5-22.)* The oil for the propeller feathering system is supplied from a fitting in the bottom of each oil tank. A propeller feathering pump is installed below and to the side of each tank. The pump supplies oil under pressure to the propeller governor mounted on the engine.

5-147. REMOVAL AND INSTALLATION OF PROPELLER FEATHERING SYSTEM COMPONENTS. For removal and installation procedures for components of the propeller feathering system, see paragraphs 5-210 through 5-217.

5-148. OIL SYSTEM BOLT TORQUE VALUES. For information on oil system bolt torque values, see paragraph 5-263.

5-149. IGNITION SYSTEM.

5-150. DESCRIPTION. *(See figure 5-27.)* The ignition system includes ignition switches, magnetos, ignition harness, spark plugs, and an induction vibrator. The ignition switch in the cockpit incorporates a master switch for the entire system as well as left and right switches for each engine. The ignition switch leads connect directly to the right and left magnetos. Two spark plugs are installed in each of the nine cylinders. Ignition is furnished to the spark plugs by two magnetos, mounted on the rear accessory section of each engine. The right magneto fires the front spark plugs of all cylinders and the left magneto fires the rear spark plugs. The single-type flange-mounted magnetos may be used on either the right or left sides if the radio shield assembly used is equipped with the correct outlet. Rotation of the magneto is counterclockwise, as viewed from the drive shaft end of the magneto.

5-151. In order to suppress disturbances caused by electro-static waves radiated by the ignition system, the entire magneto, including the primary circuit, is completely enclosed in metal shieldings. The coil cover contains a special radio shield that suppresses any electro-static waves radiated by the high-tension coil. To prevent accidental starting of the engine if the ground lead is removed from the magneto, the insulation terminal in the breaker compartment carries on its upper surface an automatic grounding spring. When the terminal assembly is removed, this acts as a safety device by automatically contacting a pin just under the rim of the breaker compartment, thereby short-circuiting the primary circuit and removing any possibility of unexpected ignition. With the grounding lead installed, the short-circuiting spring is compressed downward against the insulation terminal, and the magneto may only be short-circuited by throwing the ignition switch to the OFF position.

5-152. An AN3213-1 ignition switch is located between the two overhead electrical panels in the cockpit. The switch unit incorporates a master ON-OFF switch, and an ignition switch for each engine. The master switch must be in the ON position for the individual engine switches to operate. Each engine magneto switch has four positions, which control the four circuit conditions: OFF, L, R, and BOTH. When the switch is in the OFF position, the circuits are grounded out and are inoperative. When the switch is on R, the right magneto ground circuit is open and spark is furnished to the front spark plugs of all cylinders. When the switch is on L, the left magneto ground circuit is open and spark is furnished to the rear plugs. When the switch is in the BOTH position, both ground circuits are open and both magnetos are operating.

5-153. An antler pipe and flexible conduit assembly enclose the cables from the magneto outlet of the manifold to the magneto block of the engine. The cables are secured to a terminal insulation block within a magneto elbow attached to the conduit and are arranged and assembled in the same order as the firing sequence of the engine cylinders.

5-154. The detachable lead-type radio-shielded ignition manifold is designed to facilitate replacement of the individual spark plug leads in the field. The assembly consists of a circular manifold tube that fits on the front section of the engine and carries the ignition cables, coming from the magneto block to the spark plug. Six mounting lugs are provided on the inside of the circular manifold to fit the crankcase front section to main section attaching screws. A clamp bracket support arrangement is provided for attaching each conduit to the engine. At the spark plug outlet of the manifold there is a spring contact making electrical connection with the wire going to the spark plug. This electrical connection is enclosed in a phenolic sleeve which, in turn, is encased in a metal sleeve body. The conduits carrying the wires from the electrical connections to the spark plugs embody the wire conductor and end connectors as a single unit. They are easily attached or disassembled.

5-155. MINOR REPAIR AND REPLACEMENT OF IGNITION SYSTEM COMPONENTS. No minor repair of the ignition system components is recommended. If, upon careful inspection, a component of the system is malfunctioning, the unit should be removed from the airplane and replaced with a new or overhauled unit.

5-156. TROUBLE SHOOTING OF IGNITION SYSTEM.

Trouble	*Probable Cause*	*Remedy*
a. Failure to start.	Inoperative induction vibrator.	Vibrator is controlled by the starting switch; starter may be disconnected for the test by disconnecting starter cable at firewall.
	Defective spark plugs.	Check for fouled plugs, plugs incorrectly gapped, or plugs having cracked insulation. Install reconditioned or new plugs finger-tight and tighten to torque of 300 to 360 inch-pounds.
	Defective magneto ground leads.	Disconnect ground lead from each magneto. Insulate Bosch safety-ground device by inserting heavy paper between ground spring and magneto housing. Attempt to start engine. If engine starts, magneto ground system is defective.

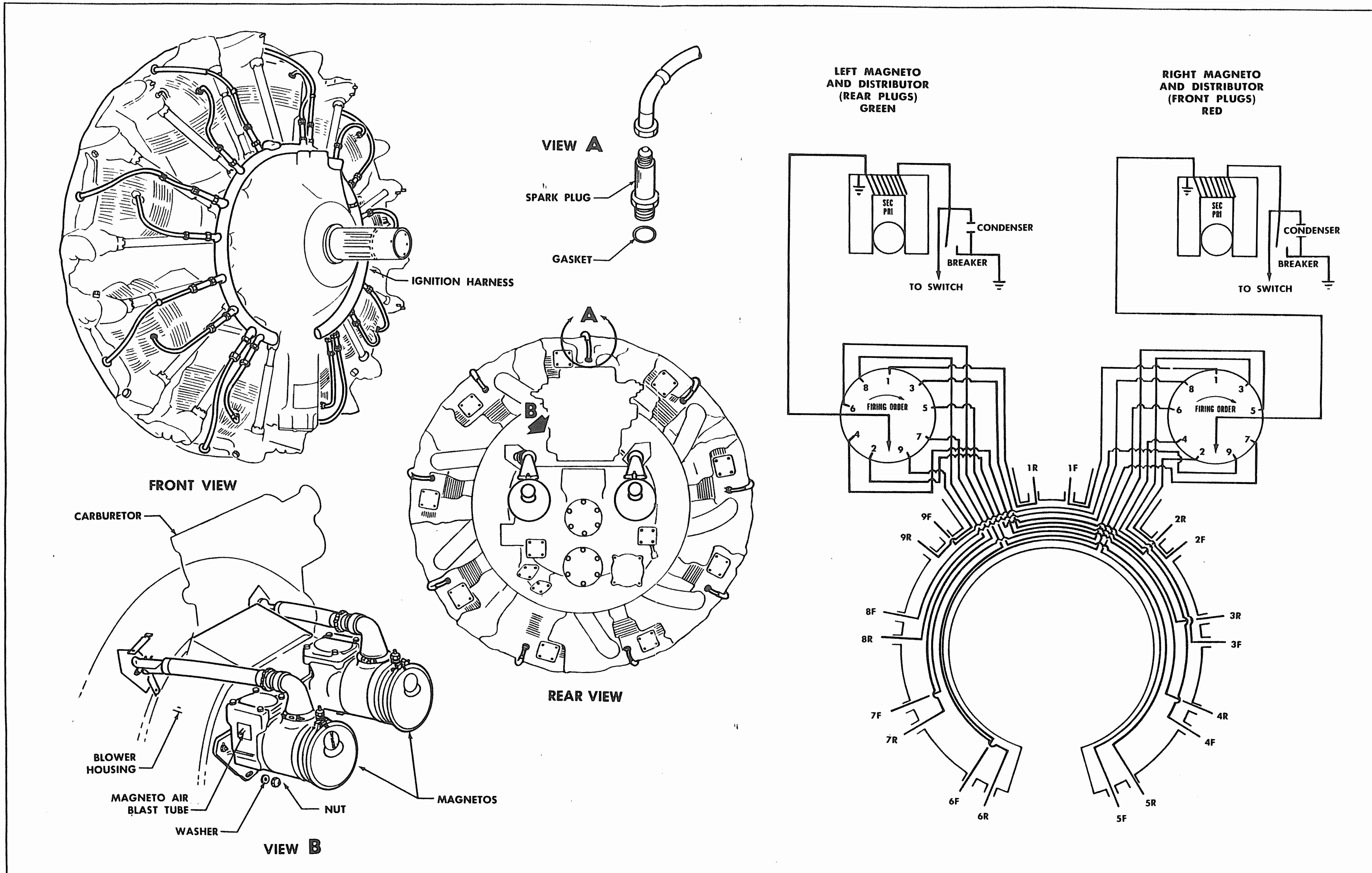

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Figure 5-27. Ignition System Installation

Trouble	*Probable Cause*	*Remedy*
a. *(Continued)*	Defective points or timing.	Replace points or time magneto.
	Moisture or oil in magnetos.	Clean distributor terminal blocks and distributor rotor with unleaded gasoline. Use lint-free cloth.
	Internal magneto trouble.	Replace magneto.
b. Rough running.	Defective spark plugs.	See step a, preceding.
	Defective spark plug lead.	Clean dirty or oily connectors with a dry clean cloth. If necessary, unleaded gasoline or solvent can be used. Replace cracked connectors.
	Moisture or oil in magnetos.	See step a, preceding.
	Defective ignition harness or internal magneto trouble.	Apply continuity check to ignition harness, using low-voltage test light or buzzer circuit. Check continuity from the distributor block electrode to corresponding spark plug lead. If test indicates open circuit, locate break by examining connections or removing spark plug lead. If necessary, replace ignition harness.
	Defective ignition timing.	Check magneto timing.
c. Low power.	Defective spark plugs.	See step a, preceding.
	Defective ignition system.	Replace magneto. Check harness.
d. Rough idling.	Defective spark plugs.	See step a, preceding.
	Defective spark plug leads, connectors.	See step b, preceding.
	Defective magneto.	Replace magneto.
	Defective ignition harness or internal magneto trouble.	See step b, preceding.
	Wrong idle mixture.	Readjust idle mixture.
	Wrong idle rpm.	Reset idle rpm.
e. Magneto remains ON and operative regardless of position of ignition switch.	Open primary lead.	Repair wire.
	Faulty ignition switch.	Remove and replace switch.
f. Magneto remains OFF regardless of position of ignition switch.	Ground in primary lead.	Repair wire.
	Faulty ignition switch.	Remove and replace switch.
	Faulty magneto.	Remove and replace magneto.
	Ground lead removed at magneto.	Reinstall ground lead at magneto.
	Faulty automatic grounding spring at magneto end of ground lead.	Replace automatic grounding spring.

5-157. MAGNETOS. Each magneto incorporates a nine-lobed cam and is a fixed-ignition, four-pole, flange-mounted, inductor-type unit driven directly through a splined coupling. The magneto drive shaft, turning at $1\frac{1}{8}$ engine speed, rotates the nine-lobed cam at $\frac{1}{2}$ engine speed through a $2\frac{1}{4}$ to 1 gear ratio between the inductor rotor and the distributor gear shaft. The magneto housing is fabricated of high-pressure aluminum alloy. Two stationary magnets are cast integrally with the pole shoe assemblies and magneto housing.

5-158. REMOVAL OF MAGNETOS. Before removing either magneto, proceed as follows:

a. Place the fuel mixture control in CUT-OFF position and the ignition switch in OFF position. Turn off the fuel supply to the fuel pump. Remove the front spark plug from No. 1 cylinder.

b. Turn the propeller shaft slowly until the piston in No. 1 cylinder is approximately 90 degrees before top center on the compression stroke. If there is no propeller on the engine, use a propeller shaft turning tool.

c. Install a pivot arm and scale on a piston position indicator. Install the indicator in the front spark plug insert of No. 1 cylinder. Turn the cap on the indicator until the slot is parallel with the vertical axis of the cylinder and the scale is at the right of the slot. Push the pointer to the top of the slot.

CAUTION

Pivot arms and scales are marked with the engine model for which they are intended to be used. Make certain that the correct parts are used.

d. Turn the propeller shaft in the direction of normal rotation until the piston in No. 1 cylinder passes through top center position, indicated by the pivot arm starting to rise in the slot. The slide pointer will

cease to travel as soon as the piston has reached top center. Set the zero degree mark on the scale opposite the pointer reference mark. Turn the propeller shaft in the direction opposite to normal rotation for approximately one-quarter of a revolution. Move the slide pointer until it is opposite the 20-degree mark on the scale. Tap the propeller, or propeller shaft turning tool handle, in the direction of normal rotation until the end of the piston position indicator pivot arm just touches the slide pointer. The bulb in the indicator will illuminate at the contact, signifying that the piston is at 20 degrees before top center. When this position has been determined, move the pointer down, away from the pivot arm to disconnect the circuit through the arm and slide.

e. Once the piston is located at 20 degrees before top center, do not turn the propeller shaft until required to do so as described in step j, following. Do not move the scale.

f. Attach two wire leads to the pin jacks on the piston position indicator. A supplementary battery may be used with the piston position indicator so that the work will not be interrupted if the small battery in the indicator unit wears down before operations are completed. The supplementary battery may be used in parallel with the small battery by connecting the "plus" terminal to the screw on the scale friction spring and grounding the "minus" terminal to any convenient place on the engine.

g. If it has been established that only one magneto requires replacement, check the timing of the satisfactory magneto in accordance with steps h through q, following.

h. Loosen the lockscrew on the clamp securing the breaker box cover to the magneto. Slip the clamp off and remove the breaker box cover.

i. Remove the ground terminal from the magneto and insulate the automatic ground spring with a clean dry piece of cardboard. Disconnect the primary coil lead and reinstall the screw that secured this lead. Connect the alligator attaching clip on one of the wire leads from the piston position indicator to the primary lead screw in the magnet.

j. Turn the propeller shaft several degrees in the direction opposite to normal rotation. The pointer on the piston position indicator should be still at the bottom of the slot. The light in the indicator, connected to the magneto by the wire lead, should be on if the breaker points in the magneto are closed.

k. Turn the propeller shaft very slowly in the direction of normal rotation until the light in the indicator goes out, signifying that the breaker points are beginning to open. Push the slide pointer on the piston position indicator up until it makes contact with the pivot arm and note this position on the scale. If the magneto is correctly timed, the pointer in the piston position indicator will be opposite the 20-degree mark on the scale, No. 1 lobe (illustrated by a red dot) of the breaker cam has just opened the breaker points, and a straightedge laid along the step on the cam will align, within 1/32 of an inch, with the mark on the timing post. Check these conditions.

l. If the straightedge does not align within the prescribed limits, the breaker points should be adjusted. First, move the pointer in the piston position indicator away from the pivot arm to break the circuit and proceed as follows:

m. If the check with the straightedge indicates that the breaker points have started to open in advance of the required timing, jar the propeller in the normal direction of rotation until the straightedge on the breaker cam step aligns with the center of the timing mark. Loosen the two contact bracket locking screws and shift the contact bracket by means of the eccentric head screw until the timing light in the piston position indicator goes out. Secure the contact bracket with its locking screws and recheck the breaker point action to insure that the locking operation did not change the contact position.

n. If the straightedge check indicates that the breaker cam has rotated beyond the desired point, back up the cam, by turning the propeller shaft in the direction opposite to normal rotation, until the breaker cam follower is on the cam dwell preceding No. 1 lobe. Jar the propeller shaft in the direction of normal rotation until the straightedge along the step on the cam aligns with the mark on the timing post. Adjust the contact bracket as described in step m.

o. If the straightedge check described in step k, preceding, is acceptable but the breaker points do not open at 20 degrees before top center, as determined by the timing light, adjust the timing of the magneto as described in step p, following.

p. Set the piston at 20 degrees before top center by turning the propeller shaft until the pivot arm makes contact with the slide pointer, placed opposite the 20-degree mark on the scale. Loosen the attaching screws that secure the magneto to its mounting pad on the supercharger rear housing cover. Tap the magneto gently until the timing light on the piston position indicator goes out. Tighten the magneto attaching nuts. Move the indicator pointer away from the pivot arm in the piston position indicator. Recheck the breaker point action to insure that the magneto has not moved during the tightening of the attaching nuts.

q. When conditions described in the last sentence in step k, preceding, are obtained, the magneto is properly timed.

r. Remove the magneto to be replaced as follows: Remove the ground terminal. Break the lockwire and loosen the spanner nut at the high-tension lead to the magneto plug-in connection elbow. Break the

lockwire and loosen the lockscrew on the ring clamp securing the plug-in connection in the magneto. Remove the connection. Remove the three attaching nuts and washers securing the magneto to the triangular mounting pad on the supercharger rear housing cover. Pull the magneto straight out from the rear cover until the magneto drive shaft and coupling are completely disengaged from the accessory drive mechanism.

5-159. INSTALLATION OF MAGNETOS. Before installing a magneto, check to see that the model to be installed is the one called for on the engine data plate. Make certain that the magneto mounting pad on the supercharger rear housing cover is clean and dry. Prepare the magneto to be installed as follows:

a. Wipe the mounting pad parting surface with a clean dry cloth. Lightly coat the magneto drive shaft coupling with anti-seize compound, Specification JAN-A-669.

b. Remove the magneto breaker cover. Ascertain that the magneto cam is compensated for the master rod location given on the engine data plate.

CAUTION

Make certain that the terminal assembly is installed properly, or insulate the automatic ground spring of the magneto with a clean piece of thin cardboard.

c. A new or overhauled magneto should have correct internal timing. Check the internal timing as follows: Turn the magneto drive shaft in the direction of normal rotation as indicated by the arrow on the breaker cam until the No. 1 lobe, identified by a red dot, is about to open the breaker points. When the breaker points are about to be opened by No. 1 cam lobe, the step on the cam must align within 1/32 of an inch with the mark on the timing post. Check the alignment with a straightedge. If the straightedge does not indicate the proper alignment, use another magneto.

5-160. When correct internal timing of the magneto has been verified, mount the magneto on the engine. Rotate the magneto drive shaft through the full length of the slots in the mounting flange and check the breaker point action. When the magneto drive shaft is rotated to the extreme right position, the breaker points should be closed, and when rotated to the extreme left, the breaker points should be open. If the points do not function in this manner, proceed as follows:

a. Remove the magneto from the engine. Turn the magneto drive shaft in the direction of normal rotation until the cam has made one complete revolution and the step will again line up with the mark on the timing post. The relative positions of the magneto drive shaft coupling and the breaker cam will now be slightly altered. Mount the magneto on the engine and recheck the breaker point action.

Note

The preceding operations may be performed four times, giving four different positions of the coupling in relation to No. 1 cam lobe.

b. If the required results are still not obtained, remove the magneto from the engine. Remove the cotter pin from the magneto drive shaft coupling and remove the nut from the end of the drive shaft. Remove the coupling and reinstall it one spline in advance of its former position. Reinstall the coupling nut finger-tight. Mount the magneto on the engine and check the breaker point action. Repeat this procedure until the breaker points open and close when the magneto is moved through the length of its mounting flange slots. Remove the magneto, tighten the coupling nut, and install the cotter pin.

c. Remount the magneto on the engine. Install two washers and attaching nuts, leaving them loose enough to allow the magneto to be shifted. Disconnect the primary coil lead and reinstall the screw that secured the lead. Connect the indicator to the primary coil lead screw in the magneto.

5-161. Time the new magneto to the engine and synchronize the timing of both magnetos as follows:

a. Turn the propeller shaft opposite normal rotation for a few degrees. Tap it in the direction of normal rotation until the timing light for the correctly timed magneto just goes out. If the timing light connected to the replacement magneto does not go out at the same time that the other light goes out, tap the replacement magneto in the direction that will open its points. When the light for the replacement magneto fades out, tighten the magneto attaching nuts.

b. Turn the propeller shaft opposite normal rotation for a few degrees, then tap in the direction of normal rotation and check the synchronization of the magnetos by observing the timing scale when both timing lights go out. Both magnetos must be timed to 20 degrees before top center, and the timing lights should, therefore, go out simultaneously.

c. When satisfactory adjustments have been made, install the remaining washers and attaching nuts and tighten all the nuts. Disconnect the timing light wires from both magnetos. Reconnect the primary coil leads. Remove the terminal assembly or the cardboard insulation from each magneto. Attach the ground leads. Reinstall the breaker box covers. Insert the plug-in connection elbow in the replacement magneto. Install the ring clamp and tighten the lockscrew. Tighten the nut at the plug-in elbow on each magneto. Remove the

piston position indicator from the spark plug insert in No. 1 cylinder. Reinstall the spark plug and connect the ignition lead contactor. If the propeller shaft turning tool has been used, remove the tool.

5-162. IGNITION HARNESS ASSEMBLY. A radio-shielded ignition harness is installed on the nose section of each engine. The harness carries the electrical leads from the distributor blocks to the spark plugs in each cylinder.

5-163. REMOVAL OF IGNITION HARNESS ASSEMBLY. Remove all parts of the engine installation interfering with the accessibility of the ignition harness. Then proceed as follows:

a. Remove the two intercylinder air deflectors located between cylinders No. 1 and 2 and 1 and 9.

b. Disconnect all spark plug leads, using a wrench.

c. Break the lockwire and loosen the ignition lead conduit nuts at the plug-in elbow, using a 2- to 4¾-inch spanner wrench.

d. Release the lockscrews on the magneto plug-in elbow clamp. Remove the clamps and withdraw the elbows from the magnetos.

e. Release the conduits from the supporting brackets on the engine.

f. Disconnect the manifold mounting lugs that are attached to the brackets secured by the crankcase front section to main section attaching screws.

g. Lift the entire ignition harness assembly from the engine and remove.

5-164. INSTALLATION OF IGNITION HARNESS ASSEMBLY.

a. Place the ignition harness assembly in position, locating the manifold on the crankcase front section and laying the conduits back along the top of the engine between cylinders No. 1 and 2 and 1 and 9.

b. Attach the manifold mounting lugs to the supporting brackets on the flange at the rear of the crankcase front section.

c. Insert the plug-in connections in the magneto and loosely install the attaching ring clamps.

d. Attach the main conduits to the appropriate supporting Y-shaped brackets on the engine.

Note

Right and left Y-shaped brackets are not interchangeable.

e. Using a 2- to 4¾-inch spanner wrench, tighten the conduit connection nuts at the manifold and at the magneto plug-in elbows.

f. Remove the protectors and install the spark plug leads in the spark plugs with a wrench.

g. Attach all rear spark plug lead clips to the clamps on the appropriate pushrod housings.

h. Tighten all the nuts. Lockwire the conduit connection nuts.

i. Install the intake pipes on cylinders No. 1 and No. 9, and install the deflectors that were removed.

5-165. SPARK PLUGS. Two spark plugs are installed in each of the nine cylinders. The following types are approved and may be used interchangeably in sets, but should not be mixed: BG RB19R-2 and Champion R-37S-1. Individual leads extend from the distributor block of the magneto to each spark plug.

5-166. REMOVAL OF SPARK PLUGS. Disconnect the ignition lead from the spark plug, using the special wrench. To avoid damaging the elbow, hold it with one hand while using the wrench. Withdraw the lead and install a protector, part No. AN 4060-1, on the contactor. Remove the spark plug, using a suitable socket wrench. Remove the spark plug seal. Install a dummy plug in the spark plug hole in the cylinder. Use one of the following procedures to facilitate removal of a seized or tight plug:

a. Treat the base of the spark plug with a mixture of ether and penetrating oil to loosen hardened carbon deposits.

b. Remove the spark plug opposite the tight plug and insert a tube through the hole thus exposed. While the cylinder head is still hot, direct CO_2 through the tube onto the sticking spark plug.

c. Improvise a conical adapter incorporating, at its apex, a hole of proper size to accommodate the spark plug. Install this adapter over the tight plug. Place the funnel of a CO_2 cylinder in the adapter. While the cylinder head is still hot, discharge the CO_2.

d. Apply vibration to the spark plug by hand or with an electric or air-driven vibrating hammer, simultaneously exerting pressure to unscrew the plug.

e. To remove a broken spark plug shell, insert a screwdriver between the ground electrodes and back out the plugs, or use an "Ezy-Out" if necessary.

5-167. INSTALLATION OF SPARK PLUGS. Remove the dummy plug from the spark plug insert. Check the threads of the insert for cleanliness, and make certain that the old seal has been removed. Use a new copper seal not exceeding .068- to .095-inch thickness. Coat the bottom threads of the spark plug with Specification JAN-A-669, keeping it from the electrodes. Install the plug, running it all the way down with the fingers. Tighten with a spark plug socket wrench and torque wrench. Torque to 300 to 360 inch-pounds. Remove the AN protector from the ignition lead contactor. Connect the ignition lead to the spark plug.

5-168. IGNITION SYSTEM BOLT TORQUE VALUES. For information on ignition system bolt torque values, see paragraph 5-263.

5-169. ENGINE STARTING SYSTEM.

5-170. DESCRIPTION. *(See figure 5-28.)* The engine starting system consists of two direct-cranking series inter-pole-type starters, one installed on the accessory drive case of each engine; four switches, located on the pilot's right overhead electrical panel; two relays, located in the firewall junction boxes; and two induction vibrators, installed in the engine accessory sections.

5-171. MINOR REPAIR AND REPLACEMENT OF ENGINE STARTING SYSTEM COMPONENTS. No minor repair of engine starting system components is recommended except for the replacement of starter brushes. If, upon careful inspection, a component of the system is malfunctioning, the unit should be removed from the aircraft and replaced with a new or overhauled unit.

5-172. TROUBLE SHOOTING OF ENGINE STARTING SYSTEM.

Trouble	*Probable Cause*	*Remedy*
a. Starter operates but fails to crank engine.	Clutch not properly set.	Remove and replace starter.
	Worn or scored clutch discs.	Remove and replace starter.
	Starter rotation incorrect.	Remove and replace starter.
	Starter jaw does not advance into engagement.	Remove and replace starter. Check for worn oil seal and binding action between screw shaft and spline nut.
	Oil on clutch plates.	Remove and replace starter.
b. Starter fails to operate or operates at too low a speed.	Low voltage in power source.	Check power source for correct voltage.
	Loose, dirty, or incorrect internal or external electric connections.	Clean or replace connections.
	Control switch or relay inoperative.	Replace control switch or relay.
	Binding, worn or improperly seated brushes.	Service brushes.
	Dirty commutator.	Clean commutator.
	Rough, pitted, or scored commutator.	Remove and replace starter.
	Commutator not concentric.	Remove and replace starter.
	Shorted, grounded, or open armature.	Remove and replace starter.
	Grounded or open field coil.	Remove and replace starter.
	Worn, broken, or incorrectly lubricated bearings and gears.	Remove and replace starter.
c. Excessive wear and arcing of brushes.	Binding, worn, or incorrectly seated brushes.	Service brushes.
	Dirty commutator.	Clean commutator.
c. *(Continued)*	Rough, pitted, or scored commutator.	Remove and replace starter.
	Commutator not concentric.	Remove and replace starter.
	Shorted, grounded, or open armature or field coils.	Remove and replace starter.
	Brush spring tension incorrect.	Remove and replace starter.
	Worn motor bearings.	Remove and replace starter.
d. Induction vibrator fails to function.	Unit not grounded.	Check mounting bolts; bolts must make good surface-to-surface contact and be securely fastened.
	Loose or disconnected positive cable.	Make certain terminal clip is secured to positive cable and positive terminal post.
	Loose or disconnected magneto cables or terminals.	Check to see that cables are secured.
	Internal open circuits, excessively worn or pitted vibrator points, or improper relay openings.	Remove and replace induction vibrator.

5-173. STARTERS. *(See figure 5-29.)* A concentric electric, direct cranking, non-reversible, series interpole starter is installed on the accessory drive case of each engine. The engine starter gear meshes with the rear accessory drive gear of the engine and is splined onto the shaft of the engine starter jaw. Meshing of the starter jaw and the engine jaw is performed when the motor is energized by actuation of the starter switches on the overhead electrical panel in the flight compartment. There are no provisions for hand-cranking the engines. The rotation, viewing the commutator end, is clockwise. The starter operates under rated load at 50 rpm with 24 volts d-c input. It has a clutch torque setting of from 800 to 900 foot-pounds. Either a Jack & Heintz or Eclipse starter is installed. The two types are interchangeable.

Note

When using auxiliary power to start the engine do not operate the starter continuously for more than one minute. Allow it to cool for one minute before making another attempt to start the engine. After the second and succeeding attempts, allow five minutes for cooling. Where the batteries in the aircraft itself are the sole source of power for cranking the engine, continuous operation of the starter should be limited to 30 seconds with 30-second intervals for cooling and battery build-up.

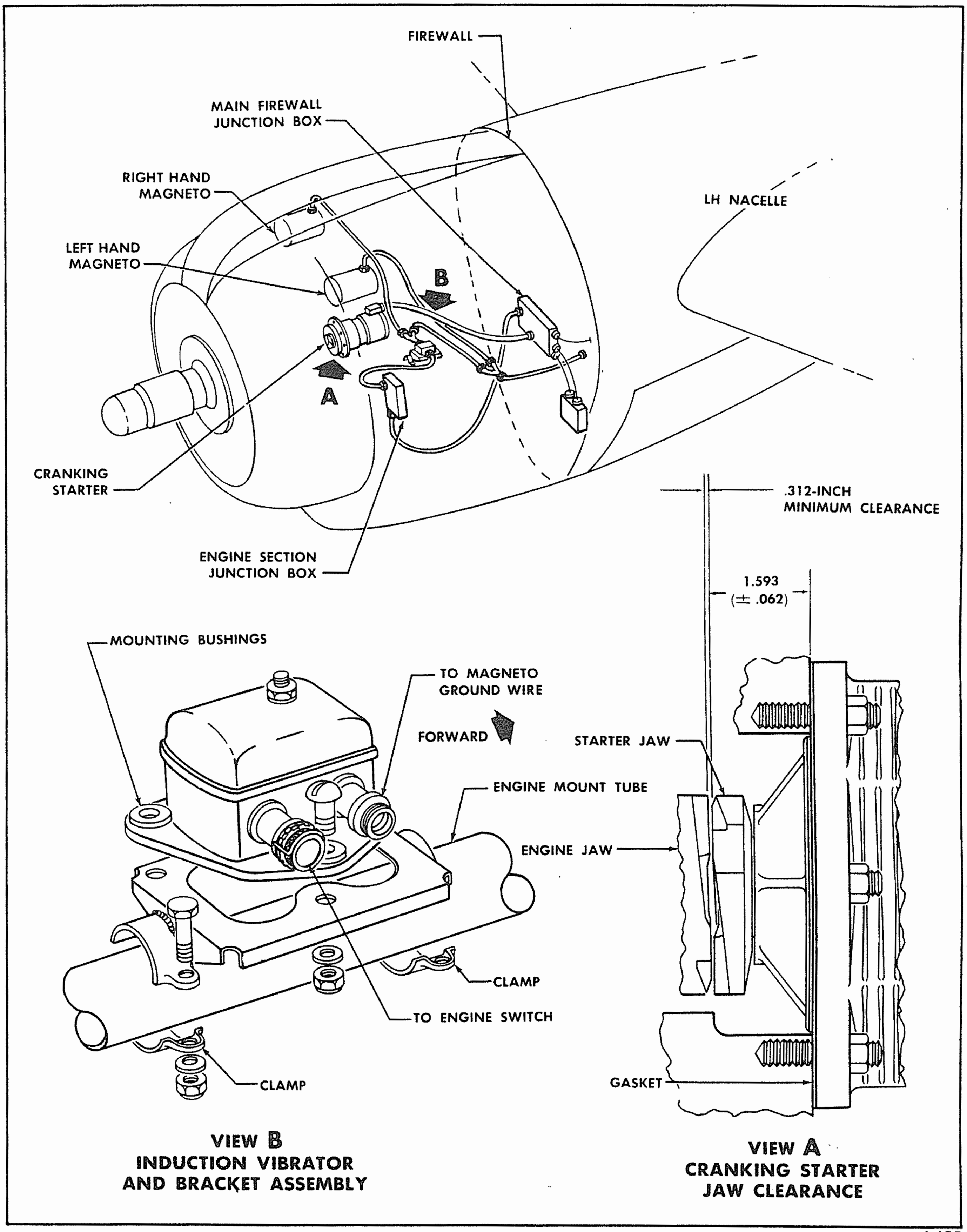

Figure 5-28. Engine Starting System

5-174. INSPECTION OF STARTER. Without removing the starter from the aircraft, remove the cover from the brush inspection openings. Examine the under side of the cover for presence of oil and particles of copper, solder, carbon, insulation, or other foreign matter. Absence of such material usually indicates that the motor is in satisfactory operating condition. Presence of oil or other foreign matter indicates faulty parts in the starter, and that starter should be removed for overhaul.

a. Check brush spring tension with a standard spring scale of suitable range. The end of the spring normally rests on top of the brush. When it is 1/8 inch above the top of the brush, the tension should be between 20 and 24 ounces. If one or more springs are weak, the starter must be removed for replacement of unserviceable springs.

b. Check for free fit of brushes in the brush box. Binding brushes and brush boxes should be wiped clean with a cloth moistened with unleaded gasoline or other suitable solvent. To insure satisfactory operation between inspection periods, replace worn brushes before the maximum wear limit is reached. The new brushes must be 1/2 inch in length and minimum brush length is 11/32 inch.

c. When replacing a worn brush, it is necessary to seat the brush properly by inserting a strip of No. 0000 sandpaper between the brush and commutator, with the sanded side next to the brush, and pull in the direction of rotation. The operation should be repeated until the brush is 75 per cent seated. Use caution when installing brushes by being sure that the insulation around the brush leads is protecting the leads from the motor housing. If the commutator is rough or dirty, smooth it with No. 0000 sandpaper.

CAUTION

Do not use coarse sandpaper or emery cloth. Remove sand or metal particles with compressed air.

5-175. REMOVAL OF STARTER.

a. Remove the starter lead cover and disconnect the starter lead.

b. Remove the nuts that secure the starter to the engine accessory case mounting pad, and remove the starter.

c. Place a cover over the engine opening.

5-176. INSTALLATION OF STARTER. *(See figure 5-28.)* A starter that has been in storage less than one year may be installed without further preparation. Starters left in storage for a period exceeding one year must be given a partial overhaul, including cleaning, inspection, lubrication, and test prior to installation.

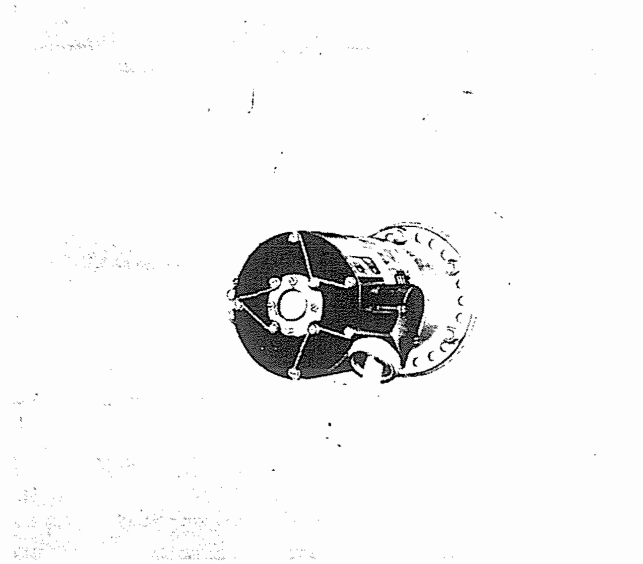

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Figure 5-29. Starter Installation

a. Remove the dust cover from the drive end of the starter, and remove the cover from the engine opening. Examine both the starter jaw and the engine jaw to make certain that they are the same size, have the same number of teeth, and are designed for the same rotation.

b. Remove the cover from the brush inspection openings. Check brushes to make certain that they move freely in the brush boxes. Brush boxes and brushes that bind should be wiped clean with a cloth moistened with unleaded gasoline or other suitable solvent. Check to make certain that all internal wiring connections are clean and tight. Assemble the cover over the inspection openings and tighten the screws.

c. Rotate the starter jaw or push it in until it is in the fully retracted position; then measure the distance from the outermost face of the flange of the baffle plate to the outermost edge of the starter jaw teeth. This dimension must be 1 19/32 (±1/64) inches. If teeth are worn or milled so that this dimension is not correct, replace the starter jaw. If this fails to correct the trouble, return the starter for overhaul.

d. Face the starter jaw and rotate the jaw clockwise until it has moved to its fully advanced position; then check to see that the jaw has moved a minimum distance of 5/16 inch from its fully retracted position. If jaw travel is less than 5/16 inch, return the starter for overhaul.

e. With the engine seal removed, measure the distance from the surface of the engine mounting pad to the outermost edge of the engine jaw. The dimension must be 1 11/16 (±1/64) inches. If the dimension is not correct, adjustment to the engine jaw position is necessary.

Note

Under all conditions, a clearance of not less than 0.055 inch must be maintained between the outermost edge of the starter and the engine jaw teeth when the starter jaw is in the fully retracted position.

f. Wipe the engine pad clean, replace the seal, and push the starter jaw to the fully retracted position.

Note

Remove paint, dirt, grease, etc., from under at least three of the nuts on the flange to insure proper grounding.

g. Mount the starter on the engine studs, giving consideration to the location of the motor terminal to facilitate electrical connections; screw the mounting nuts up tight and lock them in place. Connect the starter lead, and replace and safety the starter lead cover.

5-177. INDUCTION VIBRATOR. *(See figure 5-30.)* The induction vibrators, which are designed for use as an auxiliary booster source of ignition to facilitate starting the engines, are installed in the engine accessory sections. The vibrators furnish an interrupted battery current to the primary magneto coil when the engine starting selector switch is turned to the desired engine and the starter and safety switches are depressed. The interrupted current in the primary winding of the magneto produces a high voltage in the secondary magneto coil; this voltage is supplied to the spark plugs through the distributor of the right magneto only. This action takes place during the entire period that the magneto contact points are open. The booster system is put into operation by turning the ENGINE STARTING SELECTOR switch to the engine to be started, and then depressing the starter and safety switch. When the engine is firing under the regular magneto spark the vibrator is no longer needed.

5-178. REMOVAL OF INDUCTION VIBRATOR.

a. Disconnect the electrical leads.

b. Remove the mounting bolts and remove the vibrator assembly.

5-179. INSTALLATION OF INDUCTION VIBRATOR. Reverse the removal procedure.

5-180. STARTER RELAY. A Type B-8A starter relay is installed in each firewall junction box, accessible from within the nacelle. Each relay is connected in its respective circuit to the starter safety switches. When the switches are turned on, the starter relay is actuated, connecting the circuit between the airplane bus and the starter motor. Circuit protection between the relay and the starter in each system is provided by a current limiter, also located in each firewall junction box.

5-181. REMOVAL OF STARTER RELAY.

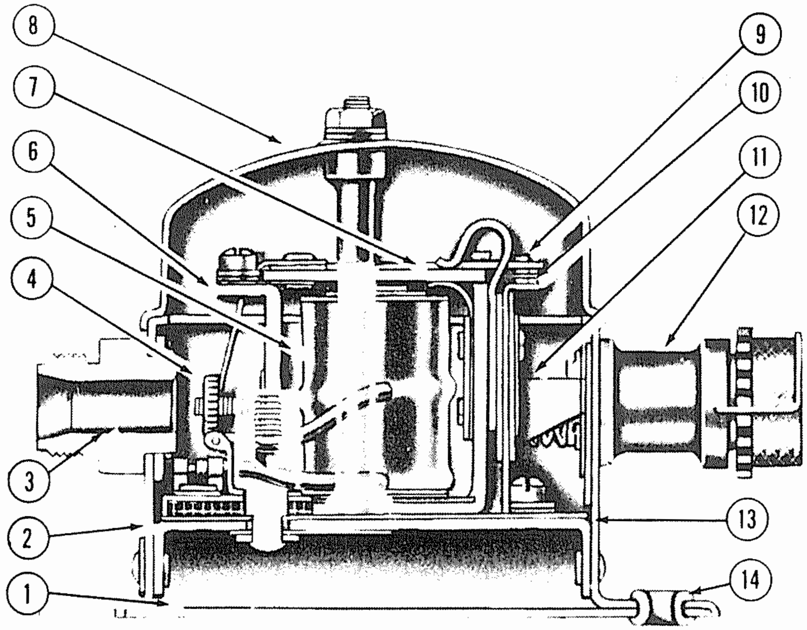
a. Disconnect the busses.

b. Remove the attaching bolts and remove the starter relay.

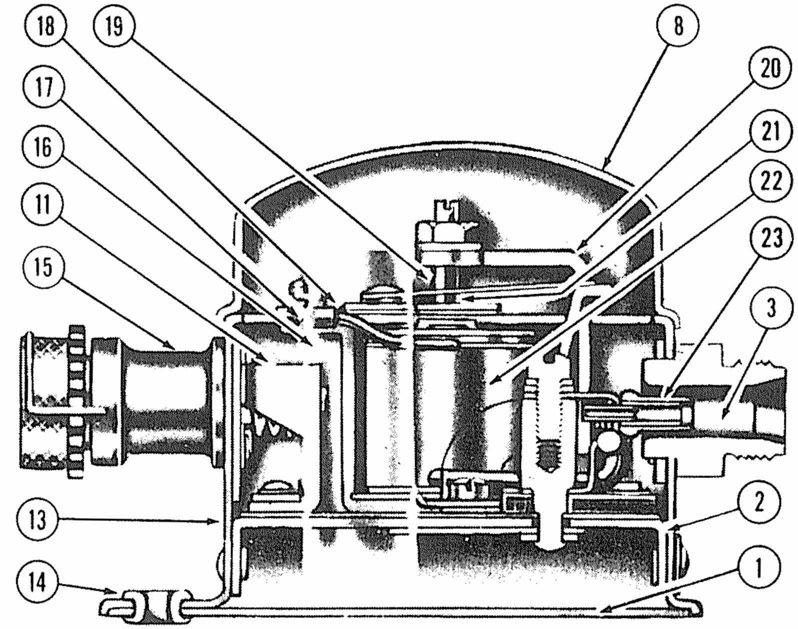
5-182. INSTALLATION OF STARTER RELAY. Reverse the removal procedure.

5-183. ENGINE STARTING SYSTEM SWITCHES. The engine starting system switches, located on the pilots' right overhead electrical panel, consist of the following: an engine starting selector switch, a primer switch, the starter switch, and a safety starter switch. To energize a starter, the selector switch must be set for the engine desired, the starter safety switch depressed, and the single-pole, single-throw starter switch must be closed. The starter safety switch, below the starter switch, is installed to prevent inadvertent cranking of the engine and must also be closed before the starter switch will function.

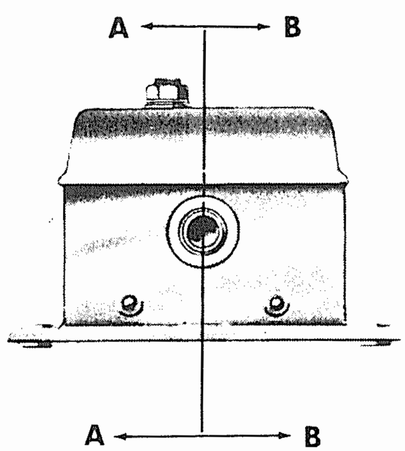
5-184. ENGINE STARTING SYSTEM BOLT TORQUE VALUES. For information on the engine starting system bolt torque values, see paragraph 5-263.



SECTION A-A



SECTION B-B



1. External Base Plate	13. Housing
2. Internal Base Plate	14. Mounting Bushings
3. "+" Terminal Outlet	15. "IGN SW" Outlet
4. Knurled Tension-Adjusting Nut	16. Vibrator Yoke
5. Relay Coil	17. Vibrator Condenser
6. Relay Yoke	18. Vibrator Armature Plate
7. Relay Armature Plate	19. Stationary Vibrator Point
8. Top Cover	20. Contact Bracket
9. Movable Relay Point	21. Movable Vibrator Point
10. Stationary Relay Point	22. Vibrator Coil
11. Terminal Strip	23. "+" Terminal Strip
12. "MAG" Outlet	

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Figure 5-30. Induction Vibrator

5-185. AIR INDUCTION AND EXHAUST SYSTEMS.

5-186. DESCRIPTION. *(See figure 5-31.)* The air induction system provides either ram air or non-ram air to the carburetor. Ram air is transmitted through an airscoop. By the operation of a door in the airscoop elbow, the ram air supply is closed off and heated non-ram air is permitted to enter from the engine section. Ram air is directed to the carburetor through an undercowl air intake scoop, located in the top segment of the anti-drag ring, and then through the carburetor airscoop elbow to the carburetor. The carburetor airscoop elbow is connected to the carburetor top deck adapter by a rubber sleeve. It incorporates a cable-operated preheat door that controls the temperature of the carburetor air by regulating the amounts of cold ram air and heated non-ram air that are directed to the top deck of the carburetor. The preheat door control levers are located in the cockpit on the right side of the control pedestal. The control levers incorporate a lock lever so that they may be held in any desired position. The carburetor air temperature is measured by an electric resistance-type thermometer bulb located on the aft side of the carburetor airscoop, directly above the carburetor. The thermometer bulbs for the two engines are connected electrically to two gages installed on the main instrument panel. The gages are calibrated between $-50°C$ and $+150°C$ ($-58.0°F$ and $+302.0°F$).

5-187. Ejection-type exhaust stacks are installed to provide for the discharge of the exhaust gases on the outboard side of each nacelle and to gain additional thrust *(see figure 5-32).*

5-188. CARBURETOR AIRSCOOP. The carburetor airscoop is a duct through which air flows from under the forward upper edge of the engine cowling to the carburetor top deck. The duct is covered by a fairing, which forms the top section of the engine section cowling. A rubber seal is provided between the anti-drag portion of the duct and the airscoop elbow. The top section of the accessory cowling that includes the airscoop elbow is attached to the inner ring and to the accessory cowling trailing edge support ring assembly. A small, round door at the top center of the airscoop fairing, held in position with Camloc fasteners, provides access to the preheat door operating mechanism and cables. The airscoop elbow incorporates piping connections for the alcohol anti-icing system, the fire extinguishing system, and the carburetor air intake temperature bulb.

5-189. REMOVAL OF CARBURETOR AIRSCOOP.

a. Place the engine cowl flap selector switch in the OPEN position and extend the flaps to the full open position.

b. Remove the upper segments of the engine accessory cowling after loosening the Camloc fasteners.

c. Remove the anti-drag ring by unlocking the Camloc fasteners for the three doors on each of the anti-drag ring segments that expose the latch assemblies. Using a screw driver, lift the plunger located at the center of the hex head of the latch assembly. Rotate the hex head of each latch assembly to the unlocked position, using a 3/4-inch socket wrench. Remove the two lower segments of the anti-drag ring first. The top segment may then be lifted free.

d. Loosen the Camloc fasteners and remove the small round access door on top of the aft airscoop fairing. Open the latch on the quick-disconnect-type pulley, located on the carburetor air (preheat) door. Disconnect the control cables.

e. Disconnect the carburetor alcohol anti-icing and fire extinguishing piping at the airscoop elbow.

f. Disconnect the electrical lead for the carburetor intake air temperature bulb at the airscoop elbow.

g. Loosen the clamp at the rubber sleeve attachment of the airscoop duct to the carburetor top deck adapter.

h. Loosen the five Camloc fasteners and remove the four bolts that attach the airscoop assembly to the aft end of the inner ring.

i. Loosen the Camloc fasteners at the aft end of the airscoop fairing, which attach the assembly to the cowling trailing edge support ring.

j. Working through the open cowl flaps and the engine accessory section, remove the bolts attaching the carburetor airscoop duct fairing to the inner ring.

k. Remove the carburetor airscoop and top fairing assembly.

5-190. INSTALLATION OF CABURETOR AIRSCOOP. Reverse the removal procedure, with the following additional step: Adjust the carburetor air (preheat) door control cables (see paragraph 5-250).

5-191. ADJUSTMENT OF CARBURETOR AIRSCOOP DOOR. For a complete description of the adjustment procedure to be followed in rigging the carburetor airscoop door, see paragraph 5-250.

5-192. EXHAUST STACKS. *(See figure 5-32.)* The exhaust stacks are fabricated of corrosion-resistant steel. Each left and right engine exhaust stack system is composed of six separate exhaust stack assemblies. On the left engine each of the stack systems, which extend from No. 1 and No. 2 cylinders and from No. 3, No. 4, and No. 7 cylinders to the common exhaust outlets, are composed of two exhaust stack sub-assemblies bolted together by flanges. The other two stack assemblies extend from cylinders No. 5 and No. 6 and from cylinders No. 8 and No. 9 to the common exhaust outlet location on the outboard side of the

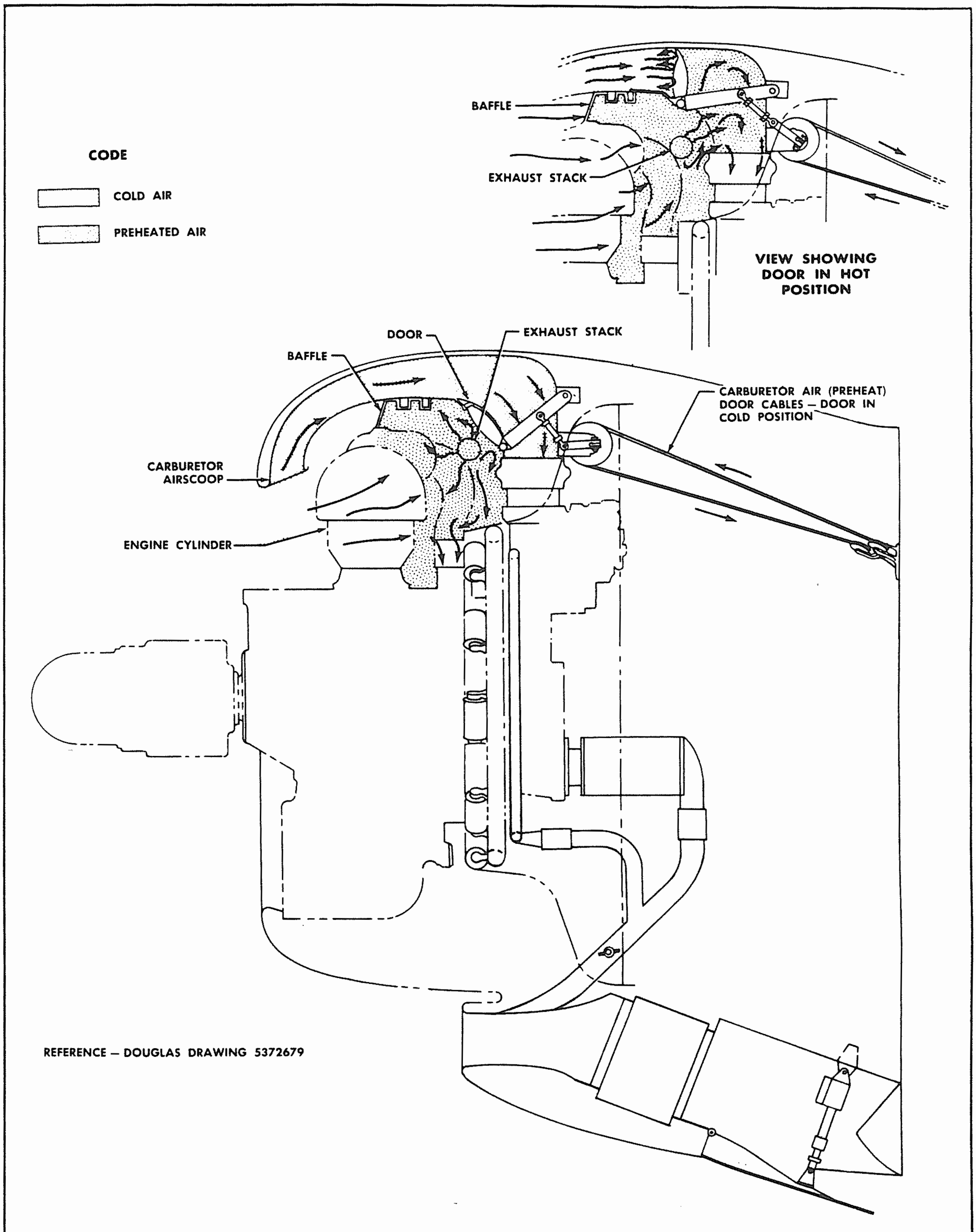


Figure 5-31. Air Induction System

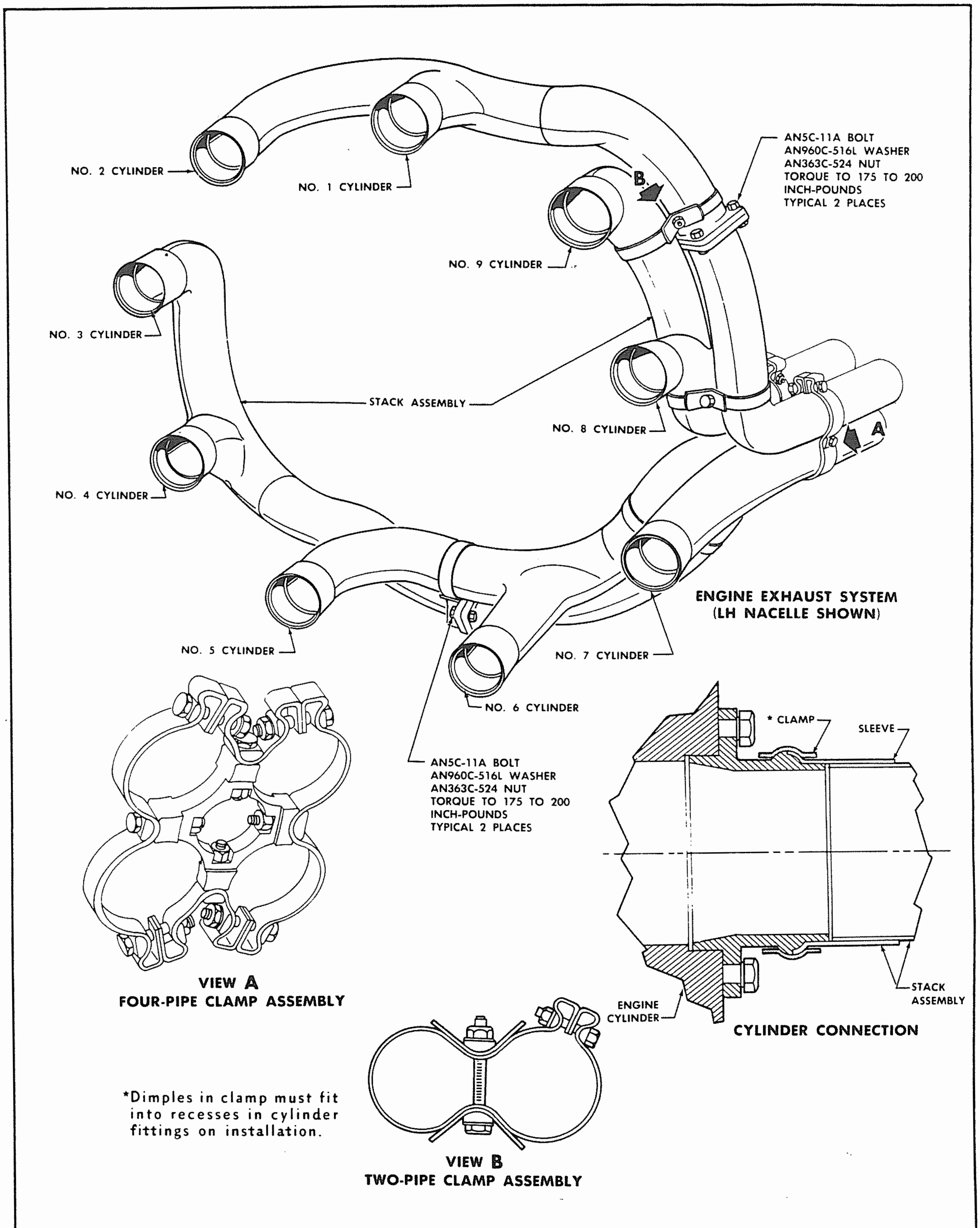
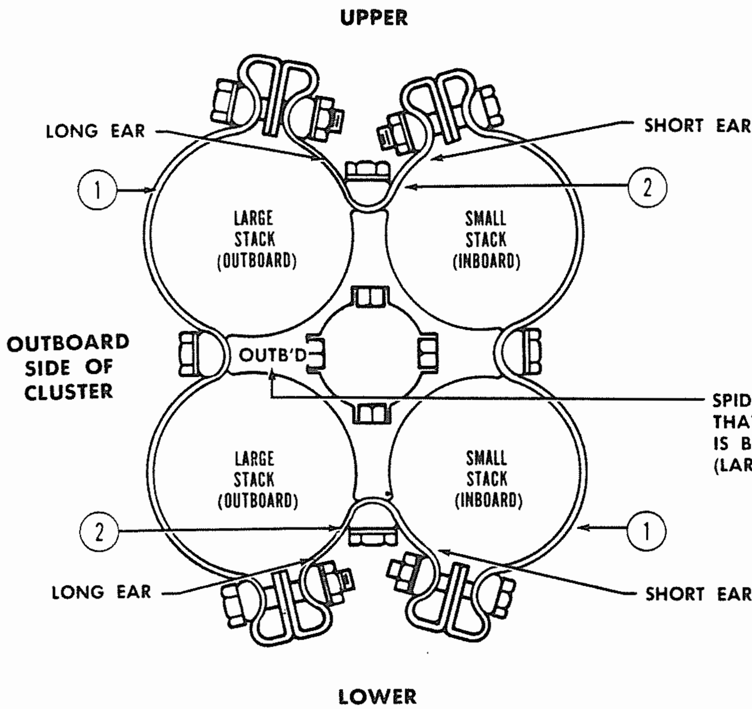


Figure 5-32. Engine Exhaust System



Notes:

1. The two long straps are the same length and must be installed on the outboard and inboard sides of stack cluster (*not* upper and lower sides).
2. Each short strap has a long ear and a short ear. Install with short ear adjacent to small stack and long ear adjacent to large stack as shown. Tighten four center bolts of clamp assembly first. Then tighten four bolts at strap ends so that clamp assembly can be moved slightly when tapped lightly with a plastic or fiber mallet.

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Figure 5-33. Exhaust Stack Clamp Installation

nacelle. The stacks are held rigidly together by one four-stack clamp, located near the common exhaust outlet location, and by four two-stack clamps. All clamps are located on the wear bands and between the stops on the stack assemblies. On the left engine, the two-stack clamps are located between No. 5 and No. 6 cylinders, No. 6 and No. 7 cylinders, No. 7 and No. 8 cylinders, and No. 8 and No. 9 cylinders. All of the right engine exhaust stack assemblies are the same assemblies as those used on the left engine. They are installed differently, however, so as to have the common exhaust outlet located always on the outboard side of each nacelle. On the right engine, each of the stack systems that extend from No. 7 and No. 8 cylinders, and from No. 9, No. 1, and No. 4 cylinders to the common exhaust outlets are composed of two exhaust stack sub-assemblies bolted together by flanges. The other two stack assemblies extend from cylinders No. 2 and No. 3, and from cylinders No. 5 and No. 6 to the common exhaust outlet location on the outboard side of the nacelle. The stacks are held rigidly together by one four-stack clamp located near the common exhaust outlet and by four two-stack clamps, which are located on the right engine as follows: between No. 2 and No. 3 cylinders, No. 3 and No. 4 cylinders, No. 4 and No. 5 cylinders, and No. 5 and No. 6 cylinders.

5-193. REMOVAL OF EXHAUST STACKS.
(See figure 5-33.)

a. Disconnect the cowl flaps from the actuating jackscrews and run a piece of wire through the jackscrew ends to prevent them from turning. (Any turning of the jackscrew shaft will necessitate rerigging the system.)

b. Remove the four-stack clamp securing the cluster of exhaust stacks at the lower outboard side of the engine.

c. Remove the four two-stack clamps that secure the adjacent stack assemblies.

d. Remove the flange bolts that secure the exhaust stack sub-assemblies.

e. Loosen the clamps attaching the exhaust stacks to the cylinders.

f. Remove the stack assemblies through the cowl flap opening.

5-194. MINOR REPAIR AND REPLACEMENT OF EXHAUST STACKS.

a. Examine the exhaust stack assemblies, including the flanges and all the clamps, for cracks or evidence of excessive burning.

b. Small cracks, holes, or other damage of a minor nature may be repaired by welding.

c. Check the flange nuts for looseness.

5-195. INSTALLATION OF EXHAUST STACKS. Reverse the removal procedure and observe the following precautions:

a. On installation of all clamps, including the four-stack clamp, the two-stack clamps, and the clamps in-

stalled at each cylinder, it is important that the attaching bolts and nuts are not overtightened.

Note

Dimples in the engine-to-exhaust stack clamps must fit into the recesses in the cylinder fittings on installation.

b. When a clamp has been installed properly, it should be possible to move it slightly by tapping with a small plastic or fiber mallet.

Note

On the four-stack outlet clamps, tighten the four center bolts of the clamp assembly first. Then tighten the four bolts at the strap ends so that the clamp assembly can be moved slightly when tapped lightly with a small plastic or fiber mallet.

c. It may be necessary to tap the clamp straps with a mallet to make them fit the stacks on which they are installed. All clamps should be checked for fit and tightness after the first engine run.

d. Normally the stacks are connected to the cylinders first, then the two-stack clamps are installed, and finally the four-stack clamp. However, if any of the clamps cannot be installed easily in this sequence, the other clamps should be loosened and then all clamps tightened simultaneously.

Note

Rotate the straps on the two-way clamp to obtain maximum clearance from the exhaust trough.

5-196. AIR INDUCTION AND EXHAUST SYSTEM BOLT TORQUE VALUES. For information on the air induction and exhaust system bolt torque values, see paragraph 5-263.

5-197. PROPELLERS.

5-198. DESCRIPTION. *(See figure 5-34.)* The two Hamilton Standard hydromatic full-feathering type propellers are 11 feet 6¼ inches in diameter. Each propeller consists of three blades and a hub. Constant propeller speeds are maintained by a cable-controlled governor, which is mounted on the engine nose section. An electrically driven propeller feathering pump is installed in the outboard side of each nacelle, aft of the firewall, which pumps oil from the engine oil tank through the governor to the propeller for feathering. The propeller low-pitch angle is 18 degrees and the feathered angle is 88 degrees.

5-199. MINOR REPAIR AND REPLACEMENT OF PROPELLERS AND ACCESSORIES. Components of the propeller system, such as feathering pump switches, relays, governors, and propeller assemblies, which are not functioning properly, should be removed and replaced with new or overhauled units. The pipes, fittings, and wiring should be inspected for general condition, security of mounting, and evidence of leakage.

5-200. TROUBLE SHOOTING OF PROPELLERS.

Trouble	*Probable Cause*	*Remedy*
a. Inability to attain take-off rpm on the blocks.	Wrong setting of governor or incorrect rigging of control system.	Reset governor or rerig.
	Improper adjustment of low-pitch stop ring in propeller dome assembly.	Reset stop ring.
	Low engine power.	See paragraph 5-28.
	Erroneous reading on tachometer.	Replace instrument.
b. Overspeeding on take-off.	Wrong setting on governor and/or incorrect rigging of control system.	Rerig and/or reset.
	Insufficient exercise of propeller mechanism prior to take-off.	Move engine rpm control through constant-speed range with engine running.
	Opening throttle too rapidly.	Advance throttle slowly.
	Poor or incorrect seals between distributor valve and propeller shaft, and between governor base and engine mounting pad.	Install correct seals.
	Sticky governor pilot or relief valve.	Disassemble and clean.
b. *(Continued)*	High engine transfer ring leakage.	Replace rings according to engine manufacturer's specification.
	Erroneous reading on tachometer.	Replace instrument.
c. Poor synchronization.	Ignition trouble.	Check with ignition tester or megger.
	Poor carburetion.	Check carburetor.
	Backlash in governor control system.	Rerig control system.
	Sluggish pilot valve.	Disassemble and clean.
	Sluggish governor relief valve.	Disassemble and clean.
	Sticky piston action in propeller dome assembly.	Remove dome shell. Clean and lubricate piston seals. If piston contact area of dome is scored, polish with "wet or dry" 000 sandpaper.
	Loose piston seal.	Tighten piston seal nut.
	Variation in engine transfer ring leakage.	Replace rings.
d. Leakage.	Loose distributor valve.	Tighten distributor valve or engine shaft extension.
	Leaky seals.	Replace seals.
	Dome breather hole seal damaged.	Replace seal.
	Loose spinner dome breather hole nut.	Tighten nut.
	Interference between breather pipe in propeller shaft and distributor valve oil transfer plate fillet.	Rework fillet.
	Dome retaining nut seal damaged; loose nut.	Replace seal. Tighten nut.
	Blade packing damaged.	Replace packing.
	Blade molded chafing ring damaged.	Replace molded chafing ring with split-type chafing ring.
	Barrel halves seals damaged; improper closure of barrel halves.	Replace seals. Remove burrs from parting surfaces of barrel halves.
	Barrel spider packing damaged.	Replace packing.
	Spider shaft oil seal damaged.	Replace seal.
	Spider shaft oil seal washer improperly installed.	Replace seal and install washer properly.

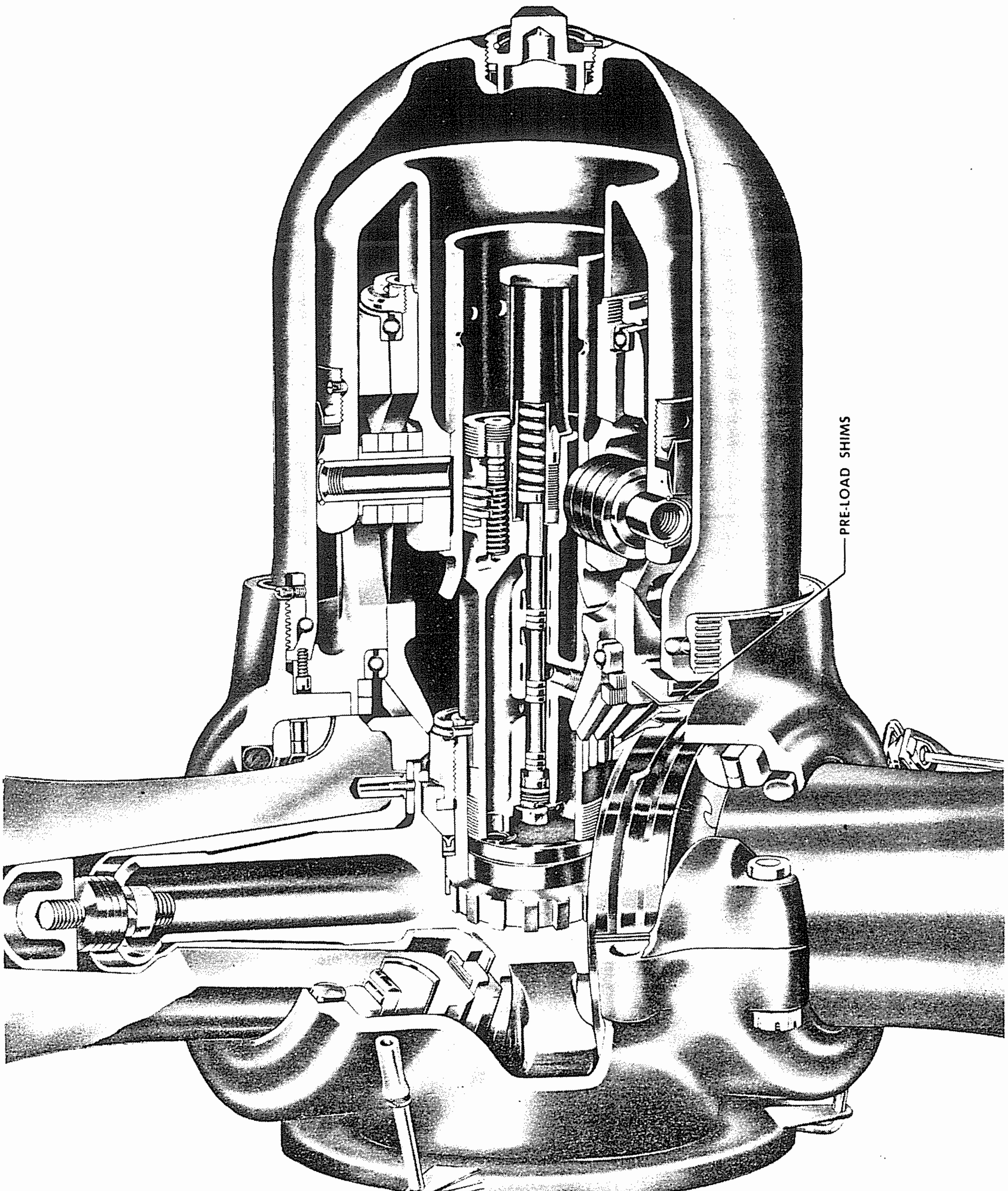


Figure 5-34. Propeller Hub and Dome Assembly

Trouble	*Probable Cause*	*Remedy*
d. *(Continued)*	Spider shaft oil seal washer missing.	Install washer.
	Engine thrust plate seal.	Remove and replace seal.
e. Roughness.	Faulty spark plugs or ignition.	Replace spark plugs; check wiring.
	Propeller unbalance (effects will be greater at high rpm).	Check propeller balance.
	Blade angles set incorrectly (effects will be greater at high rpm).	Correct blade angles.
	Ice on propeller.	Increase rpm for a short period and turn on alcohol pump and propeller alcohol needle valve.
	Engine internal trouble.	Stop engine and investigate.
f. Failure to feather.	Batteries low.	Recharge batteries.
	Faulty electrical system.	Check power and control circuits of feathering pump.
	Failure of push-button to remain engaged.	Check wiring or pressure setting of cut-out switch.
	Feathering pump failure.	Replace pump.
g. Failure to unfeather.	Batteries low.	Recharge batteries.
	Faulty electrical system.	Check control and power circuits of feathering system.
	Defective feathering pump.	Replace pump.
	Failure of distributor valve to shift.	Replace or repair distributor valve.

5-201. PROPELLER HUB AND BLADE ASSEMBLY. The propeller hub and blade assembly is installed on the splined engine propeller shaft.

5-202. REMOVAL OF PROPELLER HUB AND BLADE ASSEMBLY.

a. Install the propeller sling.

b. Remove dome assembly (see paragraph 5-205).

c. Unscrew the propeller retaining nut.

d. Cover the propeller shaft with a thread protector cap.

e. Remove the hub and blade assembly from the engine propeller shaft, and remove the rear cone.

f. Cover the engine propeller shaft until installation of the propeller.

5-203. INSTALLATION OF PROPELLER HUB AND BLADE ASSEMBLY. To install the propeller hub and blade assembly, proceed as follows:

Note

Before installation of the propeller hub and blade assembly, accomplish applicable depreservation, inspection, and cleaning of the propeller shaft as described in applicable technical orders.

a. Install the rear cone on the shaft, moving it aft until it contacts the engine propeller shaft thrust bearing nut.

b. Apply a thin film of engine oil to the propeller shaft threads.

c. Cover the propeller shaft threads with a protection cap.

d. Hoist the propeller in a position that will assure alignment with the blank spline and install the propeller on the shaft.

e. Remove the thread protection cap.

f. Apply a thin film of engine oil to the threads on the inner diameter of the propeller retaining nut.

g. Install the front split cone.

h. Turn the blades into reverse pitch so that the teeth position of the blade gear segments moves down into the hub.

i. Start and tighten the propeller retaining nut on the propeller shaft to a torque of from 1000 to 1600 foot-pounds.

j. Install the hub snap ring.

k. Install the distributor valve by checking to make certain that the copper seal and the distributor plate are in place against the adapter flange inside the propeller shaft. Install the oil transfer plate and shaft oil seal. Apply a thin film of engine oil to the threads on the base of the distributor valve. Screw the valve into the propeller shaft by hand. Tighten the distributor valve into the propeller shaft, applying a force of 100-200 foot-pounds. While this force is being maintained, strike the bar near the wrench with one light blow and repeat until one of the slots on the distributor valve housing is in alignment with the same hole

in the propeller shaft to which a slot in the propeller retaining nut was previously lined up.

CAUTION

Under no condition should the valve housing be backed off, even slightly, in order to obtain slot and hole alignment. If alignment cannot be obtained without exceeding the specified torque, remove the distributor valve and reinstall it, either by using a new oil transfer plate and shaft seal or reducing the thickness of the first seal by lapping it slightly.

l. Install the propeller retaining nut lockwire with its head placed through the retaining nut slot, propeller shaft hole, and into the distributor valve housing slot (spline groove). Turn each propeller blade by hand to the feathered position, aligning the 88-degree mark on the blade shank with the scribe mark on the inner edge of the barrel shelf. Install the propeller dome (see paragraph 5-205).

Note

After the initial flight of the aircraft has been accomplished, remove the propeller dome assembly (see paragraph 5-205). Loosen the propeller retaining nut slightly and retorque to 1000 to 1600 foot-pounds. Reinstall the dome assembly by reversing the removal procedure.

5-204. DOME ASSEMBLY. The dome assembly contains the pitch-changing mechanism. It consists of four major parts: two cylinder coaxial cams, a double-walled piston, and a dome cylinder that serves also as a housing for the entire unit. When the dome is installed in the hub assembly, the outer, or stationary cam becomes rigidly fixed in the barrel and provides support for the remaining parts in the dome unit. The inner or rotating cam, which is integral with the main gear, is supported within the stationary cam by ball bearings that take the gear reactions and piston oil forces. The piston motion is transmitted to the rotating cam by four sets of cam rollers, carried on shafts supported by the inner and outer walls of the piston.

5-205. REMOVAL OF DOME ASSEMBLY.

a. Remove the dome breather hole nut lockwire and the dome breather hole nut; drain the oil from the dome.

b. Install the dome lifting handle in the dome breather hole. Remove the dome retaining nut lockscrew cotter pin and remove the retaining nut lockscrew.

c. Back off the dome retaining nut. Lift off the dome assembly, using care not to damage the distributor valve assembly.

CAUTION

After removing the propeller dome, note if there are any preloaded shims in the propeller hub and blade assembly on the surface where the propeller dome seats *(see figure 5-34)*. These shims are for the purpose of maintaining the correct preload between the rotating cam gear in the propeller dome and the blade gear segments in the hub and blade assembly, and are prefitted at the factory. The correct thickness of preload may vary from 0.000 inch to 0.045 inch, and after it has been determined, is stenciled on the propeller dome barrel shelf in the hub and blade assembly. Shims of the correct thickness to correspond to the necessary preload are installed. If any preloaded shims are removed when the propeller dome is removed, they must be retained for reinstallation in the same hub and blade assembly from which they were removed.

d. Remove the propeller retaining nut lockwire.

e. Back off the propeller retaining nut two or three turns; then remove the distributor valve from the propeller shaft.

5-206. INSTALLATION OF DOME ASSEMBLY.

a. Check low-pitch and high-pitch stop rings in the base of the fixed cam to make certain that the dome operating range is correctly set for the installation.

Note

Be sure to reinstall the correct preloaded shims in the hub and blade assembly if they were removed during disassembly *(see figure 5-34)*.

b. Lift the dome assembly into position and install it over the fixed cam locating dowels in the hub assembly. Match the dome hole arrow to the hub dowel arrow.

c. Install the dome by sliding it straight on over the end of the distributor valve assembly. Make certain that the oil seal rings on the valve assembly properly enter into the sleeve inside the piston.

d. While installing, rotate the dome assembly in a counterclockwise direction until the fixed cam locating dowels in the barrel dome shelf enter the correct dowel holes in the fixed cam.

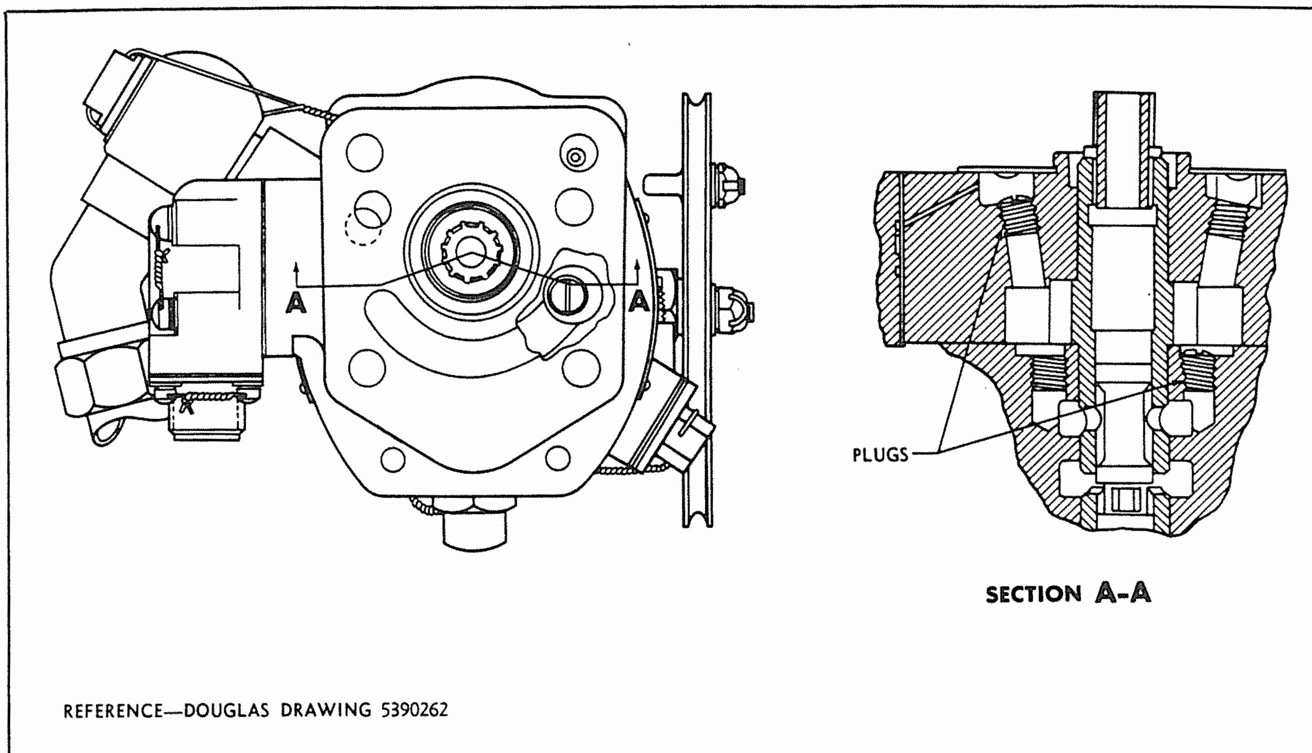


Figure 5-35. Plugs Installed in Propeller Governor

e. Start the dome retaining nut into the hub assembly by hand; then tighten the dome retaining nut, using a torque of 720 foot-pounds.

f. Tighten until the dome assembly seats on the dome-barrel shelf and one of the lockscrew lugs on the dome retaining nut is in line with the half-circle groove in the barrel retaining shoulder.

g. Remove the dome assembly. Install the dome and barrel seal on the outer perimeter of the fixed cam base, which is just aft of the dome retaining nut.

h. Reinstall the dome assembly as before and apply sufficient torque to the dome retaining nut to bring it to its previous position.

i. Install the dome retaining nut lockscrew in the dome retaining nut and safety the screw with a 1/16 x 1/2-inch steel cotter pin.

j. Install the dome breather hole seal in the dome breather hole at the outboard end of the dome assembly. Insert a dome breather hole washer over the seal.

k. Install the dome breather hole nut in the dome breather hole and tighten it with a torque of 30 to 50 foot-pounds. Tighten until one of the holes in the dome breather hole nut is aligned with a slot in the dome shell.

l. Snap the dome breather hole nut lockwire into the dome shell groove.

5-207. PROPELLER GOVERNOR. The propeller governor automatically maintains a constant engine speed by causing a change in the propeller blade angle throughout the constant speed range to meet changing conditions of altitude, attitude, and throttle setting. This change in blade angle is accomplished hydraulically by the propeller governor control through regulation of the flow of engine oil under pressure to and from the propeller.

5-208. REMOVAL OF PROPELLER GOVERNOR.

a. Disconnect the cables from the propeller governor pulley.

b. Disconnect the feathering pump high-pressure oil connection and the electrical control wire from the pressure cut-out switch.

c. Unscrew the mounting stud nuts that secure the governor unit to the engine pad; remove the governor.

5-209. INSTALLATION OF PROPELLER GOVERNOR.

Note

Before installing the propeller governor, make certain that the two holes are plugged, as shown in figure 5-35.

a. Check to make certain that the governor drive shaft turns freely by hand.

b. Place the governor on the mounting pad, making certain that the correct seal is in place.

c. Tighten the securing nuts evenly.

d. Make the electrical connection to the pressure cut-out switch, and attach the feathering oil pipes to the swivel fitting.

e. Install the pulley on the shaft, with the index mark on the end of the shaft opposite the index numeral 6 on the pulley.

5-210. PROPELLER FEATHERING PUMP. Each propeller is equipped with a feathering system powered by an individual electric motor-driven pump installed aft of the firewall. The feathering pump supplies oil, under high pressure, from the engine oil tank through the governor to the propeller for feathering or unfeathering the blades. A reserve supply of oil remains in the engine oil tanks for one feathering operation, in the event that engine oil is used down to the engine oil tank standpipe.

5-211. REMOVAL OF PROPELLER FEATHERING PUMP.

a. Drain the engine oil tank; then disconnect the oil pipes from the pump and cap the pipes.

b. Disconnect the electrical cable from the motor.

c. Remove the safety wire, nuts, and bolts and detach the feathering pump from the mounting bracket.

5-212. INSTALLATION OF PROPELLER FEATHERING PUMP. Reverse the removal procedure and in addition, fill the oil tank.

5-213. PROPELLER PITCH CONTROL CABLES. For a complete description of the system, including removal and installation procedures, see paragraph 5-257.

5-214. PROPELLER FEATHERING SYSTEM CIRCUIT BREAKERS. The propeller feathering system circuit breakers, one for each engine, are located on the circuit breaker panel in the flight compartment.

5-215. PROPELLER FEATHERING SWITCH BUTTONS. Two propeller feathering switch buttons are installed, one in each overhead electrical switch panel. Each button, in conjunction with a propeller feathering relay and a pressure cut-out switch, is used to control the motor of one of the propeller feathering pumps. When the button is depressed, the propeller feathering pump is energized. At the same time the feathering switch holding coil is energized and the button remains down until the pressure cut-out switch opens the holding coil circuit.

5-216. REMOVAL OF PROPELLER FEATHERING SWITCH BUTTON.

a. Remove the screws attaching the overhead electrical panel to the aircraft structure.

b. Pull the panel aft far enough to disconnect the wires.

c. Remove the jam-nut and remove the switch button.

5-217. INSTALLATION OF PROPELLER FEATHERING SWITCH BUTTON. Reverse the removal procedure.

5-218. PROPELLER BOLT TORQUE VALUES. For information on the propeller bolt torque values, see paragraph 5-263.

5-219. ENGINE COOLING SYSTEM.

5-220. DESCRIPTION. *(See figure 5-36.)* The engine cooling system (exclusive of the oil cooling system) includes those portions of the cowling diaphragm, etc., that direct the airflow over the engine, the cowl flaps and their operating mechanism, and the generator, magneto, and engine flexible mount blast tubes. The cowl flaps control the flow of air so as to provide engine cooling sufficient to maintain temperature limits as specified for best engine and aircraft performance. The cowl flaps are controlled by four-position selector switches located on the aft overhead electrical panel in the cockpit. The four selector switch positions are OFF, OPEN, CLOSE, and CLIMB. The flaps may be set at any desired position other than the automatic stop positions provided by the actuating unit limit switches by operating the switches to the selected position until the flaps reach that position and then switching to the OFF position. The flaps should stop in any position without coasting or creeping. The oil cooler, with its supporting structure, ducting, and fairing, forms the bottom section of each engine section cowling. A faired-in inlet in the top left side of the oil cooler entrance duct provides cooling air to the blast tubes that cool the generator, magnetos, and the engine flexible mounts. This small duct for the blast tube air slips inside a shutoff valve assembly mounted to the bottom aft side of the inner ring approximately at the engine center line. Eight aluminum-alloy cowl flaps, four on each side of each engine section, control the exit opening for the cooling air and thus control engine temperatures.

5-221. MINOR REPAIR AND REPLACEMENT OF ENGINE COOLING SYSTEM COMPONENTS. No minor repair of the engine cooling system components is recommended. If, upon careful inspection, a component of the system is malfunctioning, the unit should be removed from the aircraft and replaced with a new or overhauled unit.

5-222. COWL FLAPS AND ACTUATING UNIT. *(See figure 5-33.)* Eight aluminum-alloy cowl flaps, four on each side of each engine section, control the exit opening for the cooling air and thus control engine temperatures. The three upper flaps on each side are movable, while the lower flap on each side is fixed. The movable flaps are shock-mounted from the bow-ring support by rubber bushings in the flap hinge fittings. The cowl flaps are electrically extended and retracted by jackscrew assemblies, which are driven by an electric actuating unit mounted on a bracket attached to the aft side of the inner ring on the bottom center line of the engine section. The actuating unit consists of a reversible electric motor, a reduction gear box, and a limit switch box. Flexible shafting extends from both sides of the reduction gear box and connects to the jackscrew assemblies, which, in turn, are interconnected by flexible shafting to the other jackscrew assemblies to form a completely closed transmission circuit. The extension or retraction of the jackscrew assemblies operating the cowl flaps is accomplished by the reversible electric motor of the actuating unit. Full extended and retracted positions and an intermediate position are controlled by the actuating unit limit switches. The retract, or close, adjusting screw is located at the forward end of the switch box, and the extend, or open, and intermediate, or climb, adjusting screws are located at the aft end of the switch box, the extend adjusting screw being lower and on the right toward the reduction gear box. The actuating unit contains an electromagnetic brake for exact positioning without coasting of the jackscrew assembly. The jackscrew assemblies are shock-mounted to a box-and-bracket assembly at the aft side of the inner ring by rubber bushings in the jack assembly mounting holes. The jackshafts extend through the boxes and cut-outs in the inner ring and are attached to small brackets on the under side of the flaps. The jackshaft ends also contain rubber bushings for shock-mounting. A spring-loaded fire seal is incorporated on the housing of the jackscrew assembly and forms a tight seal on the box on the aft side of the inner ring.

5-223. REMOVAL OF COWL FLAPS.

a. Place the cowl flap selector switch for each engine, located on the left overhead electrical panel in the cockpit, in OPEN position to extend the flaps to the full open position.

b. Remove one bolt and bushing connecting each cowl flap jackscrew shaft end fitting to a bracket mounted on each flap.

Note

Unless the jackscrew assemblies are to be removed, after disconnecting all the jackshaft end fittings, be careful not to turn the shafts. Loop a wire through the end fitting mounting holes and secure to adjacent structure. This will insure the correct setting of shafts at installation.

c. Disconnect the bonding jumpers and remove two bolts attaching each movable cowl flap to the anti-drag ring and cowl flaps bow-ring support hinge assemblies.

d. Lift the cowl flaps free.

e. The lower fixed cowl flaps, one on each side of each engine section, are removable by loosening the fasteners.

Note

If desired, all the movable cowl flaps may be removed at one time by removing the anti-drag ring and cowl flaps bow-ring support.

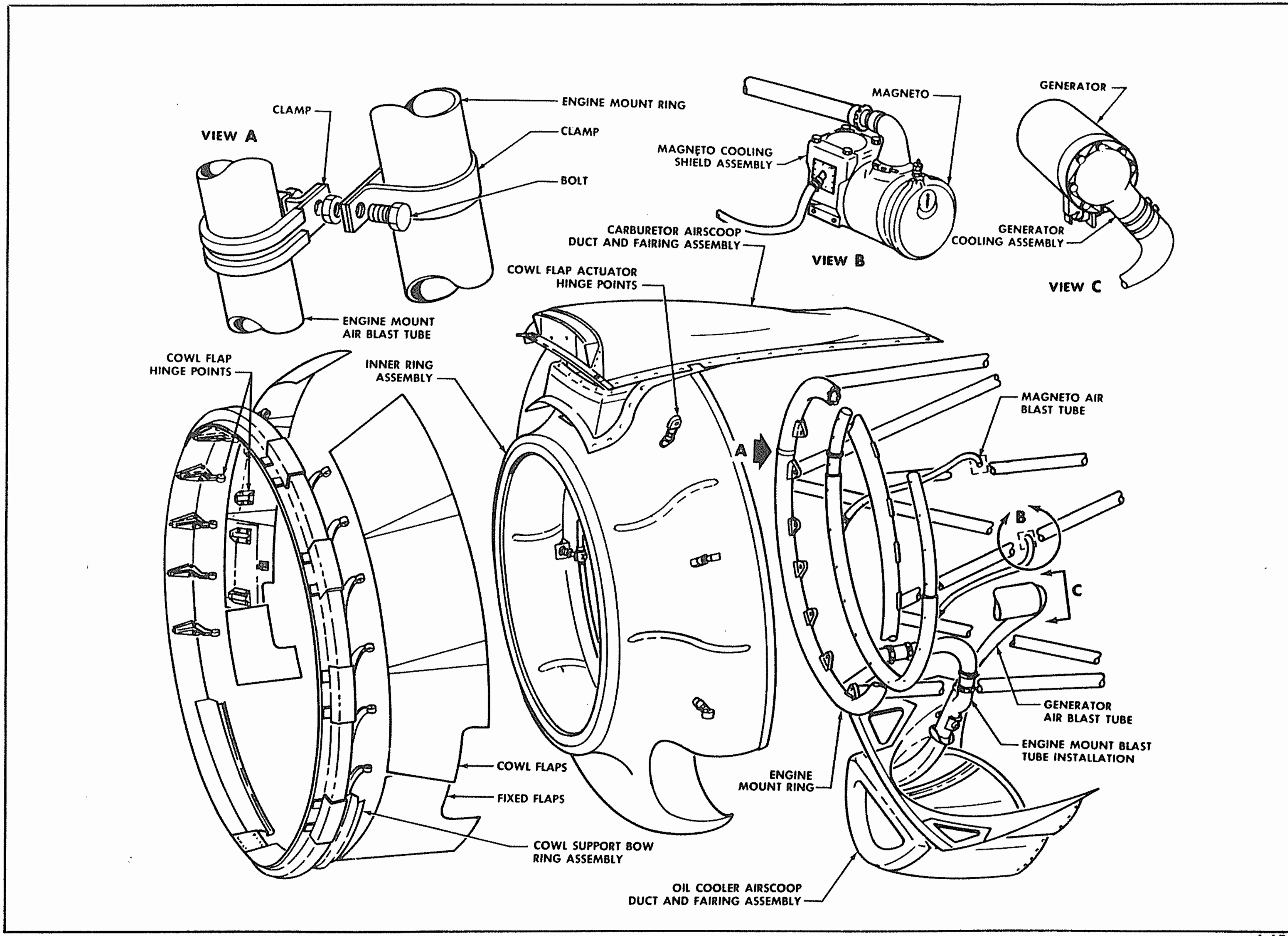


Figure 5-36. Engine Cooling System

5-224. REMOVAL OF COWL FLAPS ACTUATING UNIT.

a. Place the cowl flaps selector switch for each engine, located on the left overhead electrical panel in the flight compartment, in OPEN position to extend the flaps to the full open position.

b. Remove each engine section accessory cowling (see paragraph 5-5).

c. To remove each engine section cowl flap jack screw assembly, remove one bolt and bushing connecting each cowl flap actuator jackscrew end fitting to its flap. Then break the lockwire and disconnect the flexible drive shafting attach nuts from both sides of each jackscrew assembly. The splined ends of the flexible shafts should be covered immediately after being disconnected to protect them from nicks and burrs, which may prevent proper assembly. Remove one bolt and spacers attaching each jackscrew assembly to its mounting bracket at the aft side of the inner ring and pull the unit out through the engine accessory section.

d. To remove the cowl flaps actuating unit, first disconnect the electric lead at the actuating unit. Then break the lockwire and disconnect the flexible drive shafting attach nuts from both sides of the power unit. Break the lockwire and remove four bolts attaching each actuating unit to its mounting bracket at the aft bottom center line of the inner ring and lift the actuating unit free.

5-225. INSTALLATION OF COWL FLAPS ACTUATING UNIT. Reverse the removal procedure (see paragraph 5-224, step d).

5-226. INSTALLATION OF COWL FLAPS. Reverse the removal procedure (see paragraph 5-224, steps a through c).

5-227. ADJUSTMENT AND INSTALLATION TEST OF COWL FLAPS AND COWL FLAPS ACTUATING UNIT. To perform the adjustment and installation tests of the cowl flaps and the cowl flaps actuating unit, remove the accessory cowling from each engine, and disconnect each cowl flap jackscrew shaft end fiting from the flap. Then proceed as follows:

5-228. Adjust the cowl flaps as follows:

a. With the flexible drive shafting disconnected from each side of the cowl flap actuating units, place the cowl flaps selector switch for each engine, located on the left overhead electrical panel in the cockpit, in the OPEN position and allow the actuating unit limit switch to open the motor circuit.

b. Manually screw each jackscrew assembly jackshaft out until each shaft stops at its full extended position against the internal stop. Reverse the direction of rotation to back each jackshaft in a minimum of one-half turn and a maximum of one turn so that the hole in the mounting eye will line up with the hole in the cowl flap bracket.

c. Connect the flexible drive shafting to each side of the actuating unit. Connect each cowl flap to its jackscrew fitting. Check for interference between the forward edge of any flap and the anti-drag ring. Cowl flap hinge castings may be shimmed to obtain sufficient clearance.

d. Energize the actuating unit until the cowl flaps approach the full closed position. Stop the actuating unit if any flap appears to be in danger of striking the inner ring. Return the flaps to full open position and adjust the jackscrew assembly mounting eye fittings individually to give flap clearance at the inner ring when the flaps are closed.

CAUTION

The mounting eye fitting must never be extended to expose more than $\frac{5}{16}$ inch of thread.

e. Repeat step d, preceding, until the actuating unit can be safely energized to the flaps full closed position. Then measure the gap between each flap and the inner ring. This gap should be $\frac{3}{32}$ ($\pm\frac{1}{16}$) inch.

f. If the gap does not fall within these limits, again energize the actuating unit to the cowl flaps open position and manually screw the jackscrew assembly jackshafts in as necessary to obtain the proper gap between the flaps and the inner ring when the flaps are closed. One-half turn of the jackshaft changes the length of the unit by approximately $\frac{1}{16}$ inch.

g. Repeat the adjustment procedure until a uniform gap setting is obtained at all flaps.

CAUTION

Actuating unit limit switches are factory set for the correct number of revolutions of the actuating unit drive shaft that will produce the nominal stroke for the jackscrew assemblies. This setting should not be changed unless absolutely necessary, since the stroke setting is affected by limit switch adjustment. If it does become necessary to adjust the limit switches, this should be accomplished only by turning the limit switch screws in a clockwise direction. If counterclockwise adjustment is required, back the screw off past the point desired and turn in a clockwise direction to the desired set point. If any limit switch screw requires more than one complete turn for adjustment, a check should be made of all limit switches to prevent jamming of the jacks.

h. Energize the actuating unit to the cowl flaps full open position. If the distance between the trailing edge of each flap and the inner ring is $5\frac{3}{4}$ ($+0$, $-\frac{1}{4}$) inches, the flap adjustment is adequate. If the gap is not within the limits, a limit switch adjustment may be made, observing the precautions noted above. The full open position limit switch is located at the aft end of the actuating unit limit switch box at the lower right toward the gear reduction box.

i. The flap retract or close limit switch setting should not be changed unless the required flap opening cannot be obtained by the preceding methods. The retract limit switch adjusting screw is located on the forward end of the actuating unit limit switch box.

j. Energize the actuating unit to the cowl flaps climb position. If the distance between the trailing edge of each flap and the inner ring is $1\frac{5}{8}$ ($\pm\frac{1}{16}$) inches, the actuating unit intermediate limit switch is in proper adjustment. The intermediate limit switch adjustment screw is located on the aft end of the actuating unit limit switch box adjacent to the extend or OPEN adjustment screw. The intermediate adjustment screw is the upper left of the two adjustment screws.

5-229. The installation test of the cowl flaps actuating unit is as follows:

a. When the adjustment of the jackscrew assemblies, cowl flaps, and flexible drive shafting for each engine has been properly set and coupled to the actuating unit as in step b, preceding, place the cowl flaps selector switches in the OPEN position and allow the cowl flaps to stop. They should be fully extended, $5\frac{3}{4}$ ($+0$, $-\frac{1}{4}$) inches gap.

b. Place the cowl flaps selector switches in the CLOSE position and allow the cowl flaps to stop. They should come to the full closed position, $\frac{3}{32}$ ($\pm\frac{1}{16}$) inch gap.

c. Place the cowl flaps selector switches in CLIMB position and allow the cowl flaps to stop. They should open to the CLIMB position, $1\frac{5}{8}$ ($\pm\frac{1}{16}$) inches gap, and stop without oscillating.

d. With the cowl flaps in the fully extended (OPEN) position, place the cowl flaps selector switches in CLIMB position and allow the cowl flaps to stop. They should close to the CLIMB position, $1\frac{5}{8}$ ($\pm\frac{1}{16}$) inches gap, and stop without oscillating.

5-230. GENERATOR, MAGNETO, AND ENGINE FLEXIBLE MOUNT BLAST TUBES. *(See figure 5-36.)* Cold air for cooling the generator, the magnetos, and the engine flexible mounts is obtained through a small flush airscoop in the top of the oil cooler airscoop. Cold air is delivered through a branch of this blast tube to the generator and through another branch to the ring-type blast tube for the flexible mounting brackets. Orifices in the ring tube direct jets of air over the flexible mounting brackets. Two tees from the ring blast tube direct air through flexible tubes to shields that are attached to the magnetos. A blast-tube shutoff valve, installed in the large blast tube just aft of the inner ring, is operated in conjunction with the other fire emergency shutoff valves. When either the left or right emergency shutoff handle is pulled out to the extreme position, the respective blast-tube valve is closed in conjunction with the fluid shutoff valves in that particular area.

5-231. REMOVAL OF GENERATOR, MAGNETO, AND ENGINE FLEXIBLE MOUNT BLAST TUBES.

a. Remove one clamp attaching the connecting hose between the generator blast tube and the blast tube shutoff valve.

b Remove one clamp attaching the connecting hose between the generator blast tube and fitting on the rear case of the generator, and remove the blast tube.

c. Remove one clamp attaching the connecting hose between the engine mount and magneto blast tube to the other outlet from the blast tube shutoff valve.

d. Remove one clamp attaching the connecting hose to each magneto blast tube shield.

e. Remove one clamp attaching the connecting hose between the two sections of the blast tube ring.

f. Remove the six clamps that attach the ring-type blast tube to the front engine mount ring of the engine mount.

g. Remove the two sections of the ring-type blast tube.

5-232. INSTALLATION OF GENERATOR, MAGNETO, AND ENGINE FLEXIBLE MOUNT BLAST TUBE. Reverse the removal procedure.

5-233. ENGINE COOLING SYSTEM BOLT TORQUE VALUES. For information on the engine cooling system bolt torque values, see paragraph 5-263.

5-234. POWER PLANT CONTROLS.

5-235. DESCRIPTION. The mechanical power plant controls extend from the control pedestal aft through the fuselage to the center wing, and outboard to each of the two power plant installations. The power plant controls consist of the cable assemblies, sectors, bellcranks, shaft assemblies, pulleys, and pushrods necessary for regulating the throttle, mixture, carburetor air, and propeller pitch settings. For a description of cowl flap, propeller feathering, oil cooler air exit door, fire emergency shutoff valves, and fuel selector valve controls, see their respective sections.

5-236. TROUBLE SHOOTING OF POWER PLANT CONTROLS. The majority of the troubles encountered with power plant controls can be attributed to improper rigging. A check for correct rigging, adjustment, and cable tensions is recommended as the first step toward correcting faults in the control system *(see figures 5-38 through 5-41).*

5-237. CONTROL PEDESTAL. *(See figure 5-37.)* The control pedestal structure consists of a sheet aluminum frame incorporating a cast aluminum head. The assembly is bolted to the cockpit floor. The control pedestal assembly encloses and supports the control shaft assemblies, bearings, cable drums, bellcranks, and pushrod assemblies for the throttle, mixture, carburetor air, and propeller pitch controls. To facilitate maintenance within the assembly, a round inspection plate is provided on each side of the pedestal.

5-238. THROTTLE CONTROLS. *(See figure 5-38.)* The throttle controls consist of two-way cable systems which extend from the throttle control handles in the control pedestal, aft through the fuselage and outboard to the throttle control linkage at each power plant carburetor. The throttles are actuated by the movement of the throttle control handles on the control pedestal. Automatic pilot control cables are connected to the throttle cables between stations 156 and 177.

5-239. THROTTLE CONTROL CABLES. *(See figure 5-38.)* The throttle control cables extend from the bellcranks in the control pedestal, aft through the lower part of the fuselage, outboard through the right and left wings to the nacelles, and forward through the nacelles to the throttle control pulleys located on the engine accessory sections.

5-240. REMOVAL OF THROTTLE CONTROL CABLES.

a. Disconnect the throttle control cables at the turnbuckles aft of the firewalls in the main gear wheel wells. Thread the cable ends for removal.

b. Remove the access door on the bottom of the fuselage at station 10. Disconnect the four clevis and pin connections just aft of the pulley brackets under the control pedestal.

c. Remove the necessary access doors on the bottom of the fuselage and center wing sections. Remove the necessary floor panels in the aircraft between the control pedestal and the front wing spar to gain access to the control cables for removal.

d. Disconnect the throttle warning signal levers from the throttle control cables just aft of station 64.

e. Remove the necessary fairlead grommets, pulley guard pins, and firewall fire seals.

f. Remove the throttle servo connectors on the main throttle control cables between stations 156 and 177.

g. Disconnect the throttle servo cables from the throttle servo sectors and remove the cables.

h. Pull the main throttle control cables at the clevis connections through the access door at station 10. Pull cables slowly to prevent chipping of the micarta pulleys. Use two men to perform this operation, one to pull the cables and the other to guide the cable ends through the pulley brackets and fairleads.

i. Release the cables from the throttle control pulleys on the engine accessory sections, draw the cables forward and remove them from the aircraft.

j. Remove the access door on the left side of the control pedestal. Disconnect the cables from the throttle control bellcranks and thread the cable ends for removal.

k. Pull the cables through the pulley bracket under the control pedestal and remove them from the aircraft.

5-241. MINOR REPAIR AND REPLACEMENT OF THROTTLE CONTROL CABLES

a. Examine the push-pull rod bearing ends in the control pedestal and at the carburetors on the engine accessory sections for freedom of movement. Replace any rods or ends found to be defective.

b. Inspect all pulleys, fairleads, and fire seals for general condition. Replace any badly worn fairleads or fire seals and damaged pulleys.

c. Check all cables and replace any cable that shows more than six broken wires in any one-inch length (see paragraph 2-51).

5-242. INSTALLATION OF THROTTLE CONTROL CABLES.

a. Route the forward throttle control cables through the pulley bracket under the control pedestal. Pull the cable ends up into the pedestal and connect them to the throttle control bellcranks. Install the access door on the left side of the control pedestal.

b. Connect the nacelle throttle control cables to the throttle control pulleys on the engine accessory sections. Route them aft through the pulley brackets and the firewalls, installing guard pins and fire seals.

(Continued on Page 445)

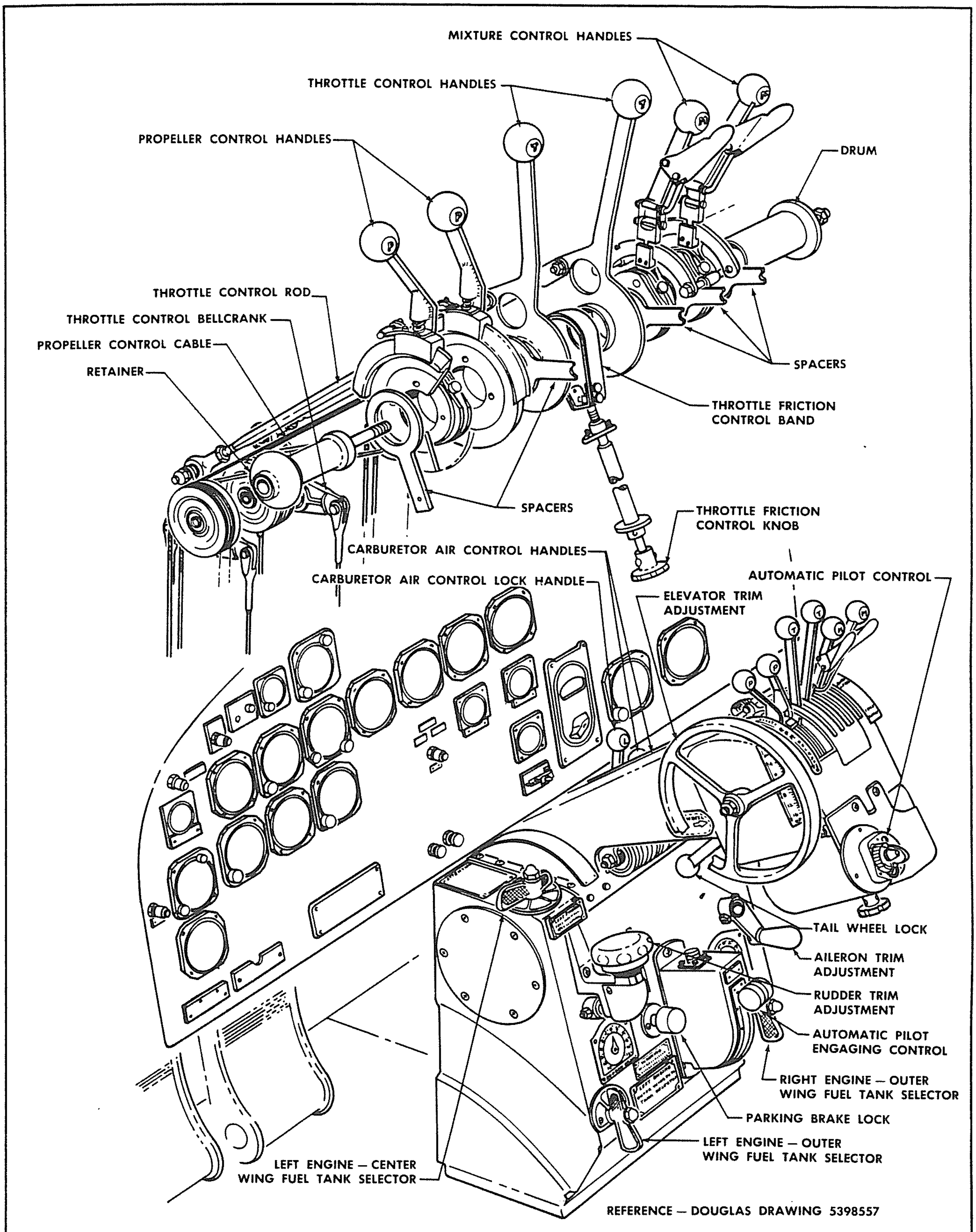
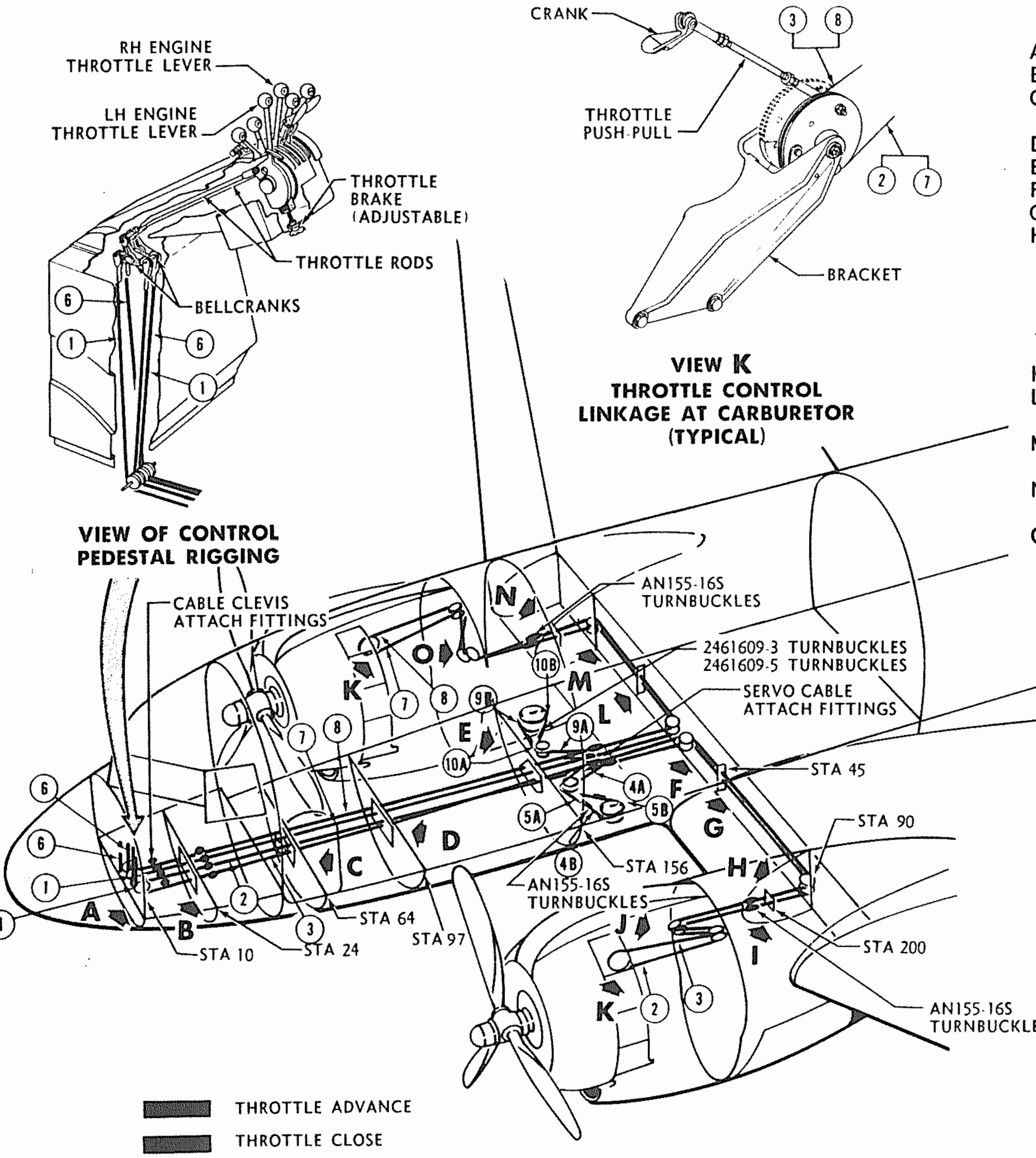


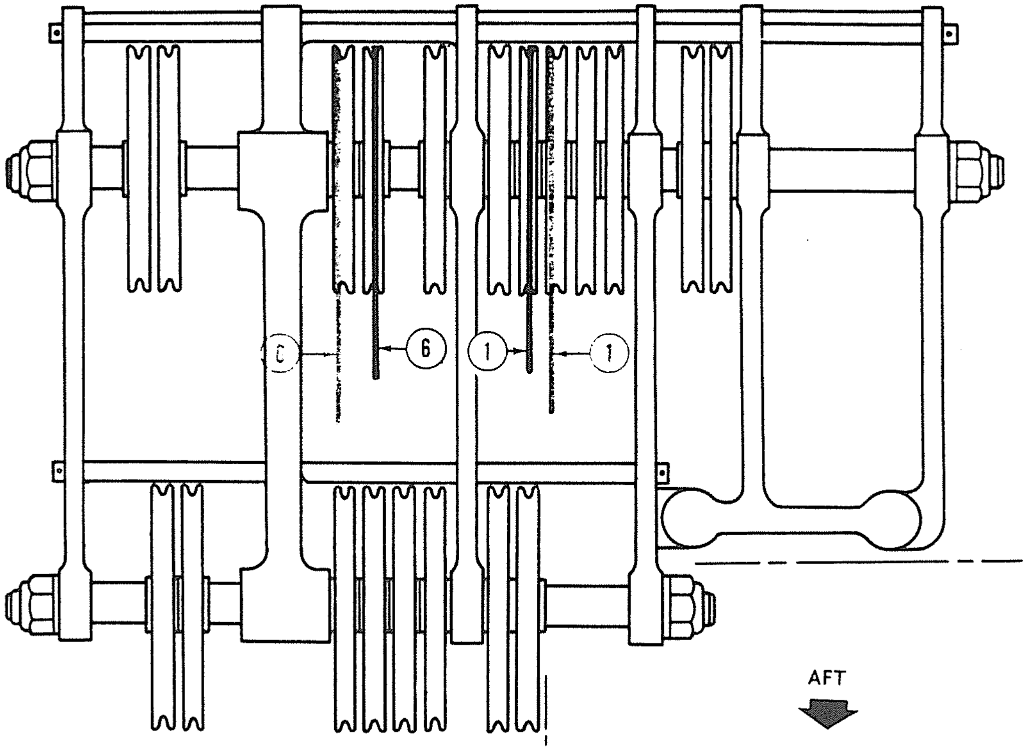
Figure 5-37. Control Pedestal



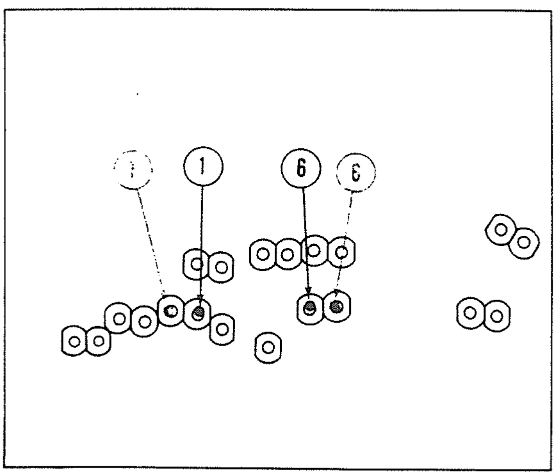
A —Pulley Bracket—Station 10, see Sheet 2
B —Fairlead—Station 24, see Sheet 2
C —Fairlead and Throttle Warning Signal Switch—Station 64 see Sheet 2
D —Fairleads—Station 97 see Sheet 3
E —Fairlead—Station 156, see Sheet 3
F —Pulley Brackets—Front Spar, see Sheet 4
G—Fairlead—Left Wing Station 45, see Sheet 4
H—Pulley Brackets—Left Nacelle Front Spar, see Sheet 4
I —Fairlead—Left Nacelle Station 200, see Sheet 4
J —Pulley Brackets—Left Nacelle Firewall, see Sheet 5
K —Throttle Control Linkage, see Sheet 1
L —Fairlead—Right Wing Station 45, see Sheet 5
M—Pulley Brackets—Right Nacelle Front Spar, see Sheet 5
N—Fairlead—Right Nacelle Station 200, see Sheet 5
O—Pulley Brackets—Right Nacelle Firewall, see Sheet 5

Notes:
1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 7.
2. For cable assembly drawings, see Sheet 8.
3. For Rigging and Adjustment Diagram, see Sheet 6.
4. Tension cables in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart, paragraph 5-264.
6. Reference—Douglas drawings 5372291, 5372513, 5372514, 5398252, 5398557, and 5399897.

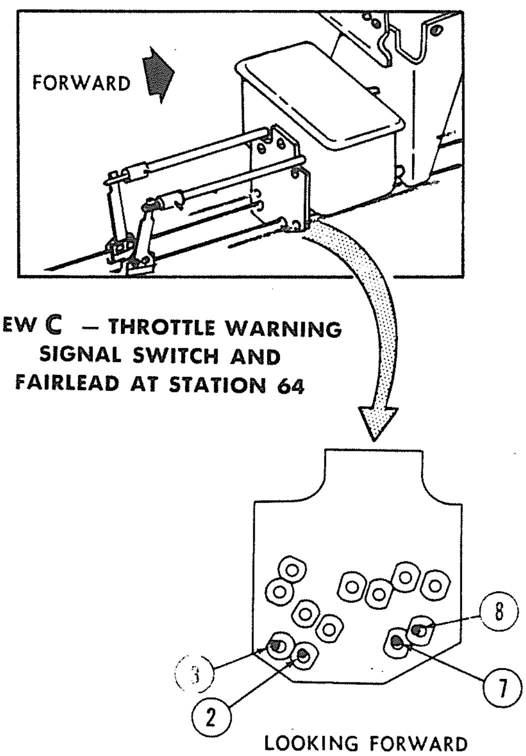
Figure 5-38 (Sheet 1 of 8 Sheets). Throttle Control System — Key Drawing



VIEW A — PULLEY BRACKET AT STATION 10 (BOTTOM VIEW)



VIEW B — FAIRLEAD AT STATION 24 (LOOKING FORWARD)



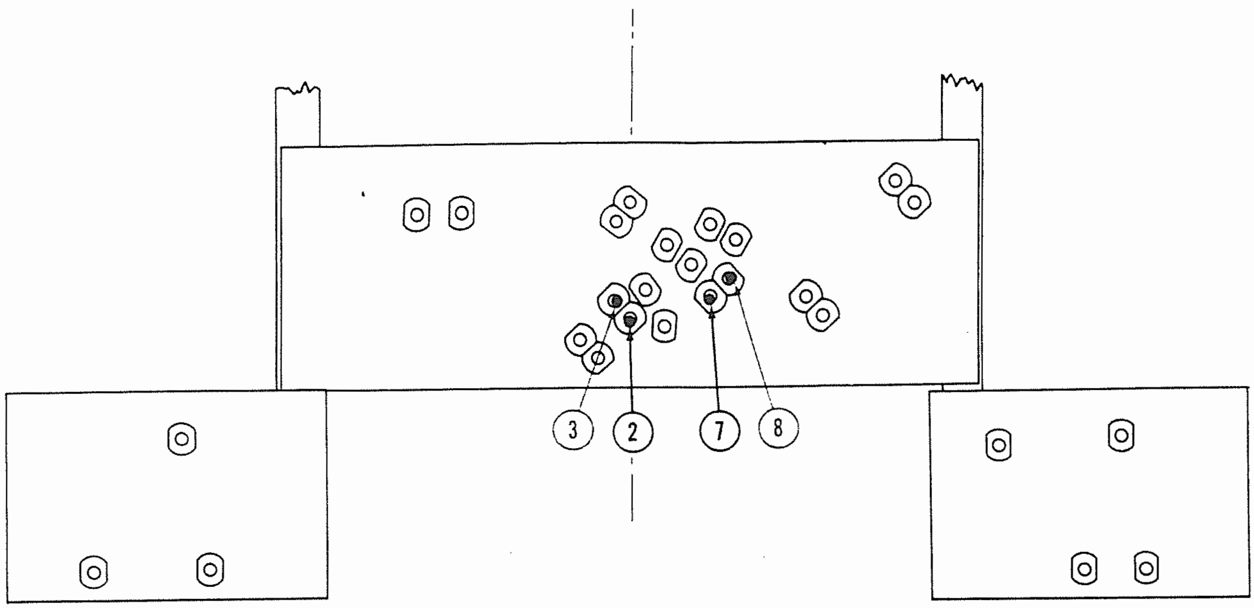
VIEW C — THROTTLE WARNING SIGNAL SWITCH AND FAIRLEAD AT STATION 64

THROTTLE ADVANCE

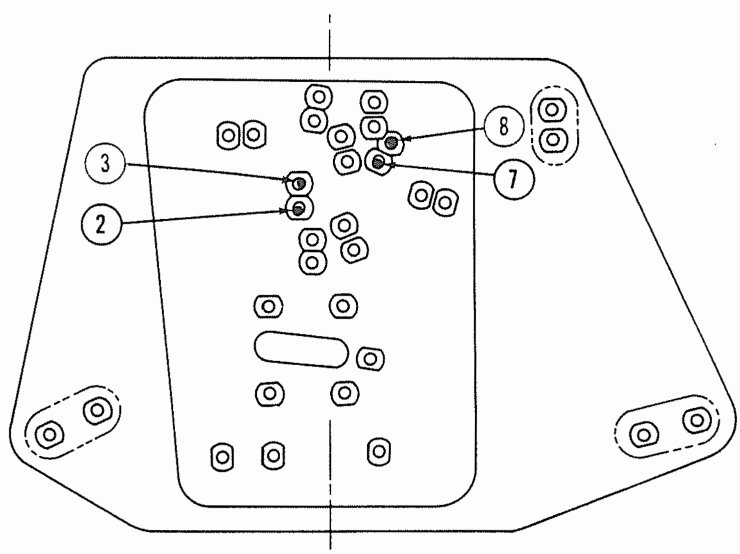
THROTTLE CLOSE

1,442

Figure 5-38 (Sheet 2 of 8 Sheets). Throttle Control System — Pulley Bracket, Station 10; Fairleads, Stations 24 and 64, and Throttle Warning Switch



VIEW D — FAIRLEAD AT STATION 97
(LOOKING FORWARD)



VIEW E — FAIRLEAD AT STATION 156
(LOOKING FORWARD)

THROTTLE ADVANCE

THROTTLE CLOSE

1,443

Figure 5-38 (Sheet 3 of 8 Sheets). Throttle Control System — Fairleads, Stations 97 and 156

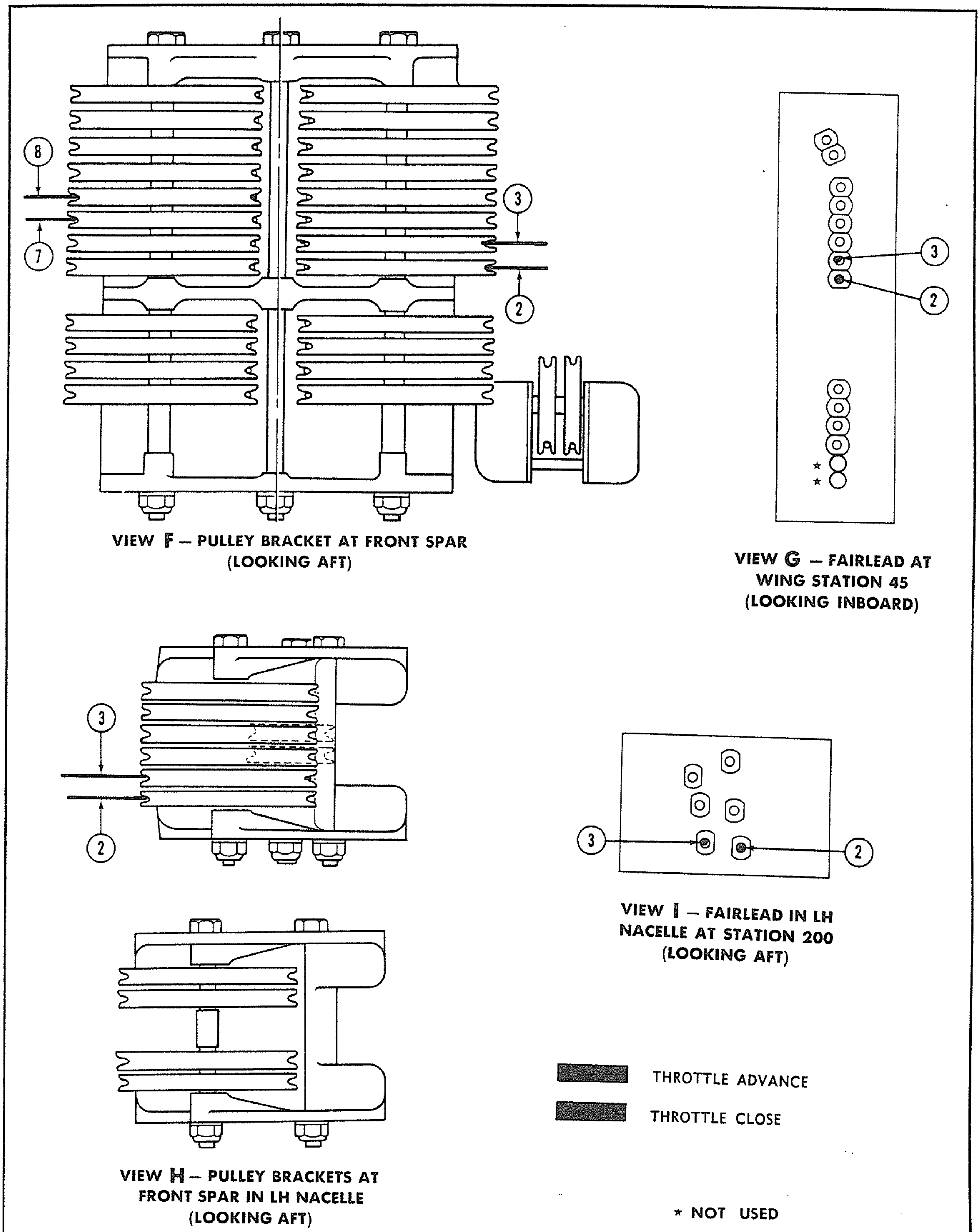


Figure 5-38 (Sheet 4 of 8 Sheets). Throttle Control System — Pulley Brackets, Front Spar and Left Nacelle Front Spar; Fairleads, Left Wing Station 45 and Left Nacelle Station 200

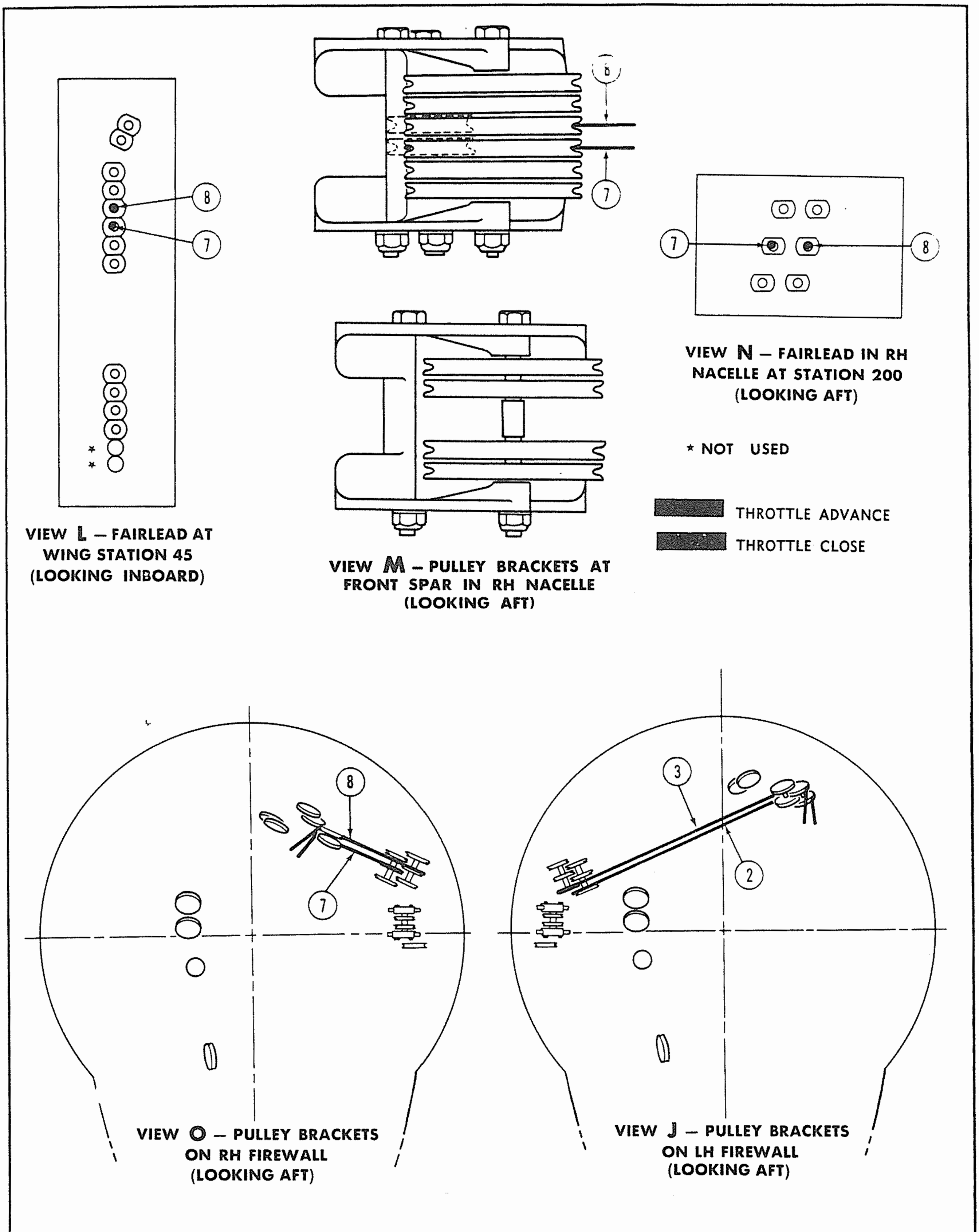
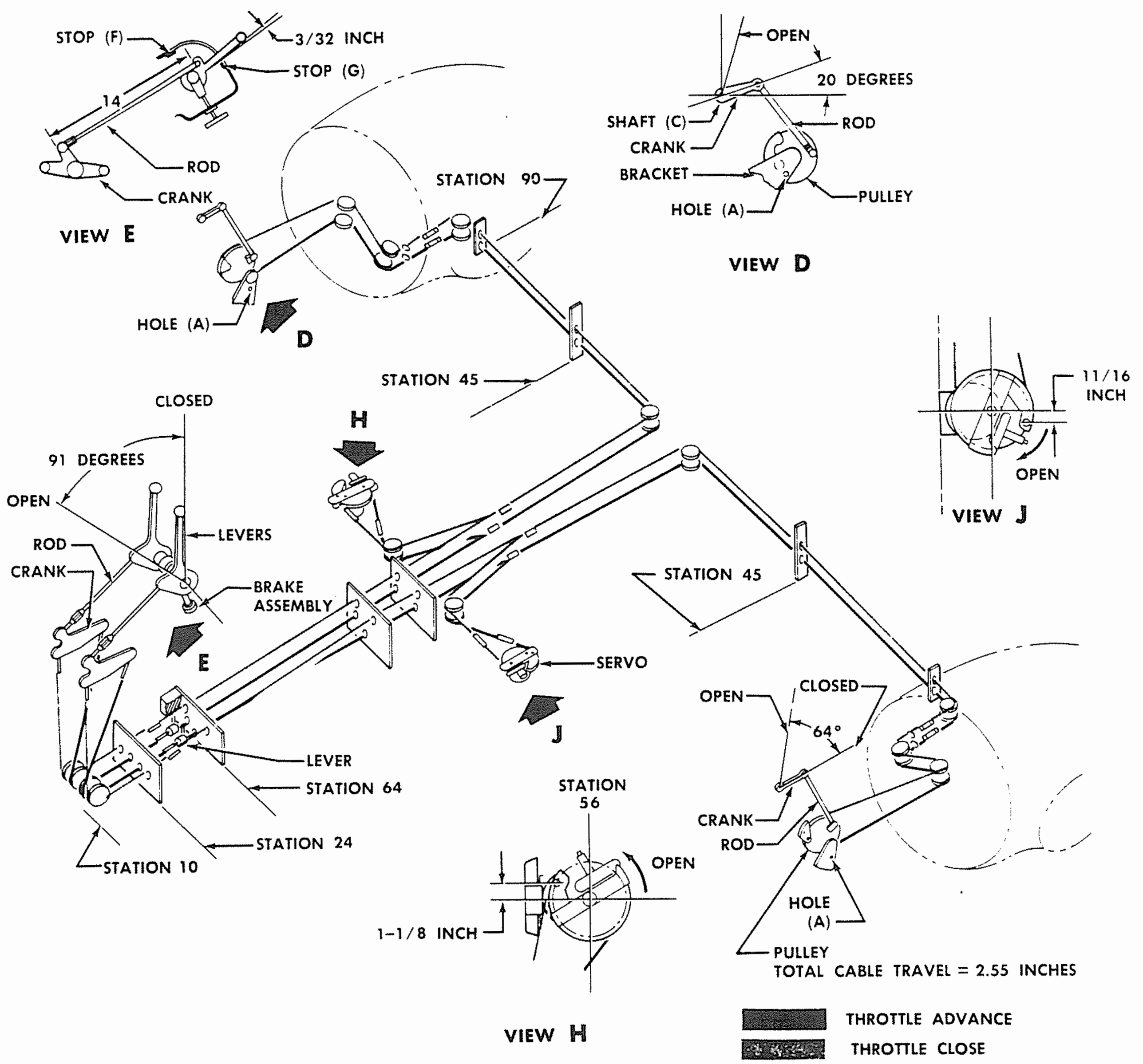


Figure 5-38 (Sheet 5 of 8 Sheets). Throttle Control System — Pulley Brackets, Right Nacelle Front Spar, Firewalls; Fairleads, Right Wing Station 45 and Right Nacelle Station 200



THROTTLE ADJUSTMENT PROCEDURE

1. Set the carburetor in throttle CLOSED position (shaft C, View D rotated full clockwise against the integral stops). Attach the throttle control crank in position 20 degrees above horizontal as shown. Note: Shaft C is serrated in increments of 15 degrees counterclockwise through an arc of 225 degrees and care should be taken to position crank 15 degrees from extreme right end of arc.
2. Align the bracket and pulley, using index hole (A) and adjust the length of the pushrod, if necessary, to install attach bolts.
3. Set the cockpit control levers in throttle CLOSED position (3/32 inch from pedestal stop G, View E). Tension the cables with the turnbuckles in the nacelles in accordance with the Cable Rigging Tension Chart, Figure 2-9.
4. Move the cockpit control lever to the OPEN position (3/32 inch from pedestal stop F, View E). Shaft C should be rotated full counterclockwise against the integral stops.
5. Position the throttle servo sectors as shown in Views H and J, and tension the turnbuckles in accordance with the Cable Rigging Tension Chart, Figure 2-9.
6. Safety the turnbuckles.

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Figure 5-38 (Sheet 6 of 8 Sheets). Throttle Control System — Adjustment Diagram

THROTTLE CONTROL CABLE CHART — LEFT ENGINE

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQD.	TYPE	CABLE LENGTH L_1 L_2	CABLE SIZE	FITTINGS 1	2	3	4
1	4408789-505 Open and Closed	2	A	39	3/32 dia 7x7 flex	1116245	AN667-3		
2	4400336-501 Open	1	B	(L_1) 337 (L_2) 74 1/2 (L_3) 152	3/32 dia 7x7 flex	AN668-3 (5) RA2487-3	AN663C3	AN669S3 RH	AN669S3 LH
3	4400336-1 Closed	1	B	(L_1) 331 3/4 (L_2) 74 3/4 (L_3)) 152	3/32 dia 7x7 flex	AN668-3 (5) RA2487-3	AN663C3	AN669S3 RH	AN669S3 LH
4A	4461608-505 Open	1	C	24	3/32 dia 7x7 flex	2400380	AN664C3	AN669S3 LH	
4B	4461608-503 Open	1	D	8 3/4	3/32 dia 7x7 flex	AN669S3 RH	RA2487-3		
5A	4461608-501 Closed	1	C	27 1/2	3/32 dia 7x7 flex	2400380	AN664C3	AN669S3 LH	
5B	4461608-1 Closed	1	D	10 3/4	3/32 dia 7x7 flex	AN669S3 RH	RA2487-3		

THROTTLE CONTROL CABLE CHART — RIGHT ENGINE

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQD.	TYPE	CABLE LENGTH L_1 L_2	CABLE SIZE	FITTINGS 1	2	3	4
6	4408789-507 Open and Closed	2	A	33 1/2	3/32 dia 7x7 flex	1116245	AN667-3		
7	4400336-505 Open	1	B	(L_1) 322 1/2 (L_2) 60 1/2 (L_3) 145 3/8	3/32 dia 7x7 flex	AN668-3 (5) RA2487-3	AN663C3	AN669S3 RH	AN669S3 LH
8	4400336-503 Closed	1	B	(L_1) 327 1/2 (L_2) 62 1/4 (L_3) 156 3/8	3/32 dia 7x7 flex	AN668-3 (5) RA2487-3	AN663C3	AN669S3 RH	AN669S3 LH
9A	4461608-513 Closed	1	E	13 1/2	3/32 dia 7x7 flex	2400380	AN664C3	RA2487-3	
9B	4461608-511 Closed	1	F	5 3/4	3/32 dia 7x7 flex	RA2487-3	RA2487-3		
10A	4461608-509 Open	1	E	19	3/32 dia 7x7 flex	2400380	AN664C3	RA2487-3	
10B	4461608-507 Open	1	F	6 1/2	3/32 dia 7x7 flex	RA2487-3	RA2487-3		

1,446

Figure 5-38 (Sheet 7 of 8 Sheets). Throttle Control System — Cable Charts

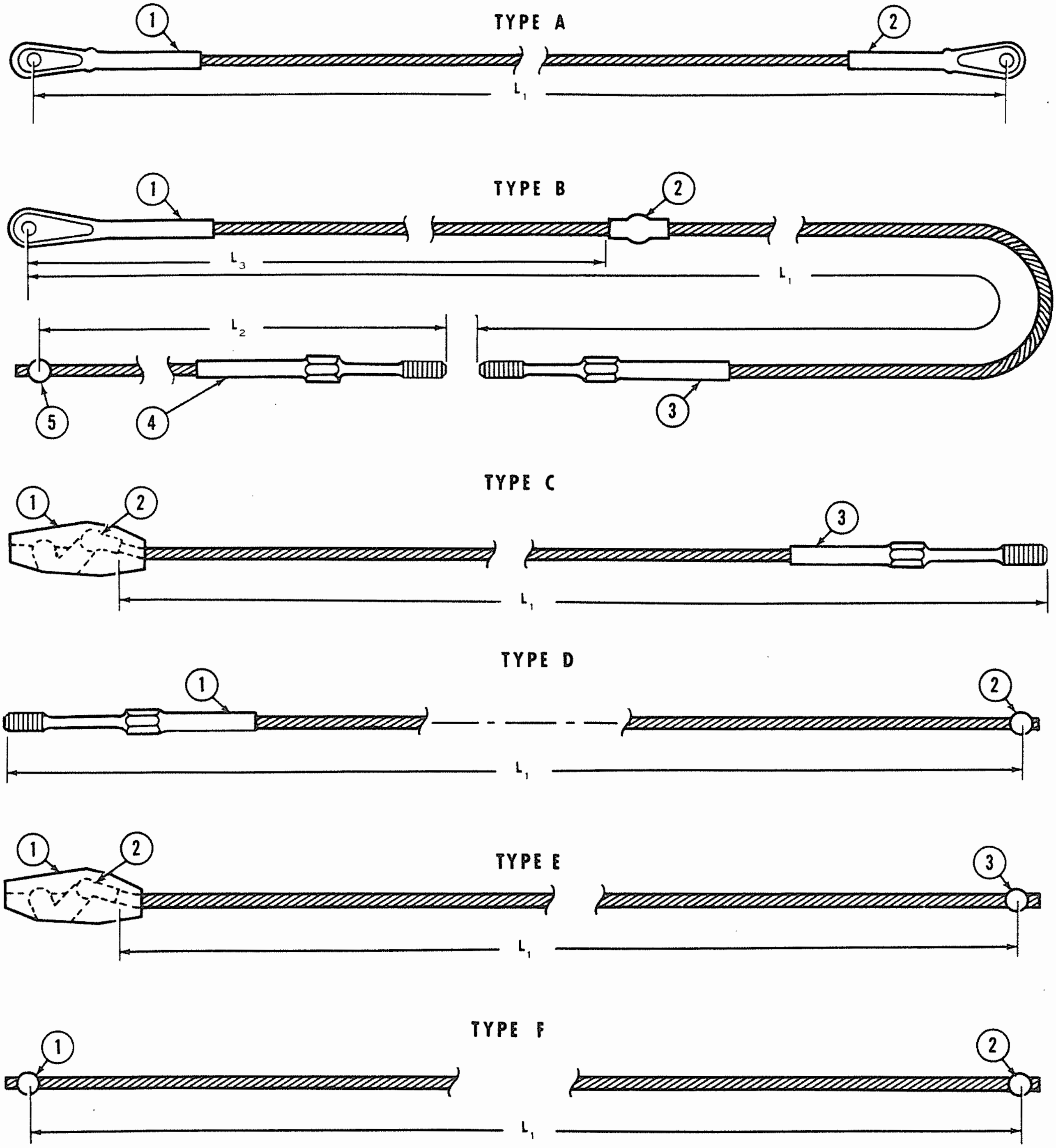
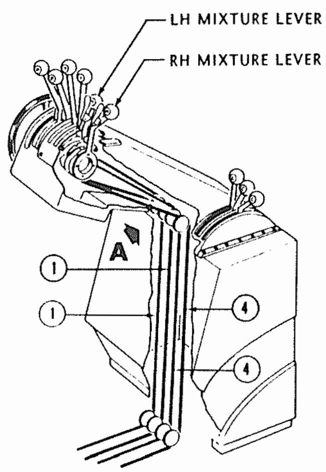


Figure 5-38 (Sheet 8 of 8 Sheets). Throttle Control System — Cable Assemblies

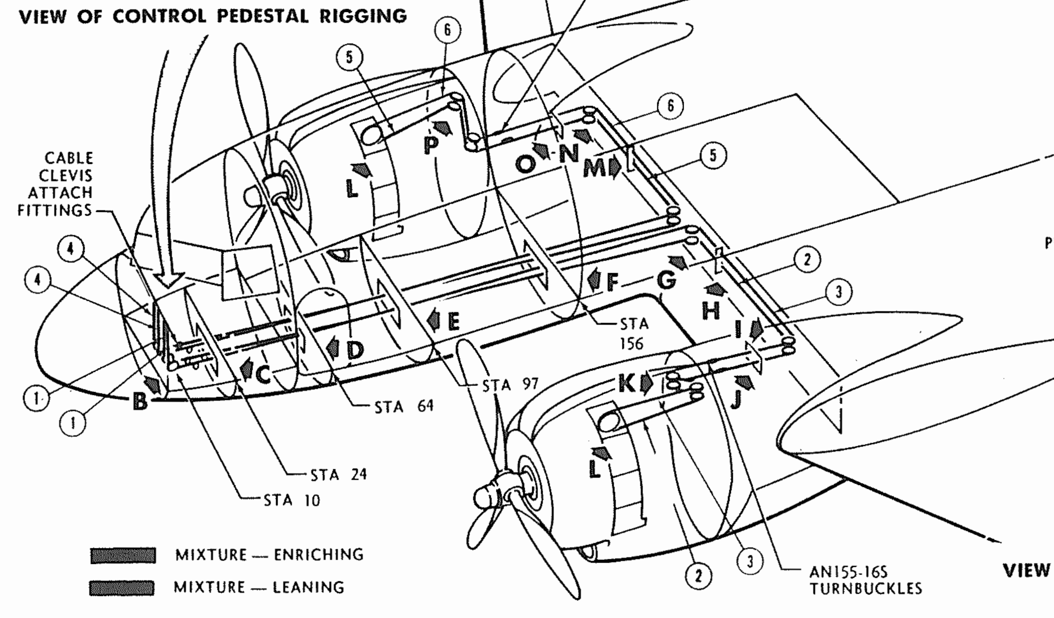


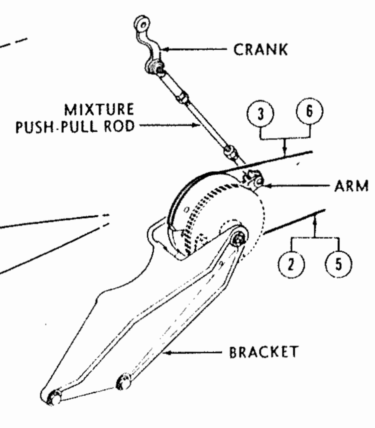
VIEW OF CONTROL PEDESTAL RIGGING

Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 7.
2. For cable assembly details, see Sheet 7.
3. For Rigging and Adjustment Diagram, see Sheet 6.
4. Tension cables in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart, paragraph 5-264.
6. Reference—Douglas drawings 5372291, 5372513, 5372514, and 5398252.

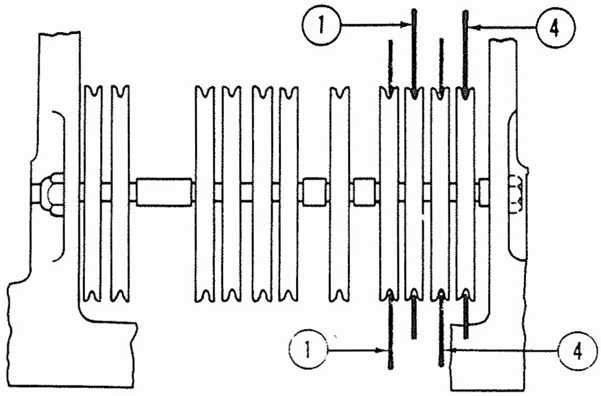
A—Pulley Bracket—Control Pedestal,	see Sheet 2
B—Pulley Bracket—Station 10,	see Sheet 2
C—Fairlead—Station 24,	see Sheet 3
D—Fairlead—Station 64,	see Sheet 3
E—Fairleads—Station 97,	see Sheet 3
F—Fairlead—Station 156,	see Sheet 4
G—Pulley Bracket—Front Spar,	see Sheet 4
H—Fairlead—Left Wing Station 45,	see Sheet 4
I—Pulley Brackets—Left Nacelle Front Spar,	see Sheet 4
J—Fairlead—Left Nacelle Station 200,	see Sheet 5
K—Pulley Brackets—Left Nacelle Firewall,	see Sheet 5
L—Mixture Control Linkage,	see Sheet 1
M—Fairlead—Right Wing Station 45,	see Sheet 5
N—Pulley Brackets—Right Nacelle Front Spar,	see Sheet 5
O—Fairlead—Right Nacelle Station 200,	see Sheet 5
P—Pulley Brackets—Right Nacelle Firewall,	see Sheet 5





VIEW L - MIXTURE CONTROL LINKAGE AT CARBURETOR (TYPICAL)

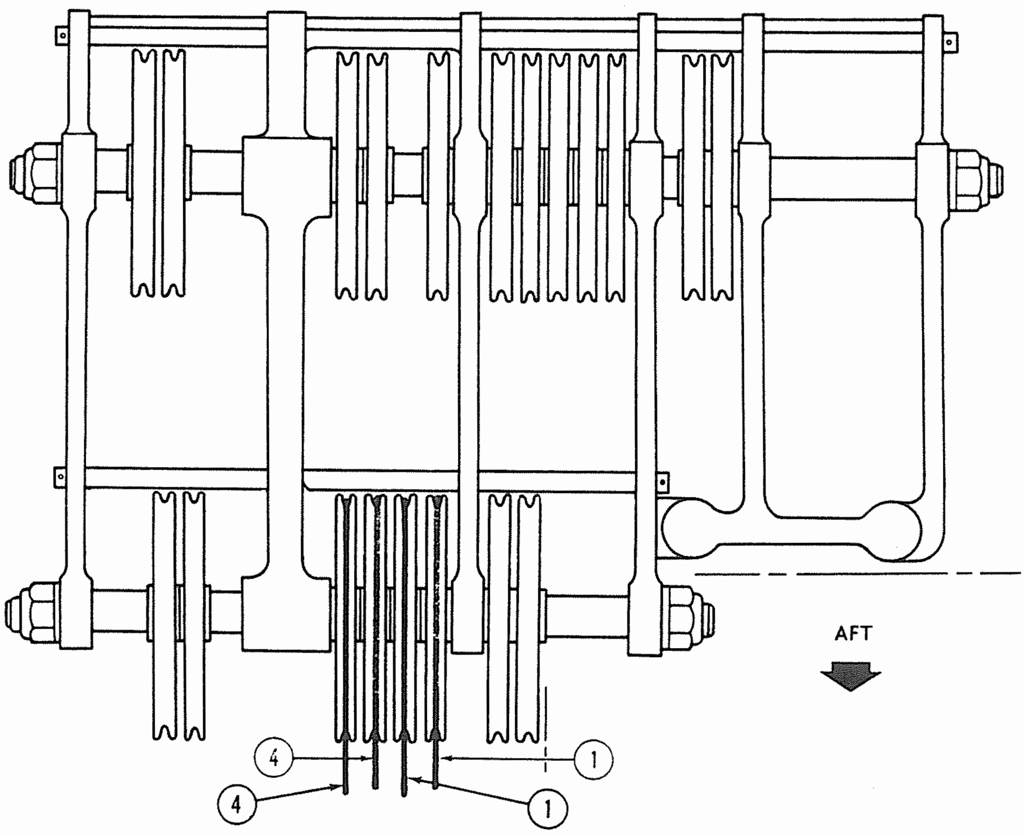
Figure 5-39 (Sheet 1 of 7 Sheets). Mixture Control System — Key Drawing



VIEW A — PULLEYS IN CONTROL PEDESTAL
(LOOKING FORWARD)

MIXTURE ENRICHING

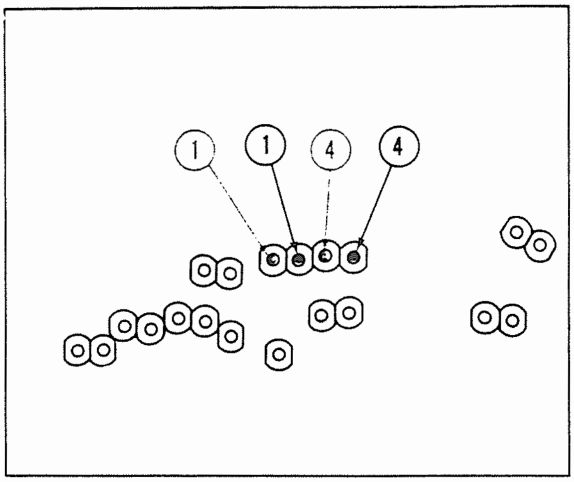
MIXTURE LEANING



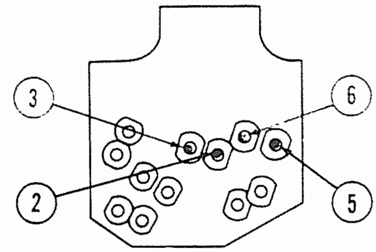
VIEW B — PULLEY BRACKET UNDER CONTROL PEDESTAL
AT STATION 10 (BOTTOM VIEW)

1,449

Figure 5-39 (Sheet 2 of 7 Sheets). Mixture Control System — Pulley Brackets, Control Pedestal and Station 10



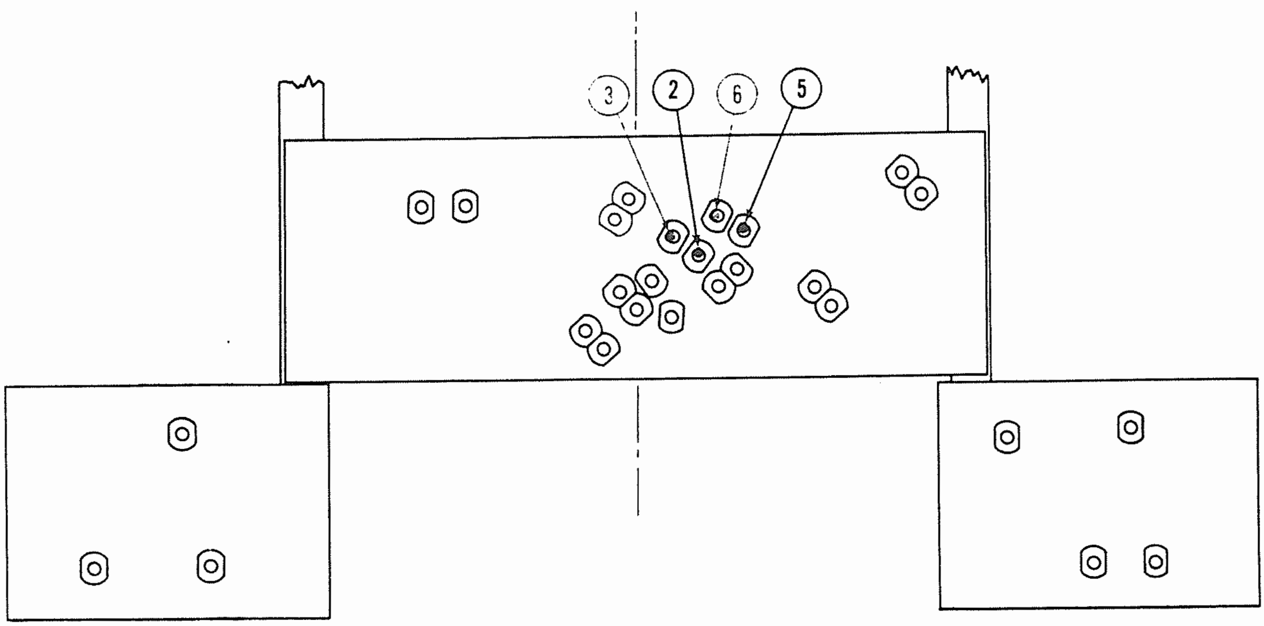
VIEW C — FAIRLEAD AT STATION 24
(LOOKING FORWARD)



VIEW D — FAIRLEAD
AT STATION 64
(LOOKING FORWARD)

MIXTURE ENRICHING

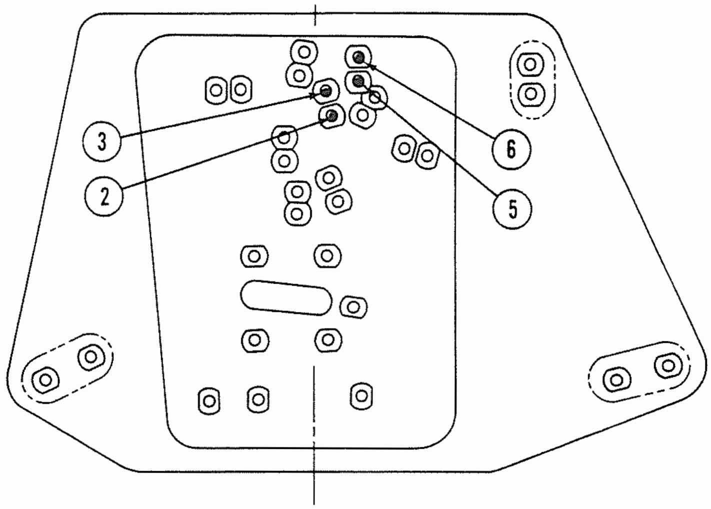
MIXTURE LEANING



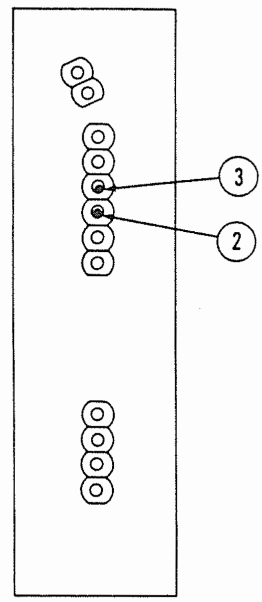
VIEW E — FAIRLEAD AT STATION 97
(LOOKING FORWARD)

1,450

Figure 5-39 (Sheet 3 of 7 Sheets). Mixture Control System — Fairleads, Stations 24, 64, and 97



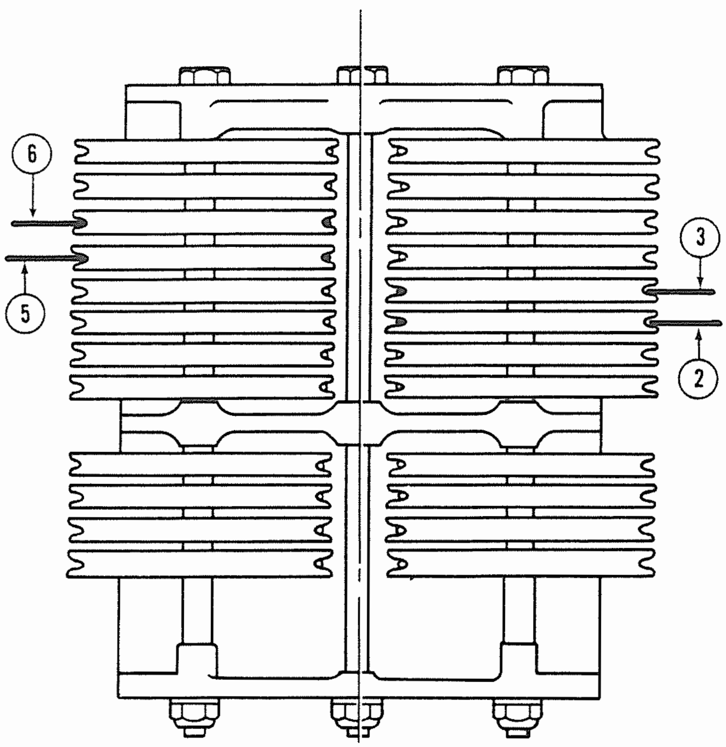
VIEW F — FAIRLEAD
AT STATION 156
(LOOKING FORWARD)



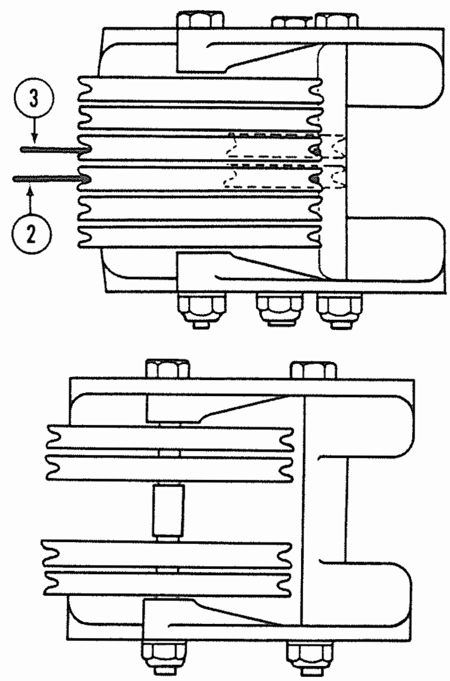
VIEW H — FAIRLEAD
AT LH WING STATION 45
(LOOKING INBOARD)

MIXTURE ENRICHING

MIXTURE LEANING



VIEW G — PULLEY BRACKET AT
FRONT SPAR STATION 222
(LOOKING AFT)



VIEW I — PULLEY BRACKETS AT
FRONT SPAR IN LH NACELLE
(LOOKING AFT)

1,451

Figure 5-39 (Sheet 4 of 7 Sheets). Mixture Control System — Pulley Brackets, Front Spar and Left Nacelle Front Spar; Fairleads, Fuselage Station 156 and Left Wing Station 45

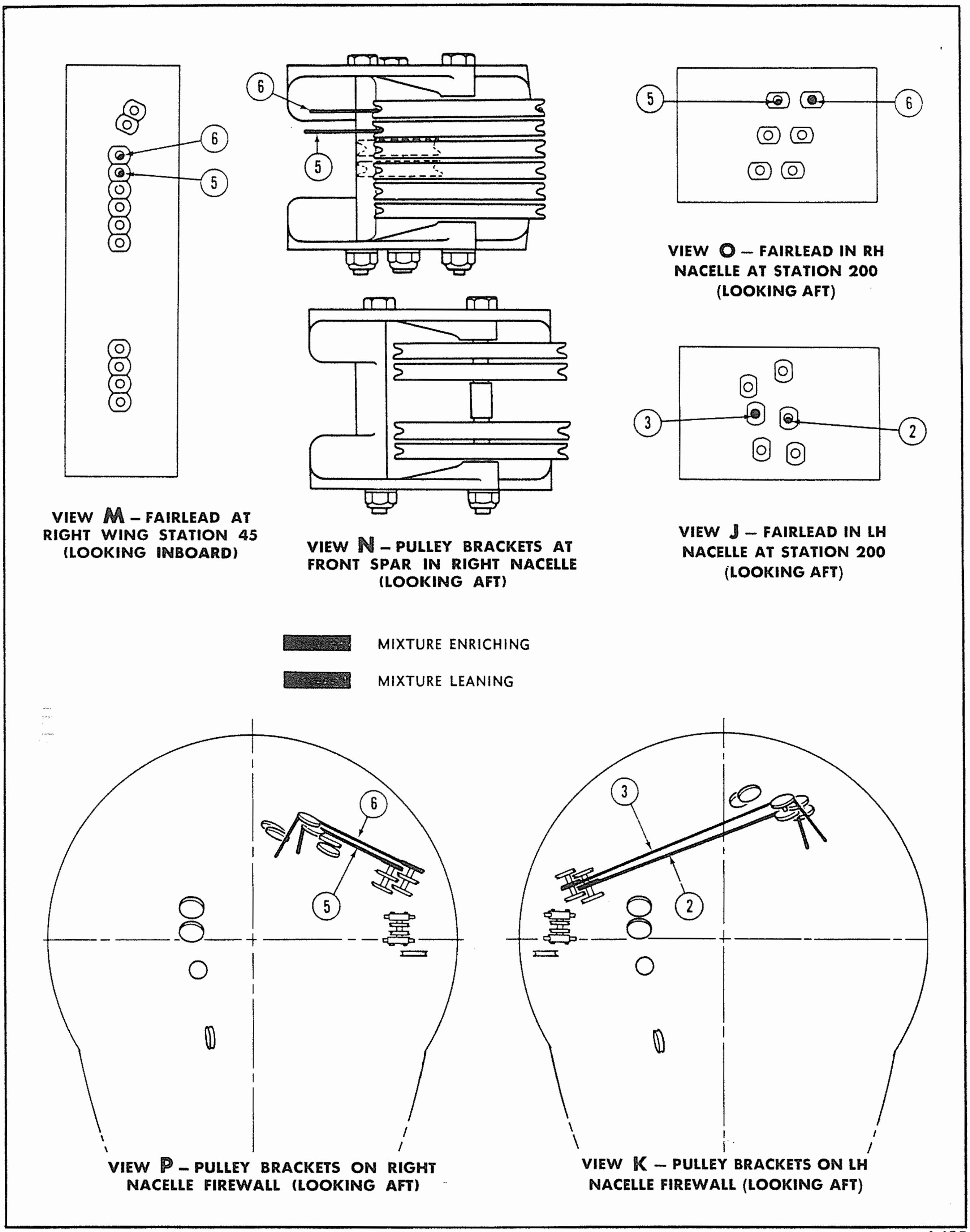
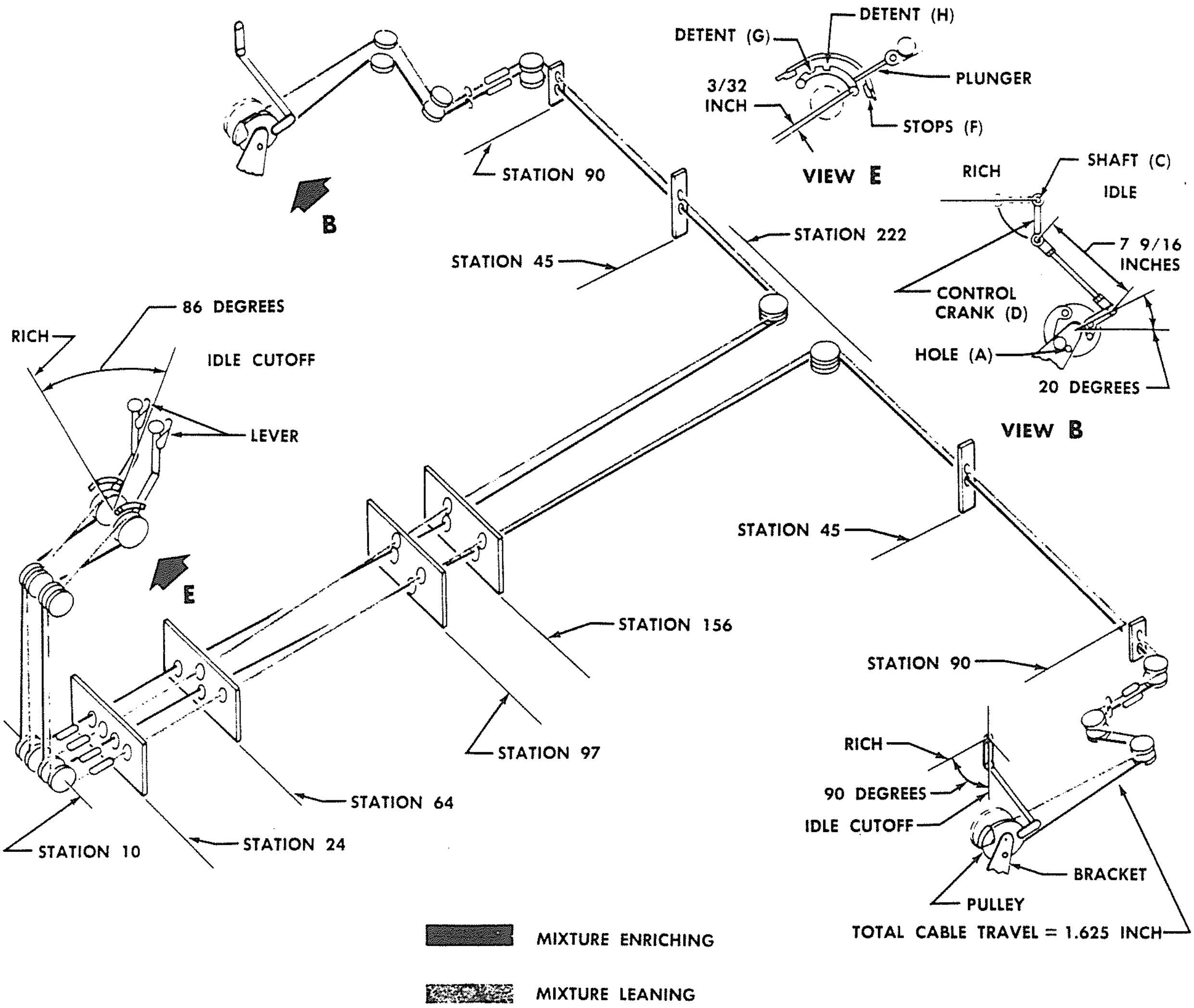


Figure 5-39 (Sheet 5 of 7 Sheets). Mixture Control System — Pulley Brackets, Right Nacelle Front Spar, Firewalls; Fairleads, Right Wing Station 45 and Right Nacelle Station 200



ADJUSTMENT PROCEDURE

1. Set the carburetor in the mixture IDLE CUTOFF position (Shaft C shown in View B rotated clockwise against the integral stop). Attach the mixture control crank D in vertical position as shown. Note: Shaft C is serrated in 15 degree incements and care should be taken to assure vertical positioning of crank.
2. Align the pulley and bracket using index hole at A.
3. Adjust the length of the pushrod, if necessary, to install the attaching bolts.
4. Remove the pin from index hole A.
5. Set the cockpit control levers at IDLE CUTOFF position (3/32 from the integral pedestal stops F shown in View E) and tension the cables with the turnbuckles in the nacelle in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
6. Move the cockpit levers to MIXTURE RICH position (plunger in detent G shown in View E). The carburetor should be in MIXTURE RICH position (Shaft C rotated full clockwise against the integral stops).
7. Move the cockpit levers to AUTO LEAN position (plunger in detent H shown in View E). The carburetor should be in AUTO LEAN position (Detent at mid position of Shaft C).
8. Safety all turnbuckles.

8232

Figure 5-39 (Sheet 6 of 7 Sheets). Mixture Control System — Adjustment Diagram

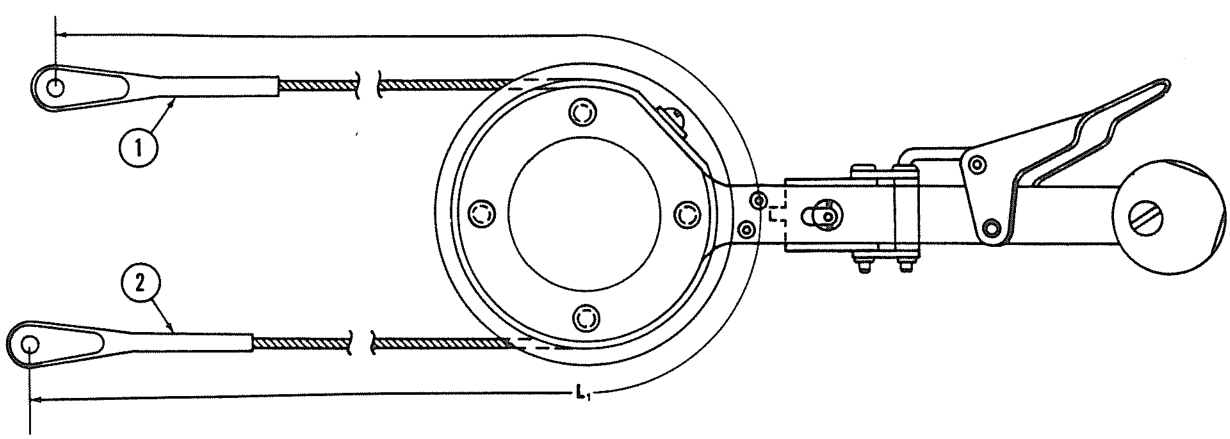
MIXTURE CONTROL CABLE CHART — LEFT ENGINE

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)
1	4115346-2 Rich and Idle Cut-Off	1	A	102 1/2	3/32 dia 7x7 flex	S-2049218 -C-3	S-2049218 -C-3		
2	3398717-505 Rich	1	B	(L_1) 326 1/2 (L_2) 82	3/32 dia 7x7 flex	AN668-3	AN669S3 RH	AN669S3 LH	RA2487-3
3	3398717-503 Idle Cut-Off	1	B	(L_1) 322 (L_2) 82 3/4	3/32 dia 7x7 flex	AN668-3	AN669S3 RH	AN669S3 LH	RA2487-3

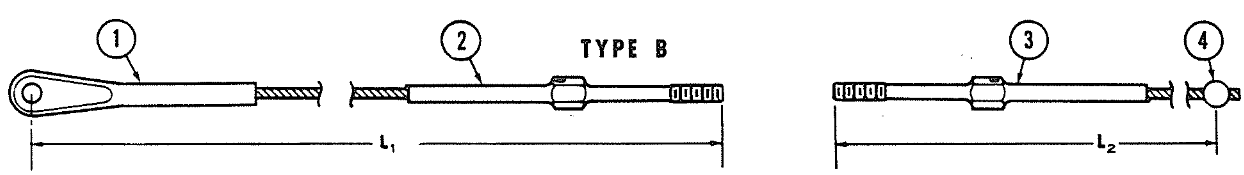
MIXTURE CONTROL CABLE CHART — RIGHT ENGINE

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)
4	4115347-2 Rich and Idle Cut-Off	1	A	102 5/8	3/32 dia 7x7 flex	S-2049218 -C-3	S-2049218 -C-3		
5	3398717-515 Rich	1	B	(L_1) 336 (L_2) 53 1/4	3/32 dia 7x7 flex	AN668-3	AN669S3 RH	AN669S3 LH	RA2487-3
6	3398717-517 Idle Cut-Off	1	B	(L_1) 331 3/4 (L_2) 54 1/2	3/32 dia 7x7 flex	AN668-3	AN669S3 RH	AN669S3 LH	RA2487-3

TYPE A



TYPE B

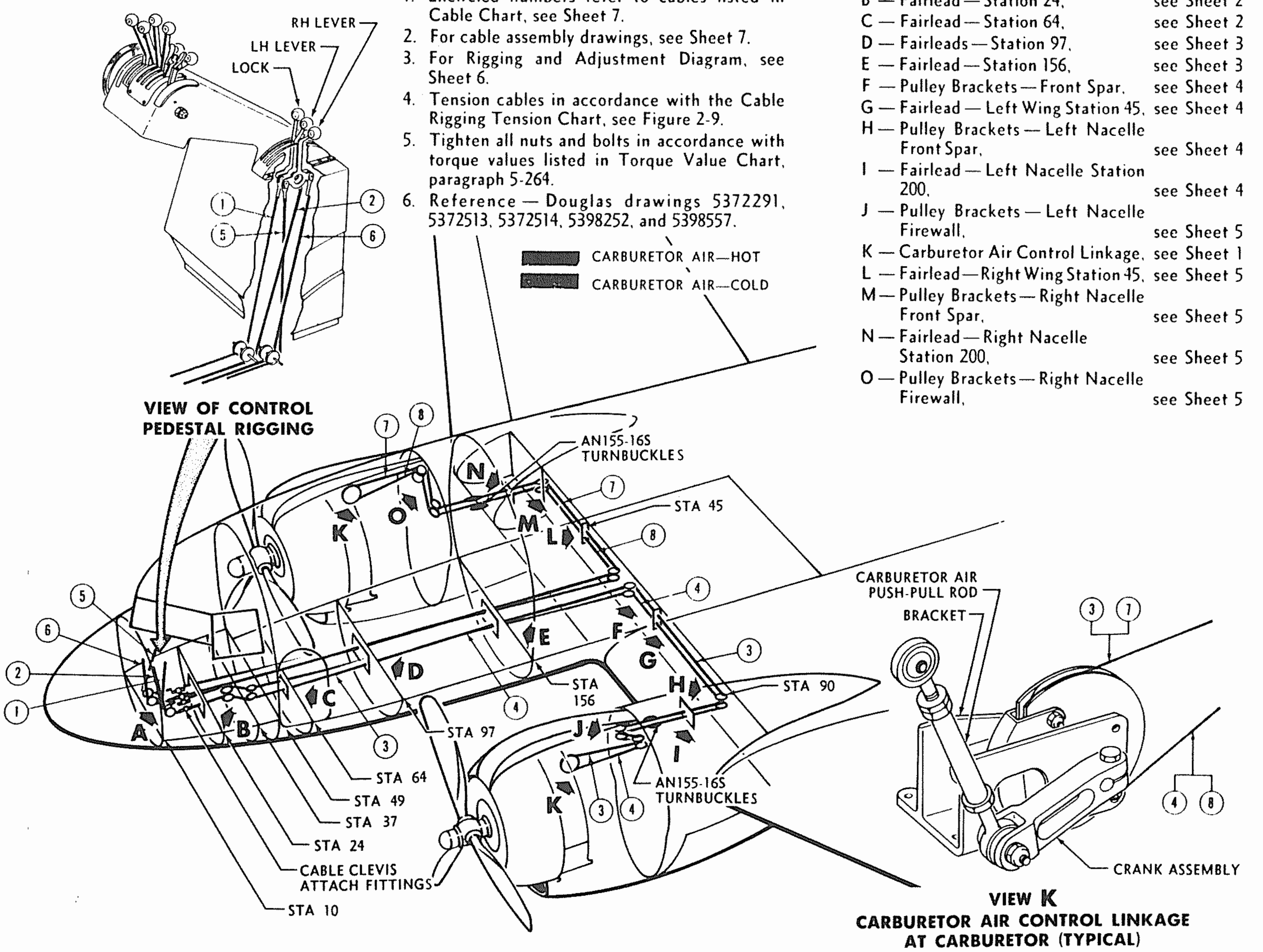


1,453

Figure 5-39 (Sheet 7 of 7 Sheets). Mixture Control System — Cable Charts and Cable Assemblies

Notes:
1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 7.
2. For cable assembly drawings, see Sheet 7.
3. For Rigging and Adjustment Diagram, see Sheet 6.
4. Tension cables in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
5. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart, paragraph 5-264.
6. Reference — Douglas drawings 5372291, 5372513, 5372514, 5398252, and 5398557.

A	Pulley Bracket—Station 10,	see Sheet 2
B	Fairlead—Station 24,	see Sheet 2
C	Fairlead—Station 64,	see Sheet 2
D	Fairleads—Station 97,	see Sheet 3
E	Fairlead—Station 156,	see Sheet 3
F	Pulley Brackets—Front Spar,	see Sheet 4
G	Fairlead—Left Wing Station 45,	see Sheet 4
H	Pulley Brackets—Left Nacelle Front Spar,	see Sheet 4
I	Fairlead—Left Nacelle Station 200,	see Sheet 4
J	Pulley Brackets—Left Nacelle Firewall,	see Sheet 5
K	Carburetor Air Control Linkage,	see Sheet 1
L	Fairlead—Right Wing Station 45,	see Sheet 5
M	Pulley Brackets—Right Nacelle Front Spar,	see Sheet 5
N	Fairlead—Right Nacelle Station 200,	see Sheet 5
O	Pulley Brackets—Right Nacelle Firewall,	see Sheet 5



1,454

Figure 5-40 (Sheet 1 of 7 Sheets). Carburetor Air Control System — Key Drawing

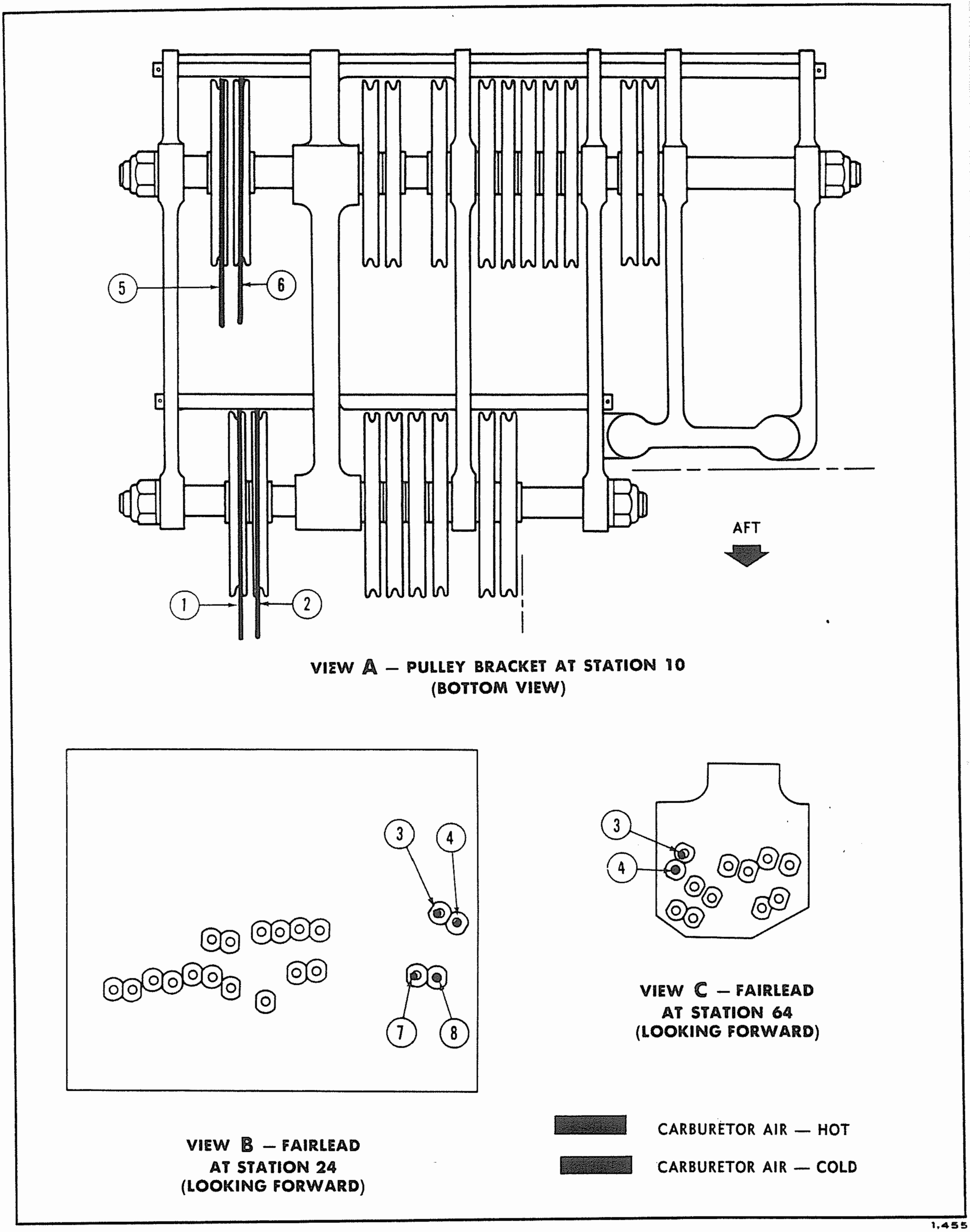
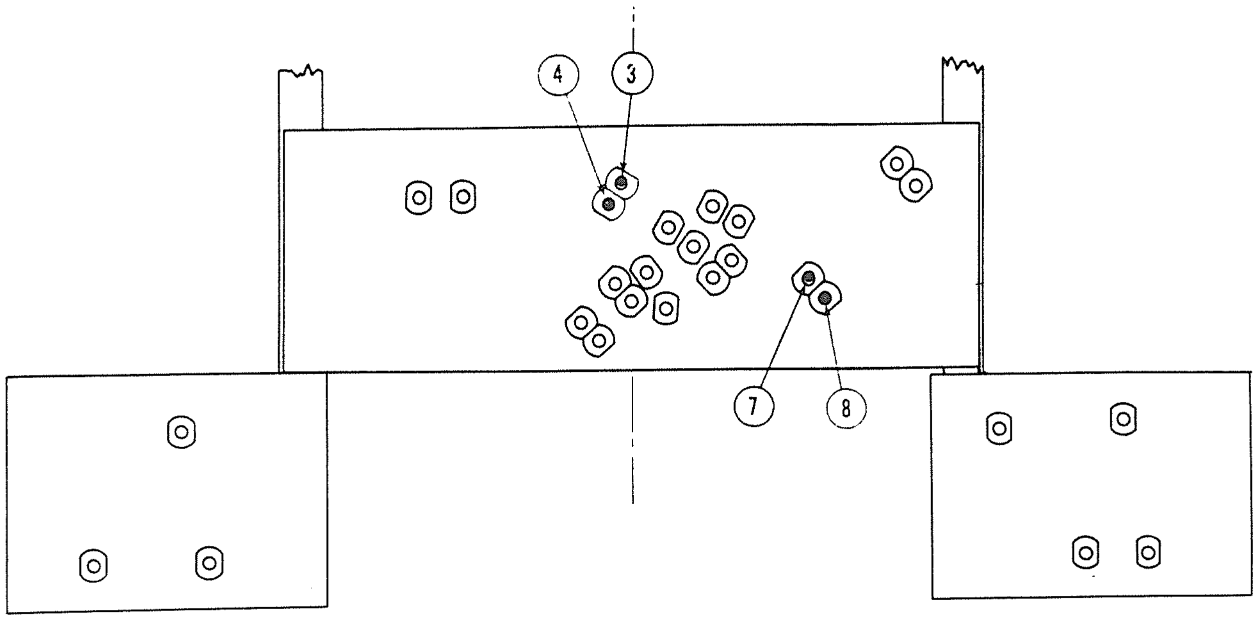
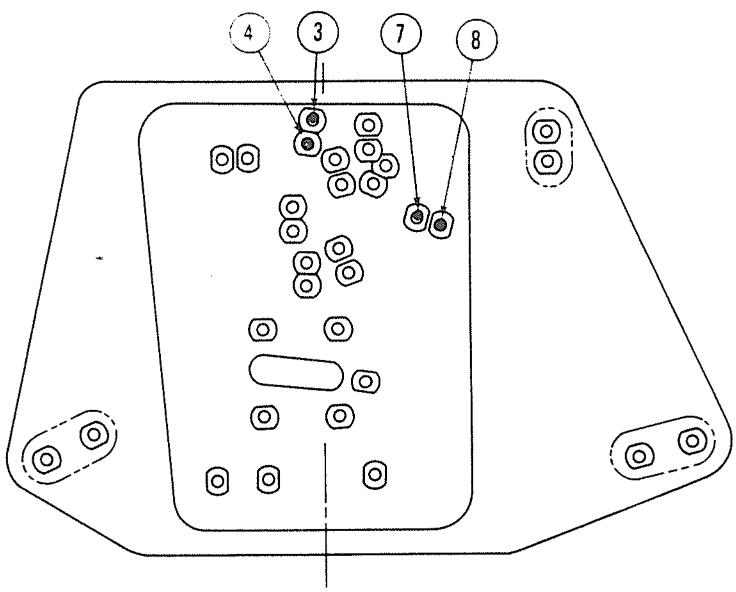


Figure 5-40 (Sheet 2 of 7 Sheets). Carburetor Air Control System — Pulley Bracket, Station 10; Fairleads, Stations 24 and 64



VIEW D—FAIRLEADS AT STATION 97
(LOOKING FORWARD)



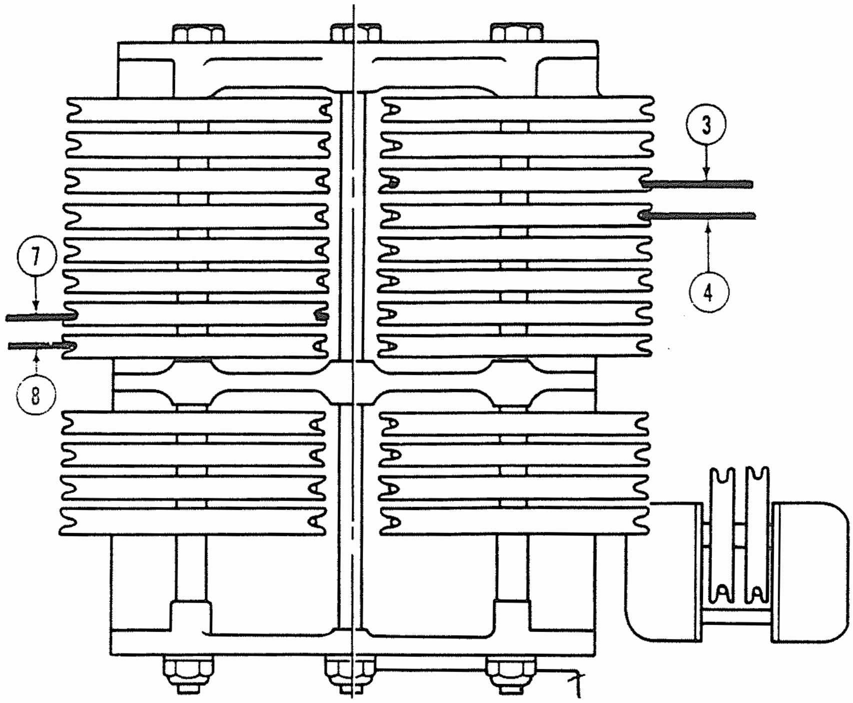
VIEW E — FAIRLEAD AT STATION 156
(LOOKING FORWARD)

CARBURETOR AIR — HOT

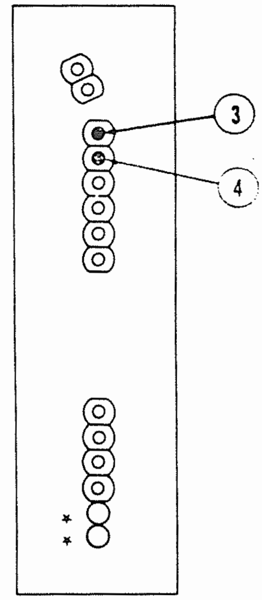
CARBURETOR AIR — COLD

1.456

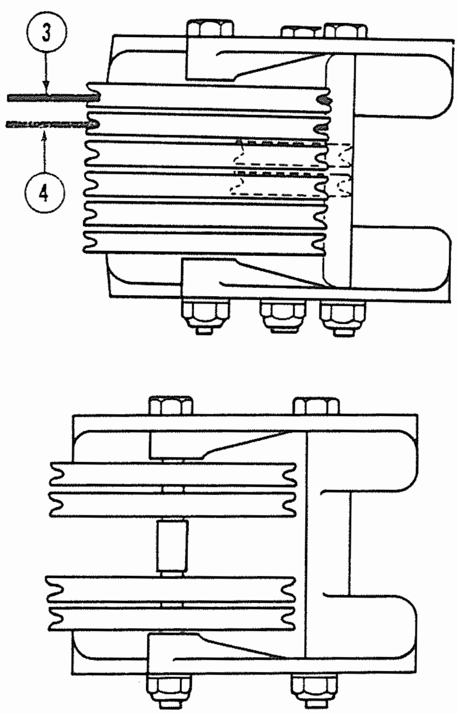
Figure 5-40 (Sheet 3 of 7 Sheets). Carburetor Air Control System — Fairleads, Stations 97 and 156



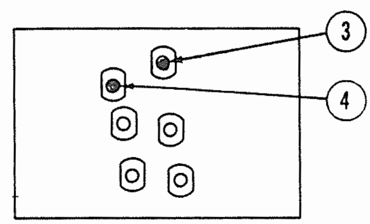
VIEW F — PULLEY BRACKET AT FRONT SPAR
(LOOKING AFT)



VIEW G — FAIRLEAD
AT WING STATION 45
(LOOKING INBOARD)



VIEW H — PULLEY BRACKETS
AT FRONT SPAR IN LH NACELLE
(LOOKING AFT)



VIEW I — FAIRLEAD AT
LH NACELLE STATION 200
(LOOKING AFT)

CARBURETOR AIR — HOT

CARBURETOR AIR — COLD

* NOT USED

1,457

Figure 5-40 (Sheet 4 of 7 Sheets). Carburetor Air Control System — Pulley Brackets, Front Spar and Left Nacelle Front Spar; Fairleads, Left Wing Station 45 and Left Nacelle Station 200

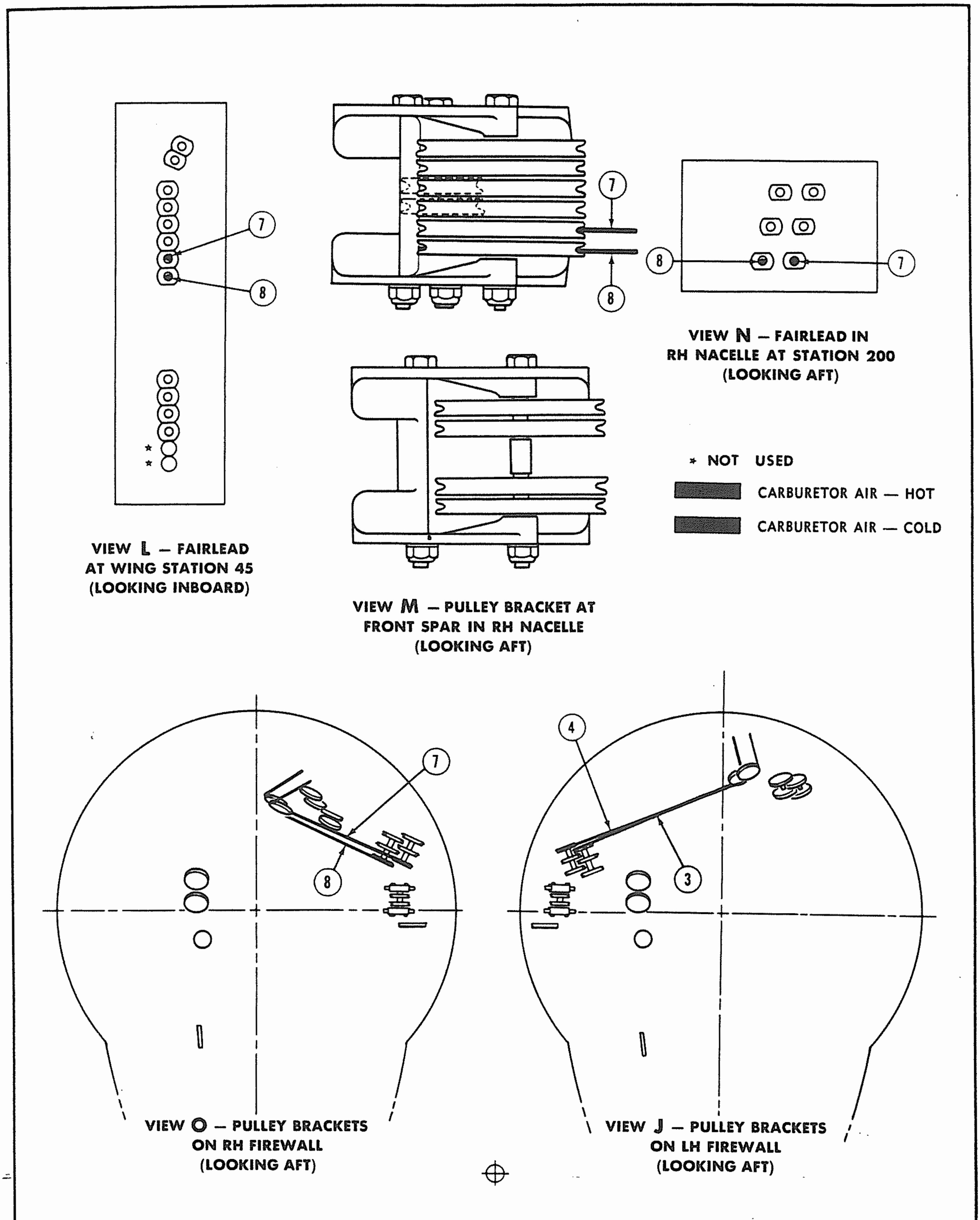
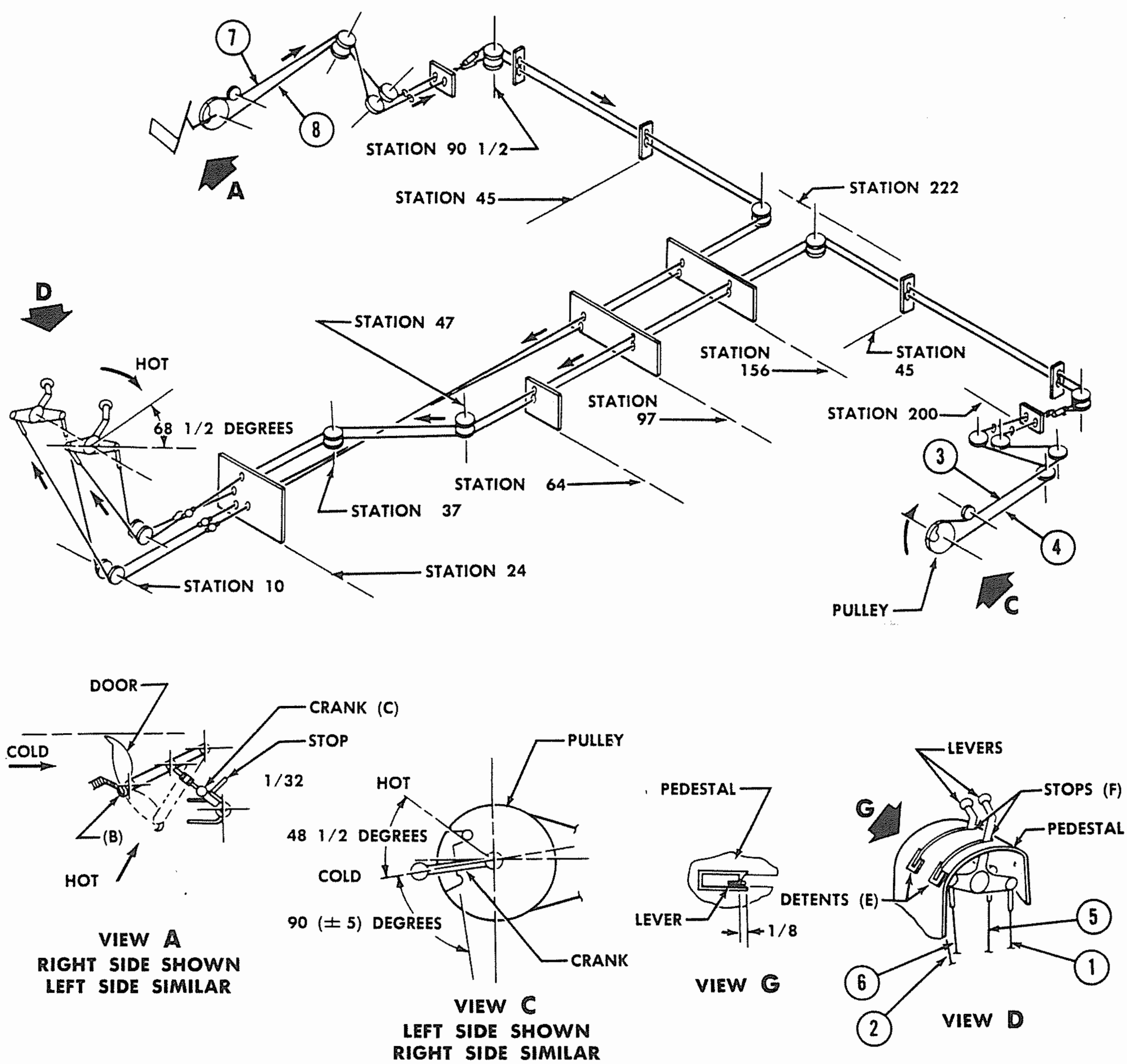


Figure 5-40 (Sheet 5 of 7 Sheets). Carburetor Air Control System — Pulley Brackets, Right Nacelle Front Spar, Firewalls; Fairleads, Right Wing Station 45 and Right Nacelle Station 200



CARBURETOR AIR RIGGING INSTRUCTIONS

1. Set the door assembly in cold air closed position shown in View A and check to assure contact at B.
2. Adjust actuating linkage to obtain a 1/32-inch clearance between the crank and the stop.
3. Rotate the crank counterclockwise until the cold air duct is full open (lip on the upper edge of the door seals at B).
4. Install the pulley, holding a 90 (± 5) degree dimension shown in View C.
5. Set the cockpit control levers in "COLD" position (full forward in detent E, View D) and tension the cables with the turnbuckles in the nacelles in accordance with the Cable Rigging Tension Chart, see figure 2-9.
6. Set the cockpit control levers in "HOT" position shown in View G. The door should be shut, sealing the cold air duct. Deflect the cable system in order to obtain balance of the lever motion necessary to detent.
7. Check to assure approximately 1/8-inch clearance between the levers and the pedestal stops F when the door is full shut. Sealing the cold air duct as shown in View A.
8. Safety the turnbuckles.

8233

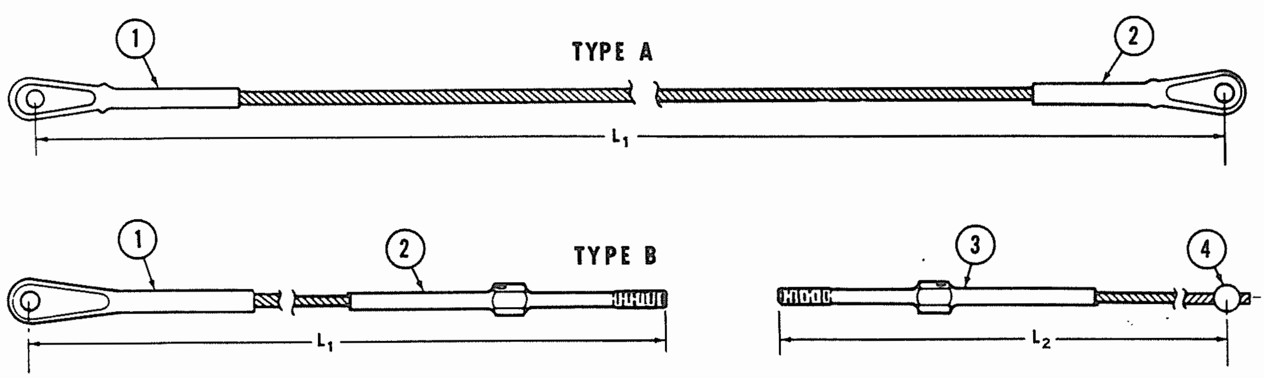
Figure 5-40 (Sheet 6 of 7 Sheets). Carburetor Air Control System — Adjustment Diagram

CARBURETOR AIR CONTROL CABLE CHART — LEFT ENGINE

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)
1	4408789-1 Cold	1	A	34	3/32 dia 7x7 flex	1116245	AN667-3		
2	4408789-503 Hot	1	A	29 1/2	3/32 dia 7x7 flex	1116245	AN667-3		
3	3398717-501 Hot	1	B	(L_1) 337 1/2 (L_2) 69	3/32 dia 7x7 flex	AN668-3	AN669S3 RH	AN669S3 LH	RA2487-3
4	3398717-1 Cold	1	B	(L_1) 332 1/2 (L_2) 70 3/4	3/32 dia 7x7 flex	AN668-3	AN669S3 RH	AN669S3 LH	RA2487-3

CARBURETOR AIR CONTROL CABLE CHART — RIGHT ENGINE

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	FITTINGS (2)	FITTINGS (3)	FITTINGS (4)
5	4408789-501 Cold	1	A	38 1/2	3/32 dia 7x7 flex	1116245	AN667-3		
6	4408789-507 Hot	1	A	33 1/2	3/32 dia 7x7 flex	1116245	AN667-3		
7	3398717-513 Hot	1	B	(L_1) 337 1/4 (L_2) 66 3/4	3/32 dia 7x7 flex	AN668-3	AN669S3 RH	AN669S3 LH	RA2487-3
8	3398717-511 Cold	1	B	(L_1) 332 1/4 (L_2) 61 1/4	3/32 dia 7x7 flex	AN668-3	AN669S3 RH	AN669S3 LH	RA2487-3



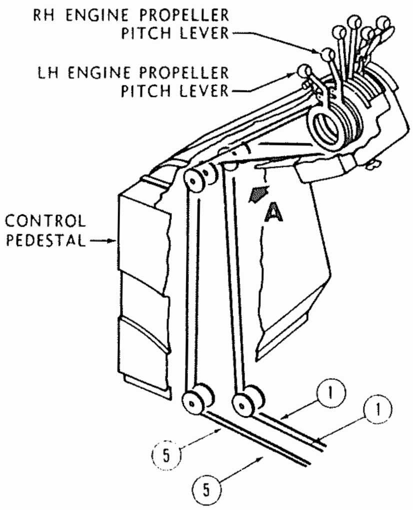
1,459

Figure 5-40 (Sheet 7 of 7 Sheets). Carburetor Air Control System — Cable Charts and Cable Assemblies

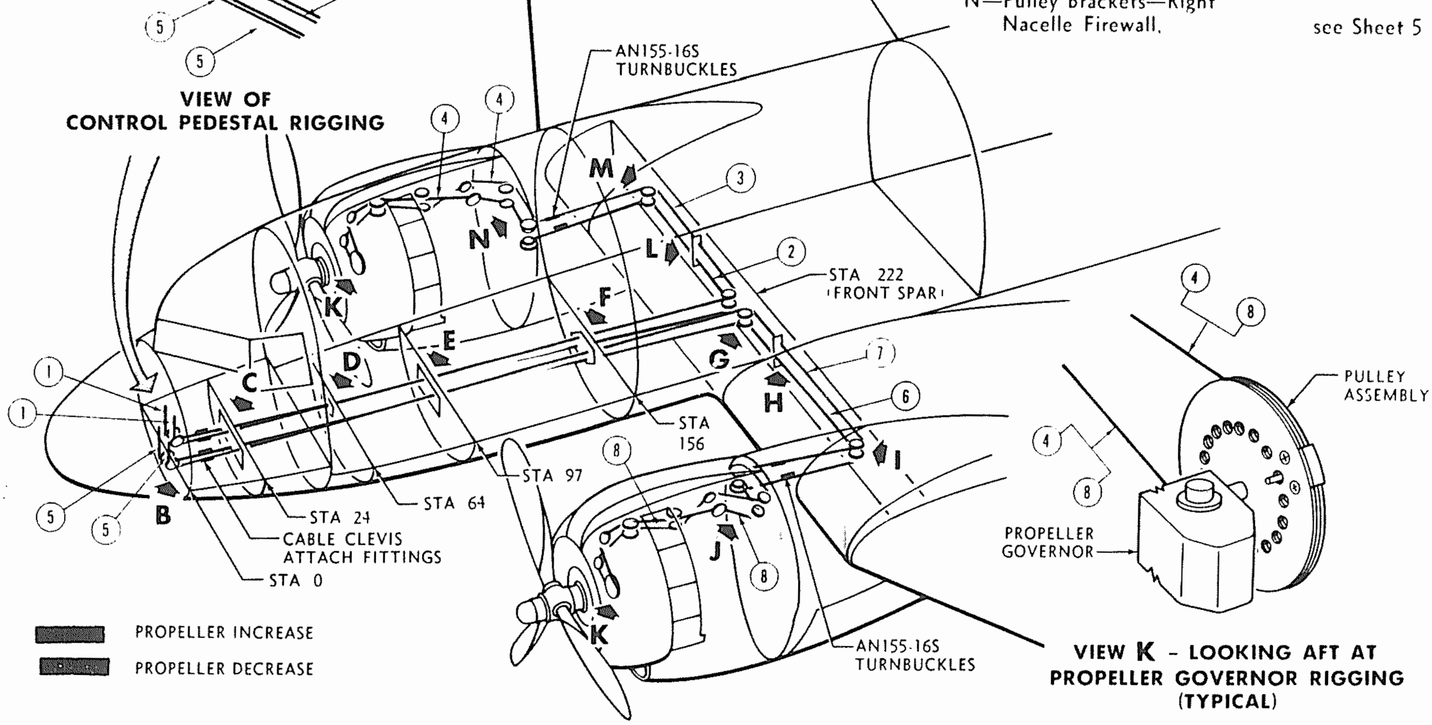
Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 7.
2. For cable assembly details, see Sheet 8.
3. Tension cables in accordance with the Cable Rigging Tension Chart. See Figure 2-9.
4. Tighten all nuts and bolts in accordance with torque values shown in Torque Value Chart, paragraph 5-264.
5. For Rigging and Adjustment Diagram, see Sheet 6.
6. Reference - Douglas drawings 5372291, 5372513, 5372514, 5398252, and 5398557.

A—Pulley Bracket—Control Pedestal.	see Sheet 2
B—Pulley Bracket—Station 10.	see Sheet 2
C—Fairlead—Station 24.	see Sheet 3
D—Fairlead—Station 64.	see Sheet 3
E—Fairleads—Station 97.	see Sheet 3
F—Fairlead—Station 156.	see Sheet 4
G—Pulley Bracket—Front Spar.	see Sheet 4
H—Fairlead—Left Wing Station 45.	see Sheet 4
I—Pulley Brackets—Left Nacelle Front Spar.	see Sheet 4
J—Pulley Brackets—Left Nacelle Firewall.	see Sheet 5
K—Propeller Governor Rigging.	see Sheet 1
L—Fairlead-Right Wing Station 45	see Sheet 5
M—Pulley Brackets—Right Nacelle Front Spar.	see Sheet 5
N—Pulley Brackets—Right Nacelle Firewall.	see Sheet 5



VIEW OF CONTROL PEDESTAL RIGGING



VIEW K - LOOKING AFT AT PROPELLER GOVERNOR RIGGING (TYPICAL)

Figure 5-41 (Sheet 1 of 8 Sheets). Propeller Pitch Control System — Key Drawing

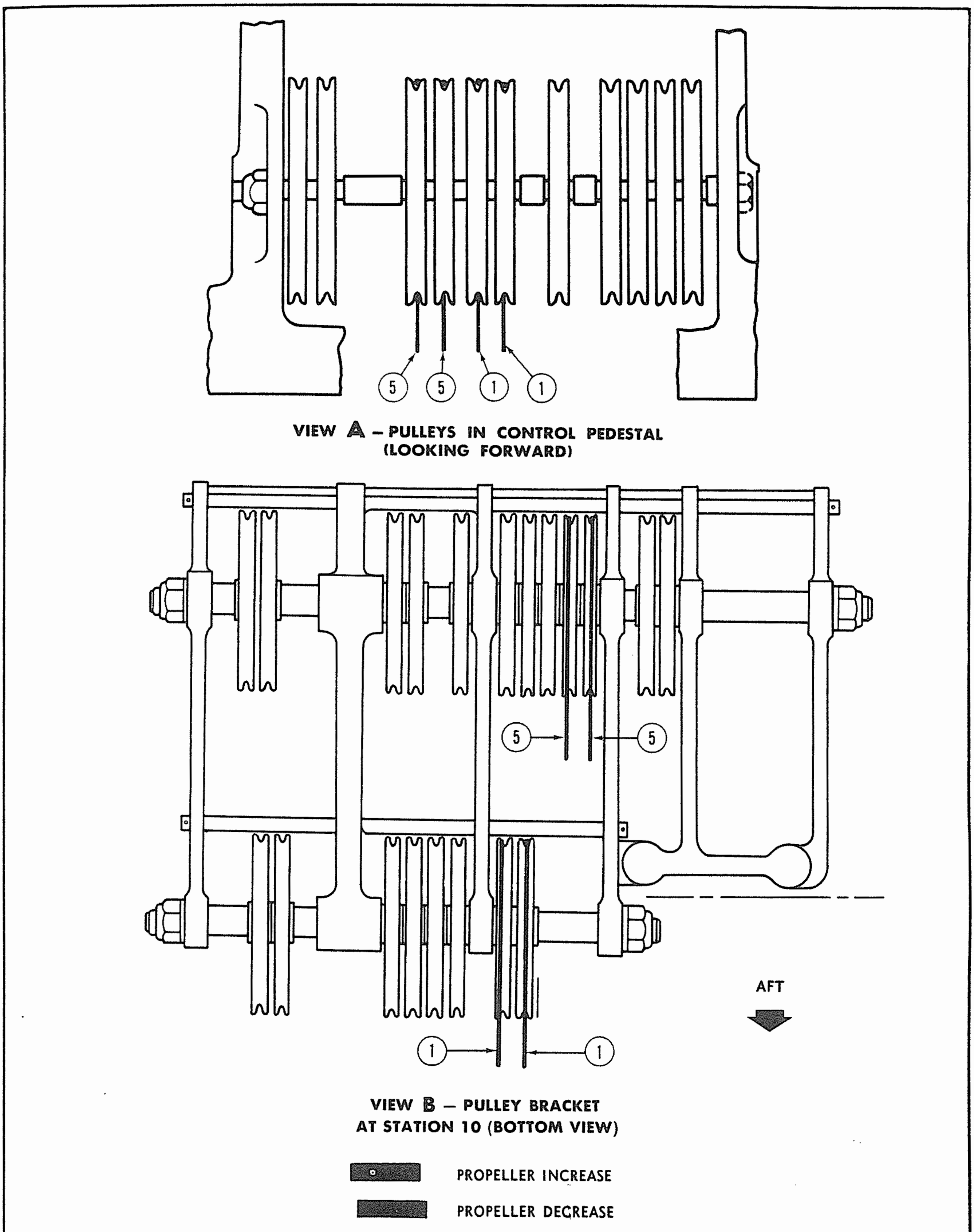
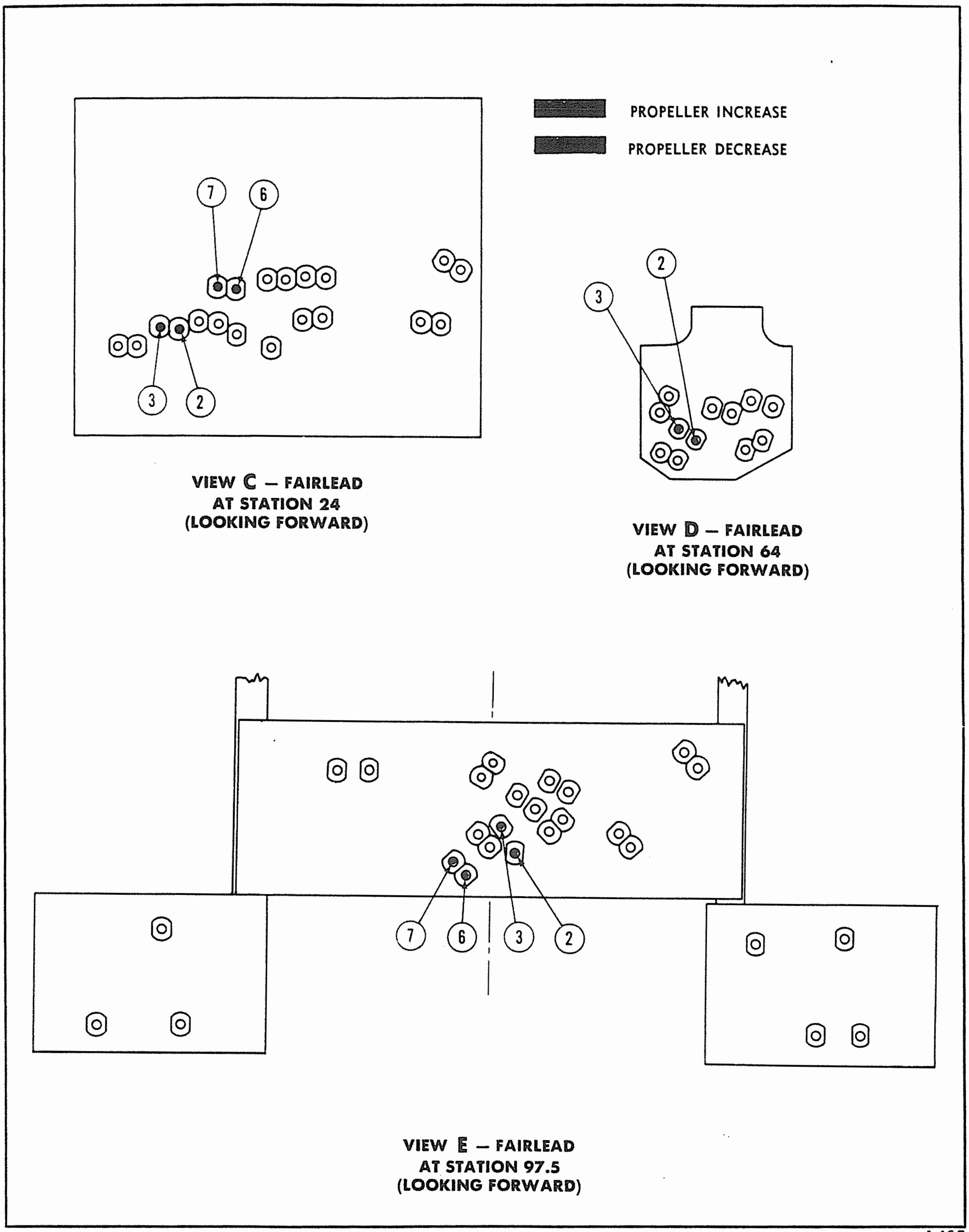


Figure 5-41 (Sheet 2 of 8 Sheets). Propeller Pitch Control System — Pulley Brackets, Control Pedestal and Station 10



VIEW C — FAIRLEAD AT STATION 24 (LOOKING FORWARD)

VIEW D — FAIRLEAD AT STATION 64 (LOOKING FORWARD)

VIEW E — FAIRLEAD AT STATION 97.5 (LOOKING FORWARD)

Figure 5-41 (Sheet 3 of 8 Sheets). Propeller Pitch Control System — Fairleads, Stations 24, 64, and 97

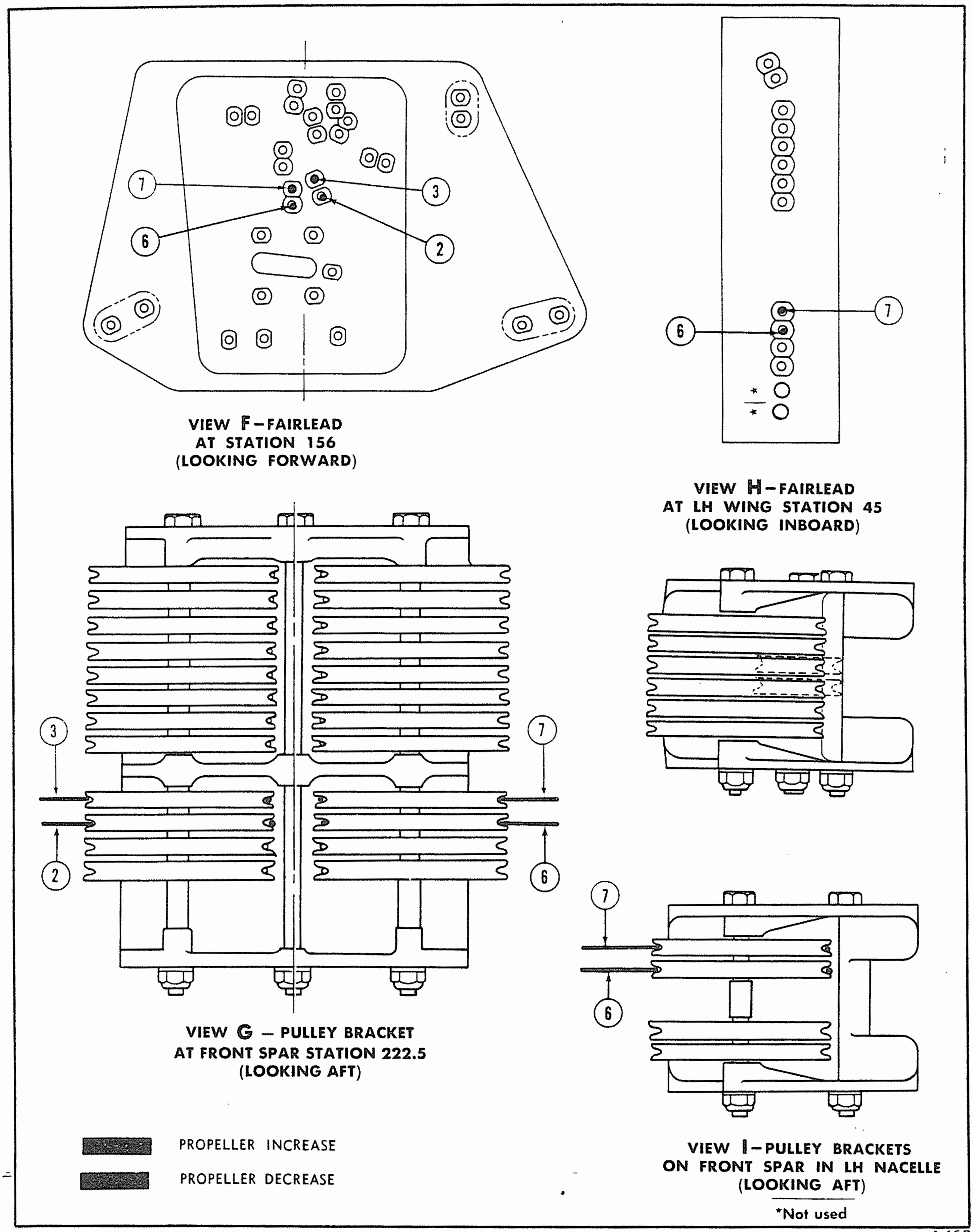


Figure 5-41 (Sheet 4 of 8 Sheets). Propeller Pitch Control System — Pulley Brackets, Front Spar and Left Nacelle Front Spar; Fairleads, Fuselage Station 156 and Left Wing Station 45

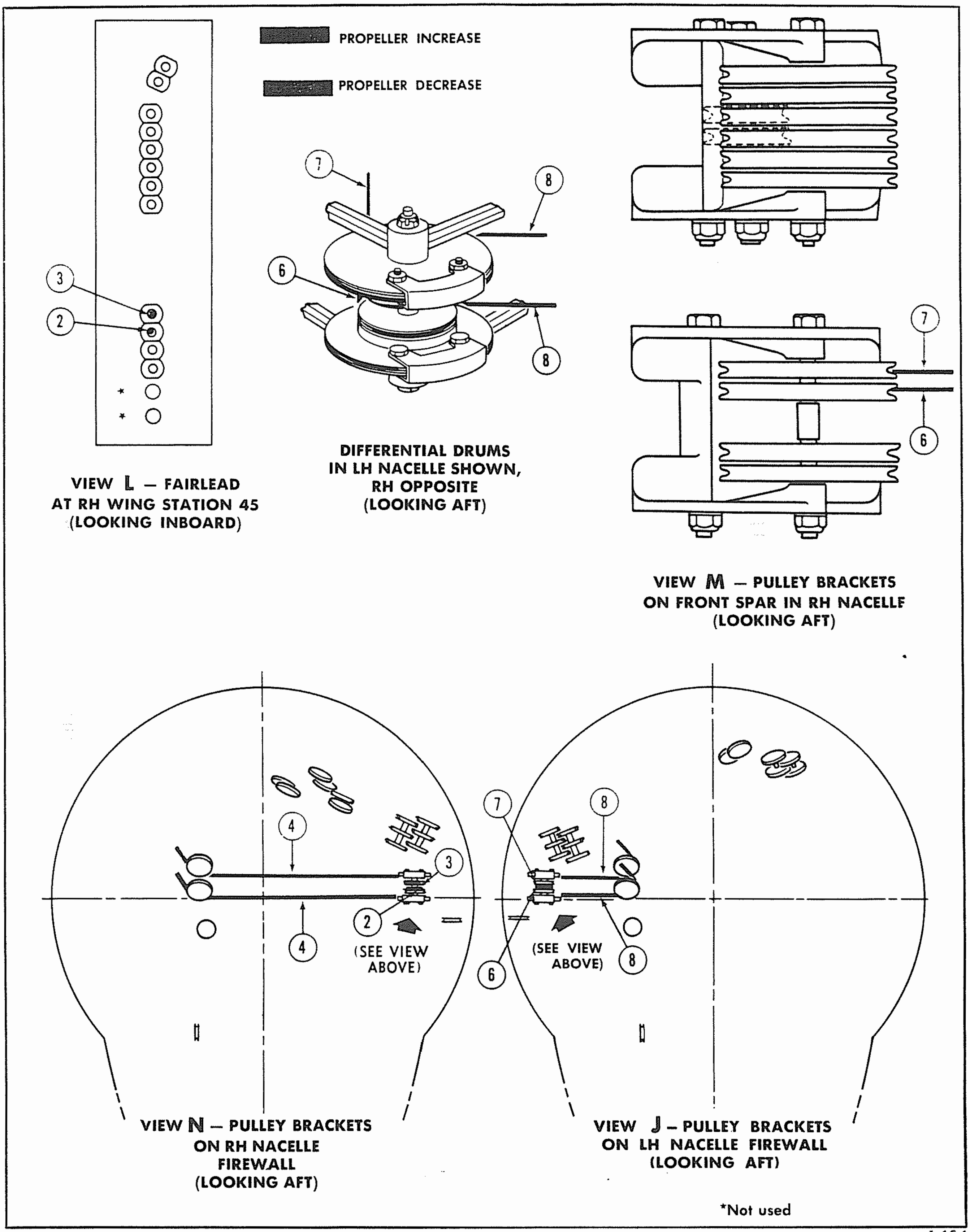
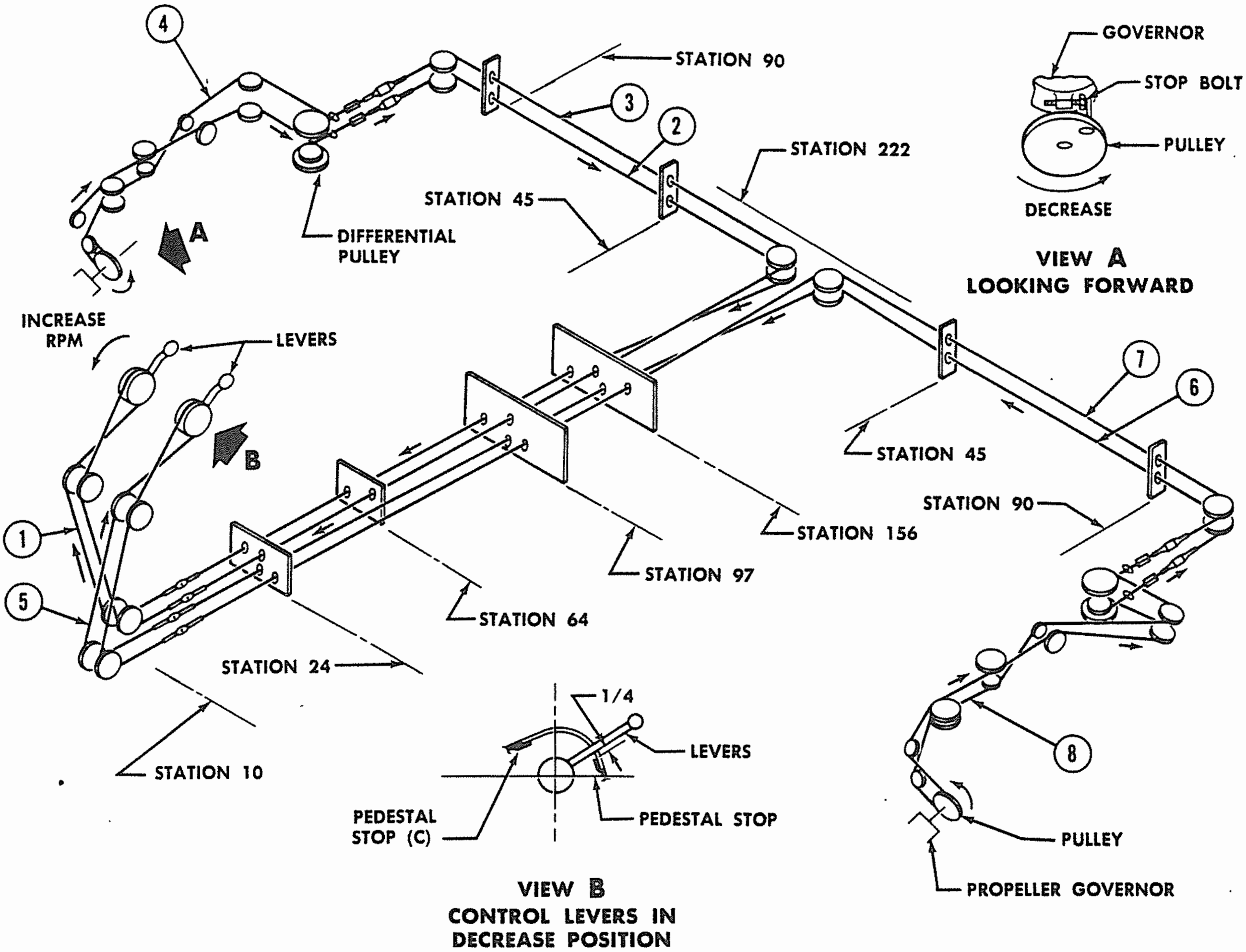


Figure 5-41 (Sheet 5 of 8 Sheets). Propeller Pitch Control System — Pulley Brackets, Right Nacelle Front Spar, Firewalls; Differential Drum; and Fairleads, Right Wing Station 45



PROPELLER PITCH RIGGING INSTRUCTIONS

1. Align the number 6 index mark on the pulley with the mark on the propeller governor shaft.
2. Rotate the pulley full counterclockwise against the governor stop bolt to set the governor in full propeller pitch DECREASE position as shown in View A.
3. Set the cockpit control levers in full "Decrease" position as shown in View B.
4. Tension the cables with the turnbuckles in the nacelles in accordance with the Cable Rigging Tension Chart, see Figure 2-9.
5. A minimum of 45 degree cable warp should remain on both the large and small drums of the differential pulleys.
6. Move the cockpit control levers to full "INCREASE" position (3/16 inch minimum from the pedestal stop C as shown in View B). The propeller governor should be in full increase position.
7. A minimum of 45 degree cable warp should remain on both the large and small drums of the differential pulleys.
8. Safety the turnbuckles.

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Figure 5-41 (Sheet 6 of 8 Sheets). Propeller Pitch Control System — Adjustment Diagram

PROPELLER PITCH CONTROL CABLE CHART — RIGHT ENGINE

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)
1	4115349-5 Increase and Decrease	1	A	101 1/4	3/32 dia 7x7 flex	S-2049218-C-3	S-2049218-C-3		
2	4367764-517 Increase	1	B	(L_1) 307 (L_2) 23 1/4 (L_3) 20	3/32 dia 7x7 flex 1/16 dia 7x7 flex	AN668-3 (5) RA2487-2	AN669S3 RH	AN669S3 LH	2344315
3	4367764-515 Decrease	1	B	(L_1) 317 3/4 (L_2) 7 1/4 (L_3) 20	3/32 dia 7x7 flex 1/16 dia 7x7 flex	AN668-3 (5) RA2487-2	AN669S3 RH	AN669S3 LH	2344315
4	4390837-501 Increase and Decrease	1	C	232 1/4 115 1/4	1/16 dia 7x7 flex	RA2487-2	AN663C2	RA2487-2	

PROPELLER PITCH CONTROL CABLE CHART — LEFT ENGINE

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)
5	4115348-5 Increase and Decrease	1	A	110 1/4	3/32 dia 7x7 flex	S-2049218-C-3	S-2049218-C-3		
6	4367764-513 Increase	1	B	(L_1) 308 (L_2) 23 1/4 (L_3) 20	3/32 dia 7x7 flex 1/16 dia 7x7 flex	AN668-3 (5) RA2487-2	AN669S3 RH	AN669S3 LH	2344315
7	4367764-511 Decrease	1	B	(L_1) 318 1/4 (L_2) 7 1/4 (L_3) 20	3/32 dia 7x7 flex 1/16 dia 7x7 flex	AN668-3 (5) RA2487-2	AN669S3 RH	AN669S3 LH	2344315
8	4390837-1 Increase and Decrease	1	C	(L_1)194 (L_2) 96 1/2	1/16 dia 7x7 flex	RA2487-2	AN663C2	RA2487-2	

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Figure 5-41 (Sheet 7 of 8 Sheets). Propeller Pitch Control System — Cable Charts

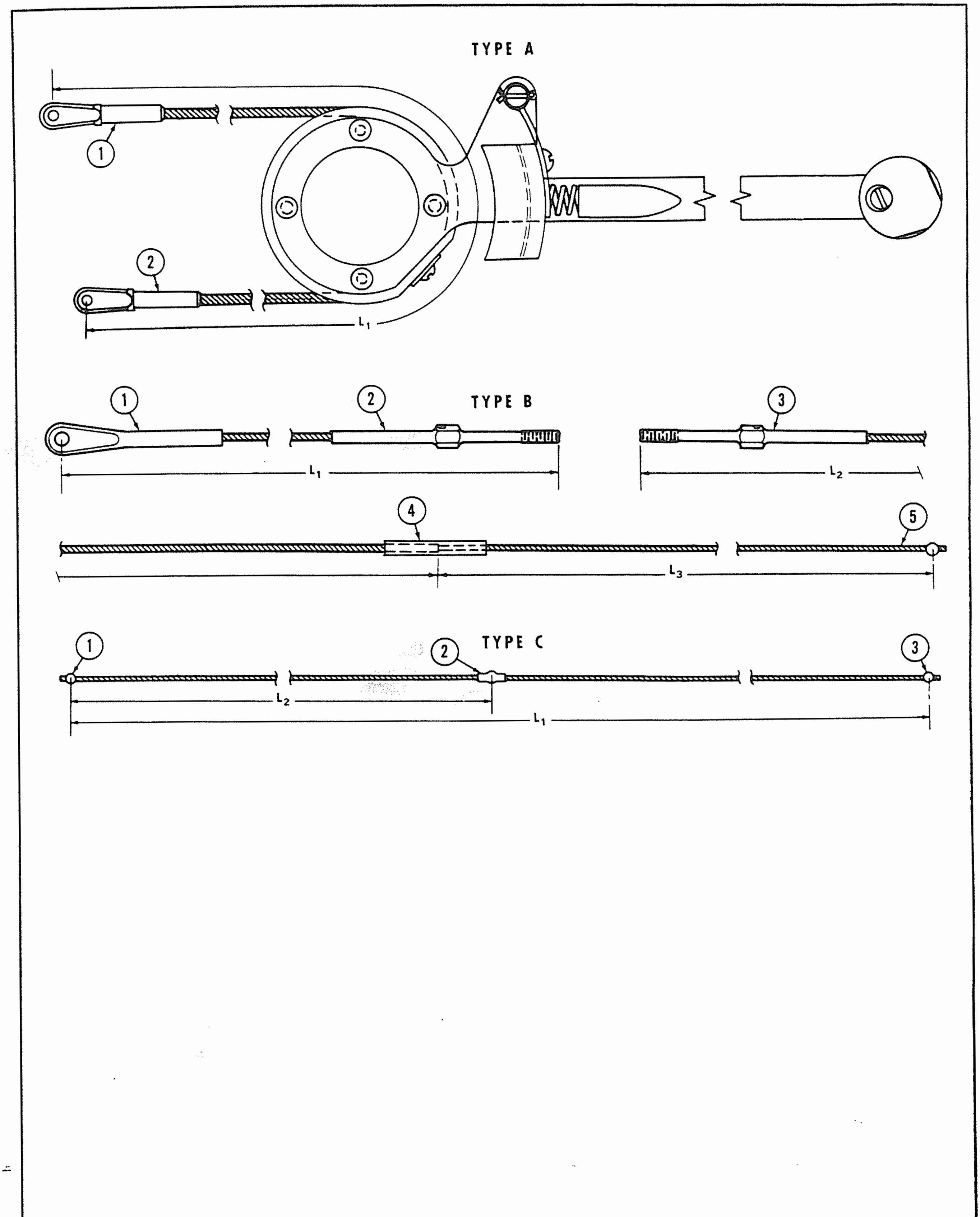

Figure 5-41 (Sheet 8 of 8 Sheets). Propeller Pitch Control System — Cable Assemblies

(Continued from Page 413)

c. Route the main throttle control cables aft through the fuselage and outboard through the wings, replacing fairlead grommets and pulley guard pins as necessary.

d. Connect the four clevis and pin connections just aft of the pulley brackets under the control pedestal.

e. Connect the throttle servo cables to the throttle servos and attach them to the main throttle control cables beween stations 156 and 177.

f. Connect the throttle control cables with turnbuckles aft of the firewalls in the main gear wheel wells.

g. Connect the throttle warning signal levers to the cables just aft of station 64.

h. Adjust the throttle controls (see paragraph 5-243).

i. Tension all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

j. Safety all turnbuckles.

k. Install the access doors and floor panels in the aircraft.

5-243. ADJUSTMENT AND OPERATIONAL TESTS OF THROTTLE CONTROLS. For a complete description of the procedure to be followed in adjusting and testing the throttle control system, see figure 5-38.

Note

Two men are required to make the throttle control adjustments. One man in the cockpit checks the position of the control levers. The other man adjusts the levers and push-pull rods at the power plant. Cable tension adjustments should be made at the turnbuckles aft of the firewalls, at the inboard sides of the main gear wheel wells, in accordance with the Cable Rigging Tension Chart, figure 2-9.

5-244. MIXTURE CONTROLS. *(See figure 5-39.)* The mixture controls consist of two-way cable systems which extend from the mixture controls in the control pedestal, aft through the fuselage and outboard to the mixture control linkage at each power plant carburetor. The mixture controls are actuated by the movement of the mixture control handles on the control pedestal.

5-245. MIXTURE CONTROL CABLES. *(See figure 5-39.)* The mixture control cables extend from the drums in the control pedestal, aft through the lower part of the fuselage, outboard through the wings to the nacelles, and forward through the nacelles to the mixture control pulleys located on the engine accessory sections.

5-246. REMOVAL OF MIXTURE CONTROL CABLES.

a. Disconnect the mixture control cables at the turnbuckles aft of the firewalls in the main gear wheel wells. Thread the cable ends for removal.

b. Remove the access door on the bottom of the fuselage at station 10. Disconnect the four clevis and pin connections just aft of the pulley brackets under the control pedestal and thread the forward cable ends for removal.

c. Remove the necessary access doors on the bottom of the fuselage and center wing sections. Remove the necessary floor panels in the aircraft between the control pedestal and the front wing spar to gain access to the control cables for removal.

d. Remove the necessary fairlead grommets, pulley guard pins and firewall fire seals.

e. Pull the main mixture control cables through the access door at station 10. Pull cables slowly to prevent chipping of the micarta pulleys. Use two men to perform this operation, one to pull the cables and the other to guide the cable ends through the pulley brackets and fairleads.

f. Disconnect the cables from the mixture control pulleys on the right and left engine accessory sections, draw them forward and remove them from the aircraft.

g. Remove the access door on the right side of the control pedestal and the plate over the mixture control drum on the upper section of the control pedestal. Release the cable retaining screws and washers from the drums.

h. Pull one end of each mixture control cable through the pulley bracket under the control pedestal, unwinding the cables from the drums.

5-247. MINOR REPAIR AND REPLACEMENT OF MIXTURE CONTROL CABLES.

a. Examine the push-pull rod bearing ends at the carburetors on the engine accessory sections for freedom of movement. Replace any rods or ends found to be defective.

b. Inspect all pulleys, fairleads, and fire seals for general condition. Replace any badly worn fairleads or fire seals, and damaged pulleys.

c. Check all cables and replace any cable that shows more than six broken wires in any one-inch length, see paragraph 2-51.

5-248. INSTALLATION OF MIXTURE CONTROL CABLES.

a. Install the forward mixture control cables in the control pedestal and wind them on the mixture control drums, using a two-wrap wind. Route the cables through the pulley bracket under the pedestal, replacing guard pins.

b. Screw the cable retaining screws and washers into the mixture control drums, install the plate over the drums and the access door on the right side of the pedestal.

c. Connect the nacelle mixture control cables to the mixture control pulleys on the engine accessory sections. Route them aft through the pulley brackets and firewall, installing necessary pulley guard pins and fire seals.

d. Route the main mixture control cables aft through the aircraft and outboard through the wings, replacing guard pins and grommets.

e. Attach the cables with the clevis and pin connections just aft of the pulley brackets under the control pedestal.

f. Connect the cables with turnbuckles just aft of the firewalls in the main gear wheel wells.

g. Adjust the mixture controls (see paragraph 5-249).

h. Tension all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

i. Safety all turnbuckles.

j. Replace the access doors and floor panels.

5-249. ADJUSTMENT AND OPERATIONAL TESTS OF MIXTURE CONTROLS. For a complete description of the procedure to be followed in adjusting and testing the mixture control system, see figure 5-39.

5-250. CARBURETOR AIR CONTROLS. *(See figure 5-40.)* The carburetor air controls consist of two-way cable systems which extend from the carburetor air control bellcranks in the control pedestal, aft through the fuselage and outboard to the carburetor air control linkage at each power plant carburetor. The carburetor air controls are actuated by the movement of the carburetor control handles on the control pedestal.

Note

Two men are required to make the mixture control adjustments. One man in the cockpit checks the position of the control levers. The other man adjusts the levers and push-pull rods at the power plant. Cable tension adjustments should be made at the turnbuckles aft of the firewalls, at the inboard sides of the main gear wheel wells, in accordance with the Cable Rigging Tension Chart, figure 2-9.

5-251. CARBURETOR AIR CONTROL CABLES. *(See figure 5-40.)* The carburetor air control cables extend from the bellcranks in the control pedestal, aft through the lower part of the fuselage, outboard through the wings to the nacelles, and forward through the nacelles to the carburetor air control pulleys located on the engine accessory sections.

5-252. REMOVAL OF CARBURETOR AIR CONTROL CABLES.

a. Disconnect the carburetor air control cables at the turnbuckles aft of the firewalls in the main gear wheel wells and thread the cable ends for removal.

b. Remove the access door on the bottom of the fuselage at station 10 and disconnect the four clevis and pin connections just aft of the pulley brackets under the control pedestal.

c. Remove the necessary access doors on the bottom of the fuselage and center wing sections. Remove the necessary floor panels in the aircraft between the control pedestal and the front wing spar to gain access to the control cables for removal.

d. Remove the necessary fairlead grommets, pulley guard pins, and fire seals.

e. Pull the main carburetor air control cables through the access door at station 10. Pull cables slowly to prevent chipping of the micarta pulleys. Use two men to perform this operation, one to pull the cables and the other to guide the cable ends through the pulley brackets and fairleads.

f. Disconnect the cables from the carburetor air control pulleys on the engine accessory sections and remove the cables from the aircraft.

g. Remove the access door on the right side of the control pedestal. Detach the cables from the carburetor air control bellcranks and thread the cable ends for removal.

h. Pull the cables through the pulley bracket under the control pedestal and remove them through the access door at station 10.

5-253. MINOR REPAIR AND REPLACEMENT OF CARBURETOR AIR CONTROL CABLES.

a. Examine the push-pull rod bearing ends in the control pedestal and at the carburetors on the engine accessory sections for freedom of movement. Replace any rods or ends found to be defective.

b. Inspect all pulleys, fairleads, and fire seals for general condition. Replace any badly worn fairleads or fire seals, and damaged pulleys.

c. Check all cables and replace any cable that shows more than six broken wires in any one-inch length, (see paragraph 2-51).

5-254. INSTALLATION OF CABURETOR AIR CONTROL CABLES.

a. Connect the forward carburetor air control cables to the carburetor air control bellcranks and route them through the control pedestal, installing guard pins as necessary. Replace the access door on the right side of the control pedestal.

b. Attach the nacelle carburetor air control cables to the carburetor air control pulleys on the engine accessory sections. Route them aft through the nacelles, replacing pulley guard pins and fire seals.

c. Route the main carburetor air control cables aft through the fuselage and outboard through the wings.

d. Attach the cables with clevis and pin connections just aft of the pulley brackets under the control pedestal.

e. Connect the cables with turnbuckles aft of the firewalls in the main gear wheel wells.

f. Adjust the carburetor air controls (see paragraph 5-255).

g. Tension all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

h. Safety all turnbuckles.

i. Replace the access doors and floor panels in the aircraft.

5-255. ADJUSTMENT AND OPERATIONAL TESTS OF CARBURETOR AIR CONTROLS. For a complete description of the procedure to be followed in adjusting and testing the carburetor air control system, see figure 5-40.

Note

Two men are required to make the carburetor air control adjustments. One man in the cockpit checks the position of the control levers. The other man adjusts the crank assemblies and push-pull rods at the power plant. Cable tension adjustments should be made at the turnbuckles aft of the firewalls, at the inboard sides of the main gear wheel wells, in accordance with the Cable Rigging Tension Chart, figure 2-9.

5-256. PROPELLER PITCH CONTROLS. *(See figure 5-41.)* The propeller pitch controls consist of two-way cable systems which extend from the propeller pitch control drums in the control pedestal, aft through the fuselage and outboard to the propeller governor forward of each power plant. The propeller pitch controls are actuated by the movement of the propeller pitch control handles on the control pedestal.

5-257. PROPELLER PITCH CONTROL CABLES. *(See figure 5-41.)* The propeller pitch control cables extend from the propeller pitch control drums in the control pedestal, aft through the lower part of the fuselage, outboard through the wings to the nacelles, and forward through the nacelles to the propeller governor pulleys located forward of the power plant.

5-258. REMOVAL OF PROPELLER PITCH CONTROL CABLES.

a. Disconnect the propeller pitch control cables at the turnbuckles aft of the firewalls in the main gear wheel wells and thread the aft ends for removal.

b. Remove the access door on the bottom of the fuselage at station 10. Unfasten the four clevis and pin connections just aft of the pulley brackets under the control pedestal and thread the forward cable ends for removal from the pedestal.

c. Remove the necesary access doors on the bottom of the fuselage and center wing sections. Remove the necessary floor panels in the aircraft between the control pedestal and the front wing spar to gain access to the control cables for removal.

d. Remove the necessary fairlead grommets, pulley guard pins and fire seals.

e. Pull the main propeller pitch control cables through the access door at station 10. Pull cables slowly to prevent chipping of the micarta pulleys. Use two men to perform this operation, one to pull the cables and the other to guide the cable ends through the pulley brackets and fairleads.

f. Disconnect the aft cables from the propeller pitch differential drums on the forward sides of the nacelle firewalls and remove the cables from the aircraft.

g. Release the forward cables from the differential drums and thread the cable ends for removal.

h. Disconnect the cables from the propeller pitch governor pulleys, forward of the power plant, draw them aft around the power plant, and remove the cables from the aircraft.

i. Remove the access door on the left side of the control pedestal, the plate over the propeller pitch control drums in the upper section of the control pedestal, and the cable retaining screws and washers from the drums.

j. Pull one end of each propeller pitch control cable through the access door at station 10, unwinding the cables from the drums.

5-259. MINOR REPAIR AND REPLACEMENT OF PROPELLER PITCH CONTROL CABLES.

a. Inspect all pulleys, fairleads, and fire seals for general condition. Replace any badly worn fairleads or fire seals, and damaged pulleys.

b. Check all cables and replace any cable that shows more than six broken wires in any one-inch length (see paragraph 2-51).

5-260. INSTALLATION OF PROPELLER PITCH CONTROL CABLES.

a. Install the forward propeller pitch control cables in the control pedestal, and wind them on the propeller pitch control drums with a two-wrap wind.

b. Screw the cable retaining screws and washers into the propeller pitch control drums. Install the plate over the drums and the access door on the left side of the pedestal.

c. Route the nacelle cables around the power plant and connect them to the propeller pitch governor pulleys, installing guard pins as necessary.

d. Connect the forward nacelle cables to the propeller pitch differential drums. Connect the short aft cables to the drums and pass the cable ends aft through the firewalls into the main gear wheel wells, replacing the fire seals.

e. Route the main propeller pitch control cables aft through the aircraft and outboard through the wings, installing fairlead grommets and pulley guard pins.

f. Connect the cables with clevis and pin connections just aft of the pulley brackets under the control pedestal, and with turnbuckles aft of the firewalls in the main gear wheel wells.

g. Adjust the propeller pitch controls (see paragraph 5-261).

h. Tension all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

i. Safety all turnbuckles.

j. Replace the access doors and the floor panels in the aircraft.

5-261. ADJUSTMENT AND OPERATIONAL TESTS OF PROPELLER PITCH CONTROLS. For a complete description of the procedure to be followed in adjusting and testing the propeller pitch control system, see figure 5-41.

Note

Two men are required to make the propeller pitch control adjustments. One man in the cockpit to check the position of the control levers. The other man to adjust the propeller governors at the power plant. Cable tension adjustments should be made at the turnbuckles aft of the firewalls, at the inboard sides of the main gear wheel wells, in accordance with the Cable Rigging Tension Chart, figure 2-7.

5-262. POWER PLANT CONTROL BOLT TORQUE VALUES. For information on the power plant controls bolt torque values, see paragraph 5-263.

5-263. POWER PLANT AND RELATED SYSTEMS BOLT TORQUE VALUES.

5-264. If structural tension bolts are not properly tightened when installed, vibration and fatigue stresses will materially shorten their service life. It is equally dangerous, however, to tighten a bolt excessively, since overtightening may result in stripped threads, a bent or sheared bolt, or eventual tension failure. It is essential, therefore, to observe the torque limits specified here when installing tension bolts. Unless otherwise noted on the installation drawing, bolts and nuts must be installed free of lubricants. Otherwise, the specified torque reading may produce an incorrect tension load. Torque readings should not be made on bolts that show painting or corrosion, as it is impossible to determine accurately the amount of tension being applied. Bolts and nuts that are to be cotter-pinned may be torqued to the special values shown in order to permit the alignment of their holes. However, these figures must never be exceeded, nor may the nuts be backed off to align the holes, unless they are backed off sufficiently to completely re-torque properly. When re-torquing bolts, the nut must be backed off part of a turn and then retightened to the correct value. This is necessary because more torque may be required to break the nut free than is actually on the bolt. To apply 700 inch-pounds and fail to move the nut does not imply that it is torqued to 700 inch-pounds. The nut may have a torque of only 650 inch-pounds, but 750 inch-pounds may be required to break it free. A value in excess of the required torque may also be needed to break the nut free in a counterclockwise direction. For this reason, the nut should be backed off and retightened, allowing no rotation of the bolt. Whenever possible, the torque reading must be taken on the nut and not on the bolt. Torque wrenches should be calibrated frequently to insure accurate readings. When adapters are used with torque wrenches, compensating torque values must be computed for each adapter and wrench involved. The following chart lists the standard torque values for structural tension bolts and also lists the bolts and nuts that should be installed with the torque values specified. Special torques are listed for all bolts to which the standard values are not applicable.

5-265. POWER PLANT AND RELATED SYSTEMS STANDARD BOLT TORQUE VALUE TENSION CHART.

Bolt Size	*AN Type Bolts*	*AN365 and AN310 Nuts*	*Torque Values*	*Cotter Pin Maximum Torque*
10-32	AN-3 AN-173	-1032	20-25 inch-pounds	40 inch-pounds
¼-28	AN-4 AN-174	-428	50-70 inch-pounds	100 inch-pounds
5/16-24	AN-5 AN-175	-524	100-140 inch-pounds	225 inch-pounds
⅜-24	AN-6 AN-176	-624	160-190 inch-pounds	390 inch-pounds
7/16-20	AN-7 AN-177	-720	450-500 inch-pounds	840 inch-pounds
½-20	AN-8 AN-178	-820	480-690 inch-pounds	1100 inch-pounds
9/16-18	AN-9 AN-179	-918	800-1000 inch-pounds	1600 inch-pounds
⅝-18	AN-10 AN-180	1018	90-105 foot-pounds	200 foot-pounds
¾-16	AN-12 AN-182	1216	190-210 foot-pounds	415 foot-pounds
⅞-14	AN-14 AN-184	1414	210-250 foot-pounds	590 foot-pounds
1-14	AN-16 AN-186	1614	310-460 foot-pounds	835 foot-pounds
1⅛-12	AN-18	1812	415-585 foot-pounds	1250 foot-pounds

Bolt Size	*NAS Heat-Treated Bolts*	*Douglas Heat-Treated Bolts*	*Elastic Stop Heat-Treated Nuts*	*Torque Values*
¼-28	NAS-144	S-2076904	12B-048	73-100 inch-pounds
5/16-24	NAS-145	S-2076905	12B-054	145-200 inch-pounds
⅜-24	NAS-146	S-2076906	12B-064	230-280 inch-pounds
7/16-20	NAS-147	S-2076907	12B-070	650-720 inch-pounds
½-20	NAS-148	S-2076908	12B-080	700-1000 inch-pounds
9/16-18	NAS-149	S-2076909	12B-098	100-120 foot-pounds
⅝-18	NAS-150	S-2076910	12B-108	135-155 foot-pounds
¾-16	NAS-152	S-2076912	12B-126	280-300 foot-pounds
⅞-14	NAS-154	S-2076914	12B-144	300-360 foot-pounds
1-14	NAS-156	S-2076916	12B-164	450-665 foot-pounds
1⅛-12	NAS-158	S-2076918	12B-182	605-845 foot-pounds

5-266. POWER PLANT AND RELATED SYSTEMS SPECIAL BOLT TORQUE VALUE TENSION CHART.

Location	*Attachment*	*Instl. Drawing*	*Bolt Number*	*Nut Number*	*No. Required Per Aircraft*	*Torque*
Engine Section	Exhaust Sys.	5397847	AN5C-11A	AN363-524	8	175-200 inch-pounds
	Firewall Fitting to Firewall	5367686	2356375	EB-054	32	145-200 inch-pounds
	Engine Mt. to Firewall	5372929	NAS150	4367565	8	1620-1820 inch-pounds
	Lord Mt. to Engine Mt.	5372929	RC-27-21	RC-27-22	32	325-400 inch-pounds
	Engine Case to Lord Mt.	5372929	RC-27-7	RC-27-8	16	750-850 inch-pounds
	Lord Mt. Assembly	5372929	RC-27-25	RC-27-15	32	60-80 inch-pounds
	Prop. Feathering Pressure Surge Dampener	3406882	2409623	AN310-10	2	72-84 foot-pounds (160 foot-pounds max. for cotter pin alignment)

AN 01-40NK-2

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z AIRCRAFT

SECTION VI INSTRUMENT SYSTEMS

THIS SECTION SUPERSEDES SECTION VI OF AN 01-40NK-2
DATED 15 MAY 1952 REVISED 15 DECEMBER 1952

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15 April 1953

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SECTION VI

INSTRUMENT SYSTEMS

TABLE OF CONTENTS

SECTION VI

INSTRUMENT SYSTEMS

6-1. INSTRUMENT SYSTEMS.

6-2. DESCRIPTION. *(See figure 6-1.)* The instrument systems, which provide visual indications of the performance of the various aircraft systems, terminate at the instrument panels in the flight compartment. The instruments are grouped in the panels with regard to the particular function they serve. For example, the flight and navigation instruments are duplicated in some instances so that they may be installed directly in front of both the pilot and co-pilot; the engine instruments, on the other hand, are grouped in a position at the center of the panel. With the exception of the hydraulic instrument panel, all panels are secured to the fuselage structure with vibration-absorbing shock mounts. The panels can be removed to facilitate the inspection and maintenance of the instruments.

6-3. The direct reading instruments consist of the following:

- Dual Manifold Pressure Indicator
- Dual Fuel Pressure Indicator
- Dual Oil Pressure Indicator
- De-Icing Pressure Indicator
- Oxygen Flow Indicator
- Oxygen Pressure Indicator
- Alighting Gear Pressure Gage
- Hydraulic System Pressure Gage
- Emergency Air Brake Pressure Gage

6-4. The electrically actuated instruments consist of the following:

- Dual Tachometer Indicator
- Dual Oil Temperature Indicator
- Dual Carburetor Air Temperature Indicator
- Dual Cylinder Temperature Indicator
- Turn-and-Bank Indicator
- Gyro Horizon Indicator
- Trim Indicator (Automatic Pilot)
- Radio Altimeter Indicator
- Master Direction Indicator
- Dual ADF (Aircraft A, B, and 1 through 68)
- Cross Pointer Indicator
- Radio Magnetic Indicator (Aircraft C, D, and 69 through 96)
- Omni-Bearing Receiver Magnetic Bearing Indicator
- Multiple-Type Fuel Quantity Indicator
- Alcohol Quantity Indicator
- Outer Wing Fuel Quantity Indicators
- Heater Temperature Indicators
- Alighting Gear Position Indicator
- Outside Air Temperature Indicator
- Ammeters

6-5. The pitot-static operated instruments consist of the following:

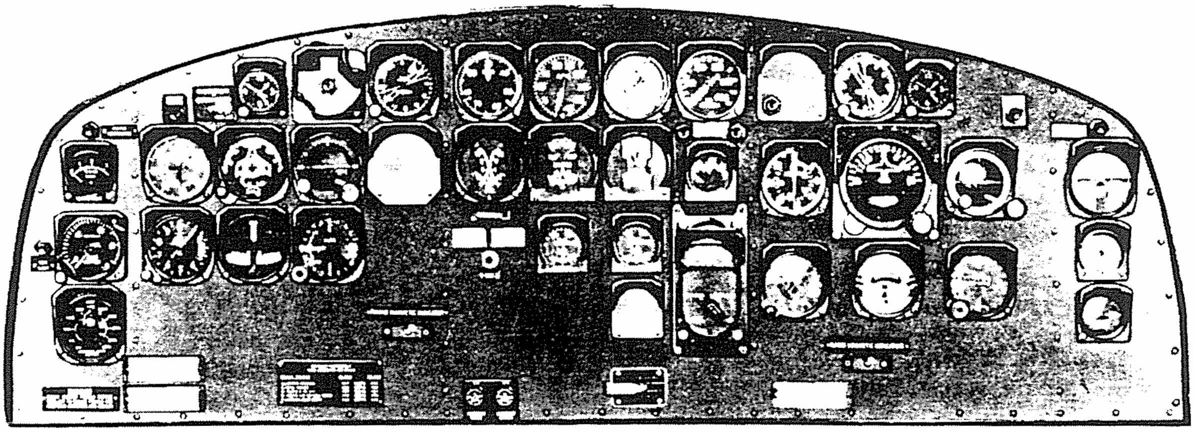
- Airspeed Indicators
- Altimeters
- Rate-of-Climb Indicators

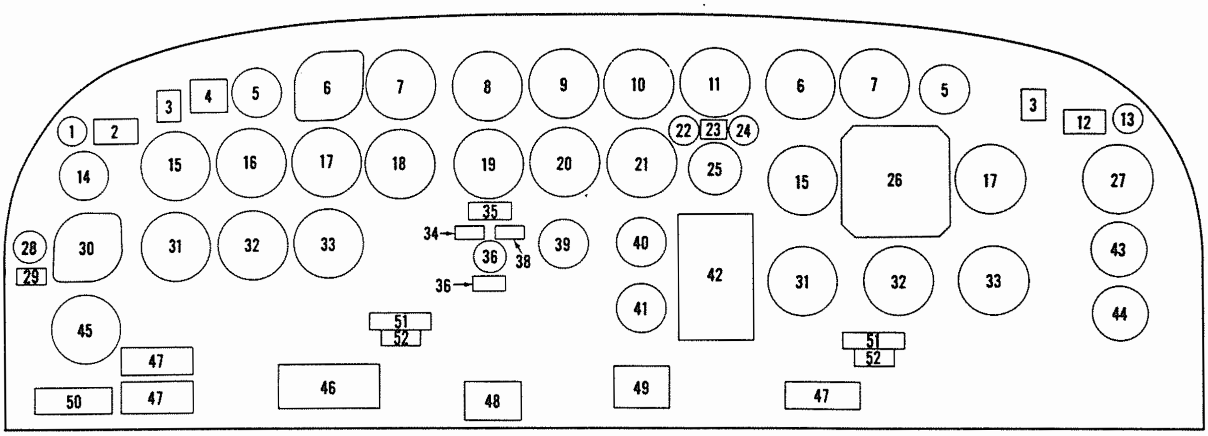
6-6. The miscellaneous instruments consist of the following:

- Clocks
- Gyro Stabilized Driftmeter
- Magnetic Compass

6-7. TROUBLE SHOOTING OF INSTRUMENT SYSTEMS.

Trouble	*Probable Cause*	*Remedy*
GENERAL		
a. Any instrument on the main instrument panel inoperative.	Faulty instrument or malfunctioning of instrument system.	Localize trouble by substituting a similar instrument. If substitution does not remedy trouble, check instrument system affected as described in work operations.
RATE-OF-CLIMB INDICATOR		
b. Pointer off zero position.	Shaft out of adjustment.	Return pointer to zero by means of zero adjustment shaft.
c. Pointer off zero position and cannot be brought back by zero shaft adjustment.	Pivot broken.	Replace instrument.





1. Heater Inoperative Warning Light
2. Heater Inoperative Warning Light Placard
3. Inverter Warning Light (Two)
4. G-2 Compass Control
5. Clock
6. Marker Beacon (Two)
7. Dual ADF Indicator (Two) Aircraft A, B, 1 through 68 Radio Magnetic Indicator (Two) Aircraft C, D, 69 through 96
8. Manifold Pressure Gage
9. Tachometer
10. Fuel Pressure Gage
11. Oil Pressure Gage
12. Gear Unsafe Warning Light Placards
13. Gear Unsafe Warning Light
14. Heater Temperature Gage
15. Air Speed Indicator (Two)
16. Master Direction Indicator
17. Gyro Horizon Indicator (Two)
18. Radio-Magnetic Indicator (Aircraft A, B, 1 through 68)
19. Cylinder Temperature Indicator
20. Carburetor Air Temperature Indicator
21. Oil Temperature Indicator
22. Fuei Pressure Warning Light
23. Fuel/Oil Pressure Warning Light
24. Oil Pressure Warning Light
25. Outside Air Temperature Indicator
26. Directional Gyro Indicator
27. Landing Gear Position Indicator
28. Low-Limit Warning Light
29. Low-Limit Warning Light Placard
30. Radio Altimeter
31. Altimeter (Two)
32. Turn-and-Bank Indicator (Two)
33. Rate-of-Climb Indicator (Two)
34. Left Engine Fire Warning Light
35. Engine Fire Warning Light Placard
36. Engine Fire Warning Test Switch
37. Engine Fire Warning Test Switch Placard
38. Right Engine Fire Warning Light
39. Left Outer Wing Fuel Quantity Indicator
40. Right Outer Wing Fuel Quantity Indicator
41. Trim Indicator (Aircraft C, D, and 59 through 96)
42. Center Wing Fuel Quantity Indicator
43. Alcohol Quantity Indicator
44. De-Icer System Pressure Gage
45. Radio Altimeter Limit Switch
46. Maximum Permissible True Indicated Airspeed Placard
47. Card Holder (Three)
48. Manifold Pressure Purge Valves
49. Static Pressure Selector Valve
50. Rubber Pedal Adjustment Check Warning Placard
51. Radio Magnetic Indicator Placard
52. Radio Magnetic Indicator Selector Switch

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Figure 6-1. Main Instrument Panel

Trouble	*Probable Cause*	*Remedy*
d. Pointer fails to respond.	Obstruction in static pipe.	Disconnect all instruments connected to static pipe. Open drain plug and blow pipe clean.
e. Instrument indicates less than actual rate of climb.	Instrument case not airtight.	Replace instrument.
	Leaking hose connection at instrument.	Tighten connection.
f. Instrument indicates sluggishly.	Instrument assembly dirty or worn.	Replace instrument.
g. Instrument pointer oscillates.	Water in static pipes.	Drain static manifolds.
	Leak in instrument static pipe.	Disconnect all instruments operating on static pipe. Check and correct pipe for leaks. Check all instruments in static system for leaks.
AIRSPEED INDICATOR		
h. Pointer fails to respond.	Pressure connection from pitot tube not connected to pitot pressure pipe. Pitot or static pressure pipe from pitot or static system clogged.	Check tubing connection from airspeed indicator to pitot and static systems for correctness and leaks. Disconnect pitot and static pipes from all instruments. Open static drain. Blow out system with dry air.
	Defective indicator mechanism.	Replace instrument.
i. Pointer indicates incorrectly.	Leak in pitot system pipes.	Check tubing connections from airspeed indicator through pitot systems for leaks.
	Leak in indicator case.	Replace instrument.
j. Pointer oscillates.	Water in pipes.	Drain pitot-tube sump.
	Leak in tubing from pitot systems or in connections.	Disconnect pipes from airspeed indicator. Check pitot pipes for leaks.
	Leak in indicator case.	Replace instrument.
ALTIMETER		
k. Instrument fails to operate.	Water in pipes.	Drain static manifolds.
	Static pipes blocked.	Disconnect all instruments operating on static pressure pipes. Blow these pipes clear with clean, dry air.
	Instrument damaged by handling or excessive air pressure.	Replace instrument.
l. Instrument indicates incorrectly.	Water in pipes.	Drain static manifolds.
	Incorrect pipe connections.	Check system connections.
	Static selector valve on alternate source or leaking internally.	Check position of static selector valve and make certain that it is not leaking.
	Pointer not set correctly.	Reset pointer to correct position by means of knob.
	Slow leak in diaphragm.	Replace instrument.
	Instrument not correctly calibrated.	Replace instrument.
m. Pointer oscillates.	Water in pipes.	Drain static manifolds.
	Irregularity of static pressures received at instrument.	Check static pressure pipes for partial block. Clean pipes and tighten connections if necessary.
	Defective instrument.	Replace instrument.
n. Instrument operates sluggishly.	Defective, worn, or dirty instrument assembly.	Replace instrument.
ENGINE MANIFOLD PRESSURE INDICATING SYSTEM		
o. Inaccurate gage readings.	Condensate in pipes to gage.	Open purge valves and drain condensate.
	Leak in pipes of fittings.	Check pipes and connections for leaks.
	Faulty gage.	Replace instrument.
FUEL AND OIL PRESSURE INDICATING SYSTEM		
p. Inaccurate indicator readings or no reading.	Leak in pipes.	Check pipes and connections for leaks.
	Faulty indicator.	Replace indicator.
HYDRAULIC PRESSURE INDICATING SYSTEM		
q. Inaccurate gage readings.	Leaks in pipes to gage.	Check pipes and connections for leaks.
	Faulty gage.	Replace gage.
TACHOMETER		
r. No reading on indicator.	Faulty indicator.	Replace instrument.
	Faulty tachometer generator.	Replace tachometer generator.
	Faulty wiring between indicator and tachometer generator.	Check and repair wiring.

Trouble	*Probable Cause*	*Remedy*
EMERGENCY AIR BRAKE PRESSURE INDICATING SYSTEM		
s. Inaccurate gage readings.	Faulty gage.	Replace gage.
BULB-TYPE TEMPERATURE INDICATING SYSTEMS		
t. No indicator reading.	Circuit breaker open or faulty.	Check circuit breaker.
	Open in power lead to indicator.	Repair wiring.
	Faulty indicator.	Replace instrument.
u. Indicator reading off scale at low-temperature end.	Short in lead to resistance bulb.	Repair lead.
	Ground in lead from resistance bulb to indicator.	Repair lead.
	Short circuit in resistance bulb.	Replace bulb. **CAUTION** Be sure that electric power is OFF before removing bulb.
	Faulty indicator.	Replace instrument.
v. Indicator reading off scale at high-temperature end.	Break in leads to resistance bulb.	Repair leads.
	Open circuit in resistance bulb.	Disconnect power immediately; then replace bulb.
	Faulty indicator.	Replace instrument.
w. Low or high reading, either permanent or intermittent.	Low battery.	Replace or charge battery.
	Loose connections.	Clean and tighten connections.
	Poor connection in leads from indicator to resistance bulb.	Clean and tighten connection.
THERMOCOUPLE-TYPE TEMPERATURE INDICATING SYSTEMS		
x. No reading, either intermittent or permanent.	Break in lead or thermocouple.	Replace lead or thermocouple.
	Defective indicator.	Replace indicator.
y. Low reading, either intermittent or permanent.	Poor connections.	Clean and tighten connections.
	Short circuit at indicator binding posts, at thermocouple, or in leads.	Repair circuit.
	Zero corrector shift.	Reset corrector.

Trouble	*Probable Cause*	*Remedy*
LIQUIDOMETER FLUID QUANTITY INDICATING SYSTEMS		
z. Failure to indicate.	Break in ground lead.	Repair lead.
	Loose connection at either indicator or tank unit.	Tighten connection.
	Open circuit in indicator.	Replace instrument.
	Binding in indicator.	Replace instrument.
	Open circuit in tank unit.	Repair or replace tank unit.
	Binding at float arm bearing.	Remove tank unit and repair.
	Reversed polarity.	Correct power supply.
aa. Low reading.	Poor connection at indicator or tank unit.	Clean and tighten connections.
	Binding in indicator.	Replace instrument.
	Binding at float arm bearing.	Remove tank unit and repair.
	Stroke adjustment shift.	Readjust stroke.
ab. Center scale reading.	Break in contact wiper arm lead.	Repair lead.
ac. High reading on indicator.	Short circuit between ground and wiper arm leads.	Repair circuit.
	Reversed polarity.	Correct polarity by reversing electric leads.
	Stroke adjustment shift.	Readjust stroke.
MAGNETIC COMPASS		
ad. Compass does not indicate correctly.	Magnetic body in close proximity.	Check and remove tools, equipment, and other magnetic material from vicinity of compass.
	Compass not properly compensated.	Swing and compensate compass. If compensation will not bring compass within tolerance, replace compass.
GYRO HORIZON INDICATOR		
ae. Bar fails to respond.	Circuit incorrectly wired.	Correct circuit wiring.
	Faulty caging mechanism.	Replace instrument.
	Worn or damaged parts.	Replace instrument.
af. Bar does not settle.	Worn or damaged parts.	Replace instrument.

TURN-AND-BANK INDICATOR

Trouble	Probable Cause	Remedy
ag. Pointer fails to respond.	Circuit incorrectly wired.	Correct circuit wiring.
	Defective instrument.	Replace instrument.
ah. Pointer does not set on zero.	Defective instrument.	Replace instrument.
ai. Pointer vibrates.	Defective instrument.	Replace instrument.
aj. In low temperatures, pointer fails to respond, or responds sluggishly and with an insufficient deflection.	Oil is too heavy or bearing clearance is insufficient.	Replace instrument.
ak. Pointer sluggish in returning to zero and does not set on zero when indicator is stationary.	Instrument assembly is dirty or worn.	Replace instrument.
al. Ball not centered in straight and level flight.	Incorrect mounting of instrument.	Level aircraft with spirit level, and shift instrument to align with leveled aircraft.

G-2 COMPASS

Trouble	Probable Cause	Remedy
am. Readings 180 degrees in error.	Faulty wiring of master direction-indicator cable.	Check and correct cable.
	Improper phasing of amplifier.	Replace amplifier.
an. Readings 120 or 240 degrees in error.	Faulty wiring of transmitter and master direction-indicator cables or open leads in the cables.	Check and correct cables.
ao. Pointer turns in reverse direction to airplane.	Faulty wiring of master direction-indicator and transmitter cables.	Check and correct cables.
ap. Pointer either swings through small arc or indicates first one and then the other of two headings as the airplane is turned.	Faulty master direction-indicator cable or transmitter cable.	Check and correct cables.
	Faulty transmitter.	Replace transmitter.
	Faulty amplifier.	Replace amplifier.
aq. Pointer does not indicate proper magnetic reading.	Failure of power supply.	Check voltage at indicator.
	Faulty power supply connections to instrument.	Check power supply, cable plugs, and receptacles.
	Faulty wiring between master indicator and compass transmitter.	Check wiring system.
	Faulty compass transmitter.	Replace compass transmitter.
	Transmitter not properly aligned.	Align transmitter.
	Defective master indicator.	Replace master indicator.

PITOT-STATIC SYSTEM

Trouble	Probable Cause	Remedy
ar. Pitot and static-pressure operated instruments do not operate correctly.	Water in pipes.	Drain main and alternate static manifolds. Drain pitot sumps.
	Pitot or static pressure pipes obstructed.	Disconnect instruments from pitot-static pressure systems and blow out pipes with clean, dry air.
	Pitot or static pressure pipes crimped or collapsed.	Check pipes and replace if necessary.
	Pitot tube opening obstructed.	Clean pitot tube head, or replace if necessary.
	Clogged static openings.	Clean static openings.
	Pitot pipes and static pipes interchanged at instruments.	Check pipe connections at instruments.
	Static selector valve defective.	Replace selector valve.
	Defective instruments.	Replace instruments.

RADIO MAGNETIC INDICATOR (AIRCRAFT C, D, AND 69 THROUGH 96)

Trouble	Probable Cause	Remedy
as. Pointer fails to respond.	Circuit incorrectly wired.	Correct the wiring.
	Defective instrument.	Replace the instrument.
	Faulty selector switch.	Replace the switch.
at. Inaccurate reading.	Faulty wiring or poor connections to the pilot's or co-pilot's receivers.	Correct the wiring.
	Inaccurate compass system.	Check the compass system trouble shooting procedures.

6-8. MAIN INSTRUMENT PANEL. *(See figure 6-1.)* The main instrument panel is fabricated in one piece of 1/8-inch thick aluminum alloy. Appropriate cutouts are provided in the panel to accommodate the various instruments.

6-9. REMOVAL OF MAIN INSTRUMENT PANEL. Access to the instrument panel is through the hinged access door in the nose of the fuselage.

a. Disconnect the plumbing and the electrical wiring to the instruments at the instrument panel.

b. Remove the eight aluminum-alloy bolts at the instrument panel shock mounts and remove the panel.

6-10. INSTALLATION OF MAIN INSTRUMENT PANEL. Reverse the removal procedure.

6-11. HYDRAULIC INSTRUMENT PANEL. *(See figure 6-1.)* The hydraulic instrument panel is attached to the fuselage longerons to the right of the main instrument panel. The hydraulic instrument panel contains the hydraulic system pressure gage and the landing gear pressure gage.

6-12. REMOVAL OF HYDRAULIC INSTRUMENT PANEL.

a. Remove the cockpit lining in the area of the instrument panel assembly.

b. Disconnect the plumbing leads to the instruments.

c. Remove the instrument panel assembly from the aircraft by removing the attaching bolts and nuts.

d. Remove the instruments from the panel assembly.

6-13. INSTALLATION OF HYDRAULIC INSTRUMENT PANEL. Reverse the removal procedure.

6-14. OVERHEAD ELECTRICAL PANELS. The overhead electrical panels are each fabricated in one piece from 0.040-inch thick aluminum alloy. Appropriate cutouts are provided to accommodate the equipment. For additional information, see figures 7-16 and 7-17.

6-15. ELECTRICALLY ACTUATED INSTRUMENTS. For a complete list of the electrically actuated instruments, see paragraph 6-4.

6-16. REMOVAL OF ELECTRICALLY ACTUATED INSTRUMENTS. The following general removal procedure is applicable to all electrically actuated instruments:

a. Remove the screws securing the instrument to the panel.

b. Remove the instrument from the aft side of the panel, and disconnect the electrical leads.

c. Remove the instrument.

6-17. INSTALLATION OF ELECTRICALLY ACTUATED INSTRUMENTS. Reverse the removal procedure.

6-18. DIRECT READING INSTRUMENTS. For a complete list of direct reading instruments, see paragraph 6-3.

6-19. REMOVAL OF DIRECT READING INSTRUMENTS. The following general removal procedure is applicable to all direct reading instruments:

a. To remove the hydraulic system pressure indicator, the alighting gear pressure indicator, or the emergency air brake pressure indicator, the system pressure must be released. To relieve hydraulic pressure, operate the wing flaps until both gages indicate zero. To relieve the air brake pressure, loosen the valve body slightly on the air brake cylinder. Do not relieve air pressure by depressing the valve core.

b. On all instruments, remove the screws attaching the instrument to the panel.

c. Disconnect the pipes and remove the instrument.

6-20. INSTALLATION OF DIRECT READING INSTRUMENTS. Reverse the removal procedure.

6-21. PITOT-STATIC OPERATED INSTRUMENTS. For a complete list of the pitot-static operated instruments, see paragraph 6-5.

6-22. REMOVAL OF PITOT-STATIC OPERATED INSTRUMENTS.

a. Remove the screws securing the instrument to the panel.

b. Disconnect the attaching pipes and remove the instrument.

6-23. INSTALLATION OF PITOT-STATIC OPERATED INSTRUMENTS. Reverse the removal procedure.

6-24. TEST OF PITOT-STATIC OPERATED INSTRUMENTS.

CAUTION

The following tests should be made by qualified instrument personnel only. Test pressures must be applied and released slowly and gently to avoid damaging the instrument.

6-25. AIRSPEED INDICATOR LEAK TEST.

a. Disconnect the static pipe as close to the indicator as possible, and attach a length of rubber tubing to the instrument.

b. Blow gently into the other end of the tubing until the pointer reaches 200 mph. Then pinch off the tube.

c. If the pointer descends faster than an indicated one mile per hour in the elapsed time of one minute, a leak in the diaphragm is indicated, and the instrument must be replaced.

d. Connect a mercury manometer into the test pipe and repeat step b, preceding. Case leakage should not exceed 0.4 inches Hg in 10 seconds.

6-26. PITOT PRESSURE (AIRSPEED) PIPE LEAK TEST.

a. Attach a hose to the pitot pressure (airspeed) opening at the end of the tube, making certain that the connection is airtight.

b. Blow *gently* into the tube until the airspeed indicator reads 150 mph. Then pinch off the source of pressure.

CAUTION

Do not apply suction to the pitot pressure (airspeed) opening of the pitot-static pipe.

c. During a 1-minute period, the indicator pointer should not change appreciably from the 150 mph reading. The indicator case must be tapped with the fingers during the test to overcome the effects of friction on the pointer.

d. If the pointer indication varies by more than 10 mph, check the pipes for cracks or faulty connections.

6-27. ALTIMETER CASE LEAK TEST.

a. Disconnect the static pressure pipe from the altimeter.

b. Connect a length of rubber tubing to the altimeter vent and, by blowing gently, decrease the altitude indication by only 500 feet.

c. Pinch off the tube and watch the pointer for 10 seconds. Tap the indicator case to eliminate friction. The indication should not change more than 20 feet.

d. If the change is more than 20 feet during the 10-second period, a case leak is indicated and the altimeter must be replaced.

6-28. RATE-OF-CLIMB INDICATOR CASE LEAK TEST.

a. Disconnect the static pipe from the instrument, and attach a rubber hose and portable manometer to the inlet port of the instrument case.

b. Set the indicator pointer to zero by means of the adjusting screw on the face of the instrument.

c. Gently and *slowly* apply a suction of 15 inches Hg at a rate not to exceed 1500 feet per minute ascent, then pinch off the vacuum pipe near the instrument. The level of the mercury should not change more than 0.4 inches Hg in 10 seconds.

6-29. PITOT-STATIC SYSTEM.

6-30. DESCRIPTION. A pitot and static pressure system is installed for the operation of the altimeter, airspeed, and rate-of-climb indicators. The pitot-static system consists of two pitot tubes projecting from the fuselage nose, a main and an alternate static source, a static selector valve, and the necessary tubing *(see figure 6-2)*. The pitot tubes are protected against an accumulation of ice by integral heating elements. For a description of the pitot heaters, see paragraph 4-89.

6-31. TROUBLE SHOOTING OF PITOT-STATIC SYSTEM. For information on trouble shooting of the pitot-static system, see paragraph 6-7.

6-32. PITOT TUBES. *(See figure 6-2.)* Two pitot tubes are installed on the nose section of the fuselage. The left pitot tube supplies ram air to the pilot's airspeed indicator and the right pitot tube supplies ram air to the co-pilot's airspeed indicator. Each pitot tube is a separate assembly, installed at an angle of —3 degrees to the longitudinal axis of the aircraft, in order to improve the accuracy of the pitot system when the aircraft approaches a stall condition. A sump is incorporated in each pitot tube pipe to prevent water from entering the pressure pipe to the instrument. A drain valve is installed at the lowest point of each sump. Each pitot tube has an electric heating element, near the opening, to prevent ice from collecting and causing erroneous readings on the airspeed indicators. The pitot tube heater switch is located on the right overhead electrical panel.

6-33. REMOVAL OF PITOT TUBES.

a. From within the fuselage nose section, remove the four screws from the pitot sleeve, just inside of the fuselage skin.

b. Remove the three screws from the external portion of the pitot tube sleeve, and pull the sleeve out and over the pitot tube head.

c. Disconnect the pitot tubing at a point just inside of the fuselage nose section. Disconnect the pitot head anti-icing unit connector, and pull the pitot head unit out to remove it.

6-34. INSTALLATION OF PITOT TUBES. Reverse the removal procedure.

6-35. STATIC VENTS. *(See figure 6-2.)* In the main static system, a static pressure vent is flush-mounted in the skin of the fuselage, on each side of the aircraft, at station 560, at a point where the pressure equals static pressure for all conditions of flight. In the alternate static system, the static vent is located within the tail cone section at station 625; this vent provides an ice-free static source in the event that the main static vents are inoperative due to icing, entry of foreign matter, or damage to the system piping. The main static vents, each riveted over a small opening in the skin, consist of a two-inch diameter metal disc, incorporating four air inlet holes, to which the tubing is connected. The main static pipes and the alternate static pipes are each routed through their own drain manifolds, located in the nose section of the fuselage, forward of the flight compartment.

6-36. STATIC SYSTEM MANIFOLDS. *(See figure 6-2.)* The static system incorporates both a main and an alternate system manifold. In both systems the vent pipe is routed to a manifold, and from each manifold to a common static selector valve. Both the main static system manifold and the alternate system manifold, located in the fuselage nose section forward of the flight compartment, are provided with a drain valve to permit draining of accumulated moisture from the respective static systems.

6-37. REMOVAL OF STATIC SYSTEM MANIFOLDS.

a. Remove and cap the plumbing pipes on the inlet and outlet ports of both the main and alternate static manifolds.

b. Remove the three mounting bolts and nuts from the common support bracket for both manifolds, and remove the manifolds.

6-38. INSTALLATION OF STATIC SYSTEM MANIFOLDS. Reverse the removal procedure.

6-39. STATIC SYSTEM PRESSURE SELECTOR VALVE. *(See figure 6-2.)* The static system pressure selector valve, located on the main instrument panel center section, is used to select either the main or alternate static pressure source. The static selector valve is connected by 1/4-inch hose to both the main and alternate static manifolds.

6-40. REMOVAL OF STATIC SYSTEM PRESSURE SELECTOR VALVE.

a. Remove the screws securing the selector valve to the instrument panel.

b. Remove and cap the tubing leading to the selector valve, and remove the valve.

6-41. INSTALLATION OF STATIC SYSTEM PRESSURE SELECTOR VALVE. Reverse the removal procedure.

6-42. SUCTION TEST OF STATIC SYSTEM.

a. Connect a source of suction to one of the static pressure sources.

b. Move the static selector valve to the corresponding static pressure source.

c. Set the altimeter pointers at zero.

d. Apply the suction slowly in a sufficient amount to cause the altimeter pointers to indicate 1000 feet.

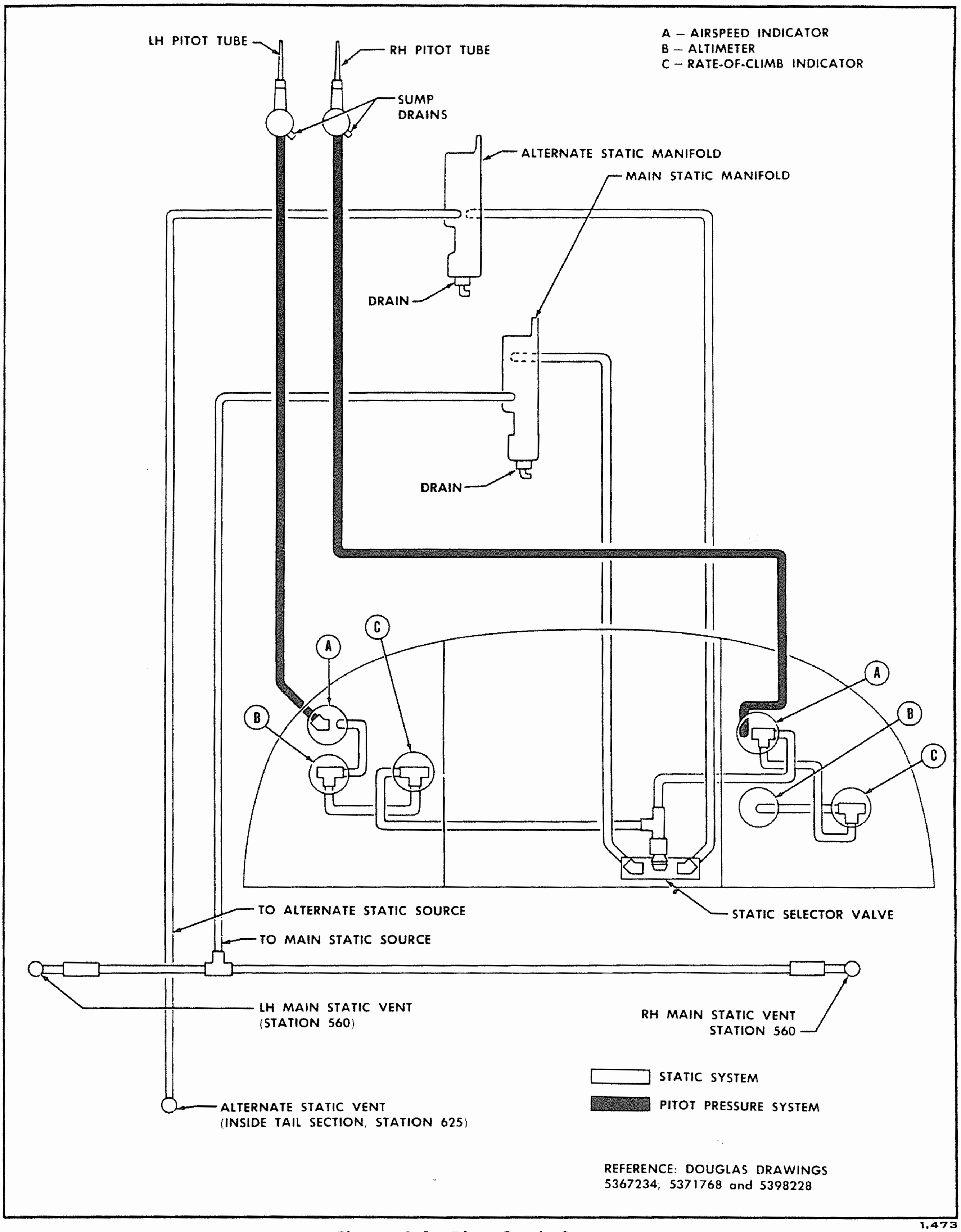


Figure 6-2. Pitot-Static System

A suction of approximately 1.05 inches Hg or 14.24 inches of water is required.

Note

During a period of 1 minute, the altimeter pointers should not change their positions by more than 150 feet.

e. Tap each altimeter indicator or the instrument panel lightly during the one-minute period to eliminate the effects of friction on pointer indication.

f. Regulate the application and removal of suction by noting the rate-of-climb indication. This reading must not exceed 1500 feet per minute.

CAUTION

Do not apply pressure to the static pressure pipes.

6-43. PRESSURE TEST OF PITOT PRESSURE PIPES.

a. Close the pitot pressure chamber sump drain valves.

b. Connect a source of pressure to the left pitot pressure opening.

c. Make certain that the airspeed indicator is at zero.

d. Slowly apply pressure. Do not allow the airspeed indicator to increase to a point over 300 mph. Pinch off the air pressure.

Note

Allowable leakage is seven mph per minute at 150 mph.

e. Tap the airspeed indicator or the instrument panel during the one-minute period to eliminate any effect of friction on the pointer.

f. Repeat the above procedure for the right airspeed pitot tube and for the airspeed indicator.

CAUTION

Do not apply suction to the pitot pressure pipes.

6-44. WARNING SYSTEMS.

6-45. DESCRIPTION. The warning systems installed in this aircraft consist of the alighting gear warning signals and horn, the cabin heater inoperative warning light, the cabin heater fire warning light, the engine fire warning lights, fuel and oil pressure warning lights, and the door-open warning light.

6-46. ALIGHTING GEAR WARNING SIGNAL LIGHTS AND HORN. *(See figure 6-1.)* The alighting gear warning circuit provides visual and audible warning if the alighting gear or throttle positions are not coordinated with the flight conditions. An alighting gear position indicator and a red gear-unsafe lamp are installed on the co-pilot's instrument panel, to provide visual indications of the alighting gear position; a warning horn is installed in the cockpit to provide audible indications when the gear is in an unsafe position for landing. Two switches, one actuated by each throttle control cable, cause the warning horn to sound if either throttle is retarded to a position below that which is necessary for a safe flight speed, except when the alighting gear is fully down and latched. A gear-up switch, operated by the nacelle alighting gear door, is opened when the doors are fully closed. The tail gear-up switch is opened by the tail wheel fork when the tail gear assembly is in the fully retracted position. The opening of all three switches de-energizes the gear-unsafe light and shows, by means of indicators in the three windows of the alighting gear position indicator, that the three gears are fully retracted. A switch, mounted on each gear latch unit, supplies bus voltage to the alighting gear position indicator; this switch shows the silhouette of an alighting gear wheel through each of the three windows of the alighting gear position indicator when the alighting gear is down and latched and when the power is on the aircraft electrical bus. A fourth switch, so installed that the circuit is opened when the alighting gear control lever is in the DOWN position, is also connected to the alighting gear UNSAFE light which will glow if the lever is in any intermediate position. This switch is required since hydraulic pressure, in addition to the alighting gear latches, is employed to retain the alighting gear in the DOWN position.

6-47. CABIN HEATER INOPERATIVE WARNING LIGHT. The cabin heater inoperative warning light is located on the pilot's instrument panel, just above the heater temperature indicator. The light comes on if the heater switch is turned to the ON position when the drop-out fuse is blown, as a result of excessive temperatures in the heater, or when the scoop air inlet valve is closed.

6-48. CABIN HEATER FIRE WARNING LIGHT. The cabin heater fire warning light is located on the pilot's electrical switch panel, just above the cabin heater ON-OFF switch. The light comes on when the fire detectors in the heater compartment and/or the heater duct are actuated by fire or excessive temperatures. A test switch, located adjacent to the warning light, tests wire leads to the fire detectors for circuit continuity.

6-49. ENGINE FIRE WARNING LIGHTS. Fire detectors of the thermocouple-type are located in Zones 1 and 2 of the engine section. The warning lights are mounted on the main instrument panel. The test switch is located adjacent to the warning lights and is used to test the entire system, including the detectors. The Edison sensitive relay box is installed in the radio rack on the second shelf from the top.

6-50. FUEL AND OIL PRESSURE WARNING LIGHTS. The fuel and oil pressure warning lights are located adjacent to the fuel and oil pressure indicators. When a light comes on, low fuel or oil pressure is indicated. The pressure warning switches are installed in the fuel and oil pipes in the engine accessories compartment to actuate the lights.

6-51. DOOR-OPEN WARNING LIGHT. A red door-open warning light is located above the ignition switch, between the two overhead electrical panels. The light is actuated by either the forward service door microswitch or the main cargo door push-button type switch if either of the doors is not properly closed.

6-52. AUTOMATIC PILOT SYSTEM.

6-53. DESCRIPTION. *(See figures 6-3 and 6-4.)* The Pioneer P-1A automatic pilot system is an electrically controlled and actuated system that automatically operates the flight controls of the aircraft to maintain a desired heading and attitude. The system controls the aircraft through the use of three electrically driven gyros. The relative positions of the vertical and rate gyros and the G-2 compass system are constantly compared with the position of the aircraft by electric signals, which are electronically amplified to control the three servo motors. Operation of the automatic pilot depends upon the signals from the vertical gyro, which indicate the attitude of the aircraft about the pitch and bank axes; signals from the G-2 compass system, which indicate the heading of the aircraft; and signals from the rate gyro, which indicate the rate of course deviation.

6-54. In addition a controller, mounted on the control pedestal, provides a means of manual control of the automatic pilot for coordinated turns or for a change in bank or pitch attitude. The system is self-synchronous and can be engaged into the aircraft control system at any time without changing the aircraft attitude; however, the aircraft should be trimmed before engaging the automatic pilot to permit equal travel of the servos about the normal central null.

6-55. In aircraft A, B, and 1 through 58, the automatic pilot system consists of the following components: a gyro horizon indicator and a turn-and-bank indicator, located on the pilot's instrument panel; a G-2 compass transmitter, located in the vertical stabilizer; a master direction indicator, mounted on the pilot's main instrument panel; a compass amplifier and signal generator unit, mounted on the left side of the fuselage forward of the main junction box; a G-2 compass coupling adapter, servo amplifier, and power junction box, located in the top of the radio rack; three controlling servos and a servo amplifier adapter junction box, located in the lower fuselage at station 156; a mechanical engaging control on the lower aft face of the control pedestal; and release buttons on the control wheel. In aircraft C, D, and 59 through 96, the following components are added to the automatic pilot system: a trim indicator, located on the pilot's main instrument panel; a barometric altitude control unit, located in the fuselage nose section; an altitude control switch, mounted on the pedestal controller; and a trim tab servo and servo adapter, located in the lower fuselage at station 177. The automatic pilot can be turned on whenever the main inverter switch is positioned to PILOT'S INSTR A-C RADIO AND AUTO-PILOT. Ground operation of the system, required for maintenance or preflight functional testing, is possible through use of the ground load limiting interlock override switch, mounted on the cockpit right sidewall.

Note

The override switch must be returned to the OFF position at the termination of any ground operation of the automatic pilot in order to prevent inadvertent overloading of the generator system.

The differential pulley assembly is installed between the automatic pilot servos and the aircraft primary control system, as a means for reducing the linear displacement of the control cables caused by the rotation of the servos.

Note

The automatic pilot system will not operate when the auxiliary inverter is the only source of a-c power.

The automatic pilot can be instantly DISENGAGED mechanically by depressing the automatic pilot release button, located on each control wheel.

Note

If the main inverter switch is moved out of the PILOT'S INSTR A-C RADIO AND AUTO-PILOT position to either of the two remaining positions, the automatic pilot will be disengaged automatically from the flight control system and will be electrically de-energized.

6-56. On aircraft C, D, and 59 through 96, automatic approach control equipment is installed in addition to the Pioneer P-1 automatic pilot system components described in paragraph 6-55.

6-57. AUTOMATIC APPROACH CONTROL (AIRCRAFT C, D, AND 59 THROUGH 96). *(See figure 6-4.)* The automatic approach system provides a means of instrument guidance for the aircraft to an "on-course" landing approach under low ceiling and visibility conditions. The system controls the aircraft by means of signals, applied to the throttle and control-surface servo operating circuits, which are originated by the flight path computer through interpretation of the localizer and glide path radio beam signals. These signals are the same as those applied to the instrument landing cross pointer, on the pilot's main instrument panel, which provides visual indication of the position of the aircraft relative to the localizer and glide path beams. The automatic approach system can be disengaged simultaneously with the automatic pilot by depressing the release button on either control wheel or by turning the pedestal selector switch to the OFF position. The automatic approach system incorporates the following components: an automatic approach selector switch, mounted on the pedestal controller; a flight path computer and throttle servo amplifier, located in the radio rack; a throttle servo and throttle servo re-

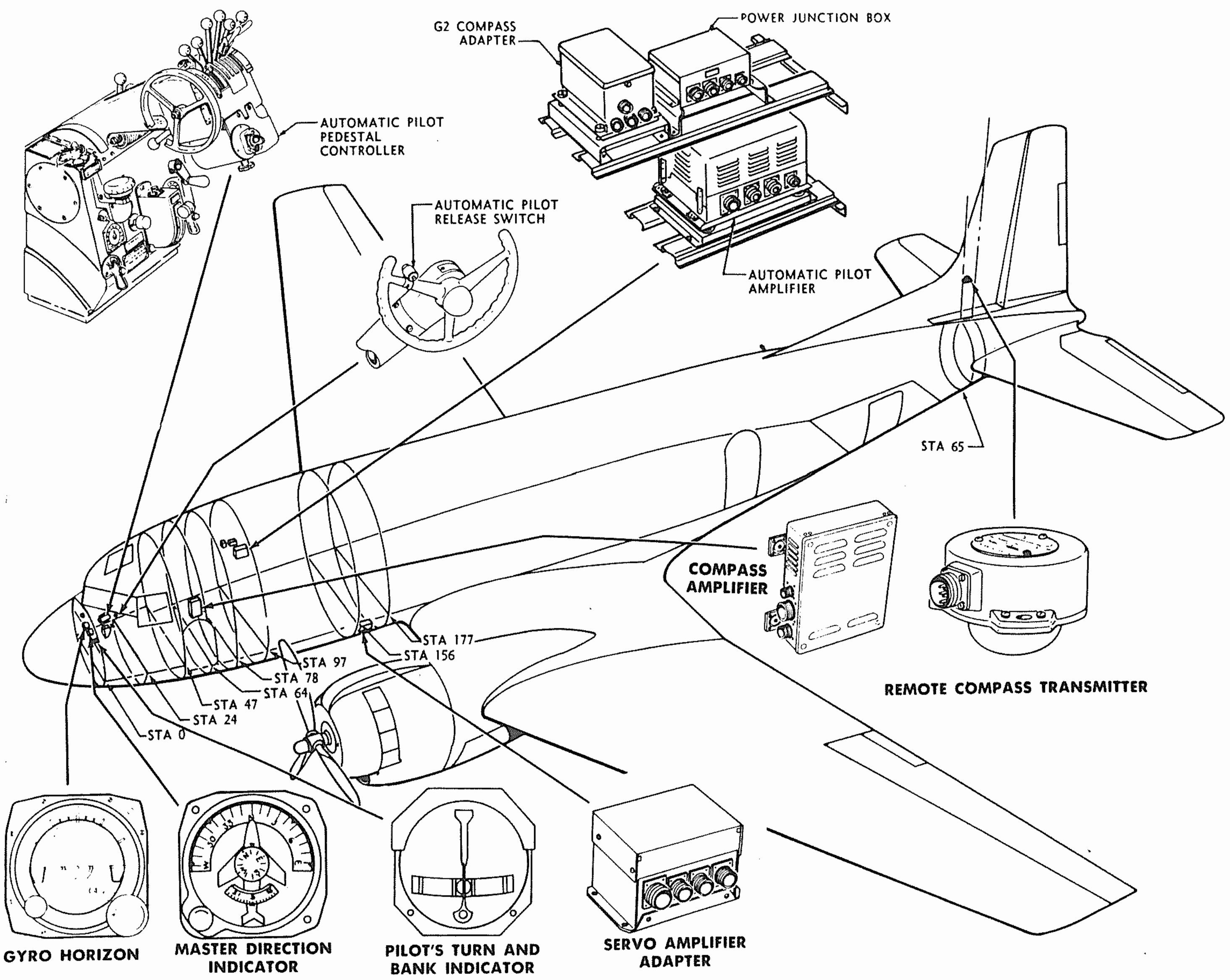


Figure 6-3. Automatic Pilot Installation

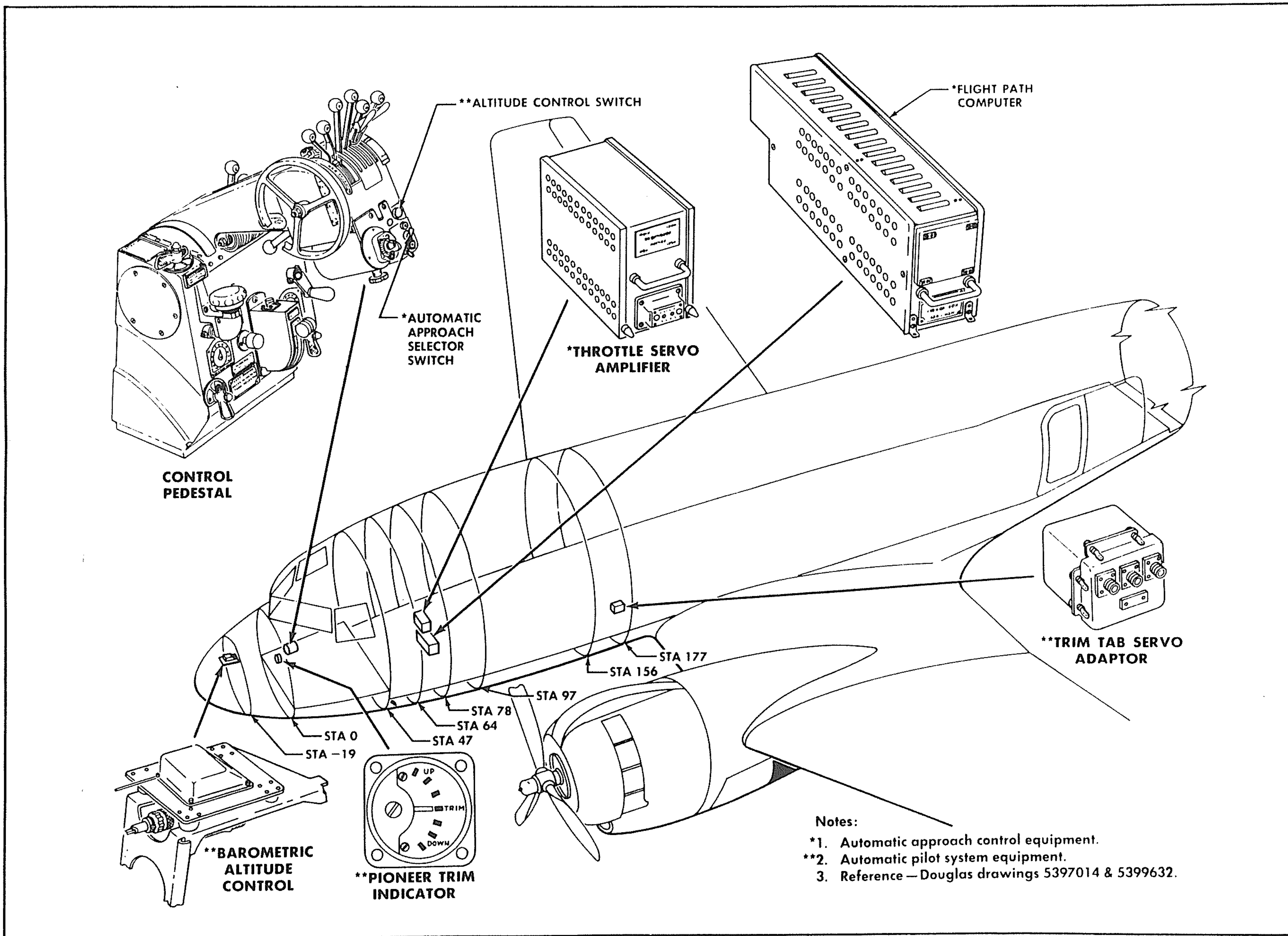


Figure 6-4. Automatic Pilot Installation (Aircraft C, D, and 59 through 96)

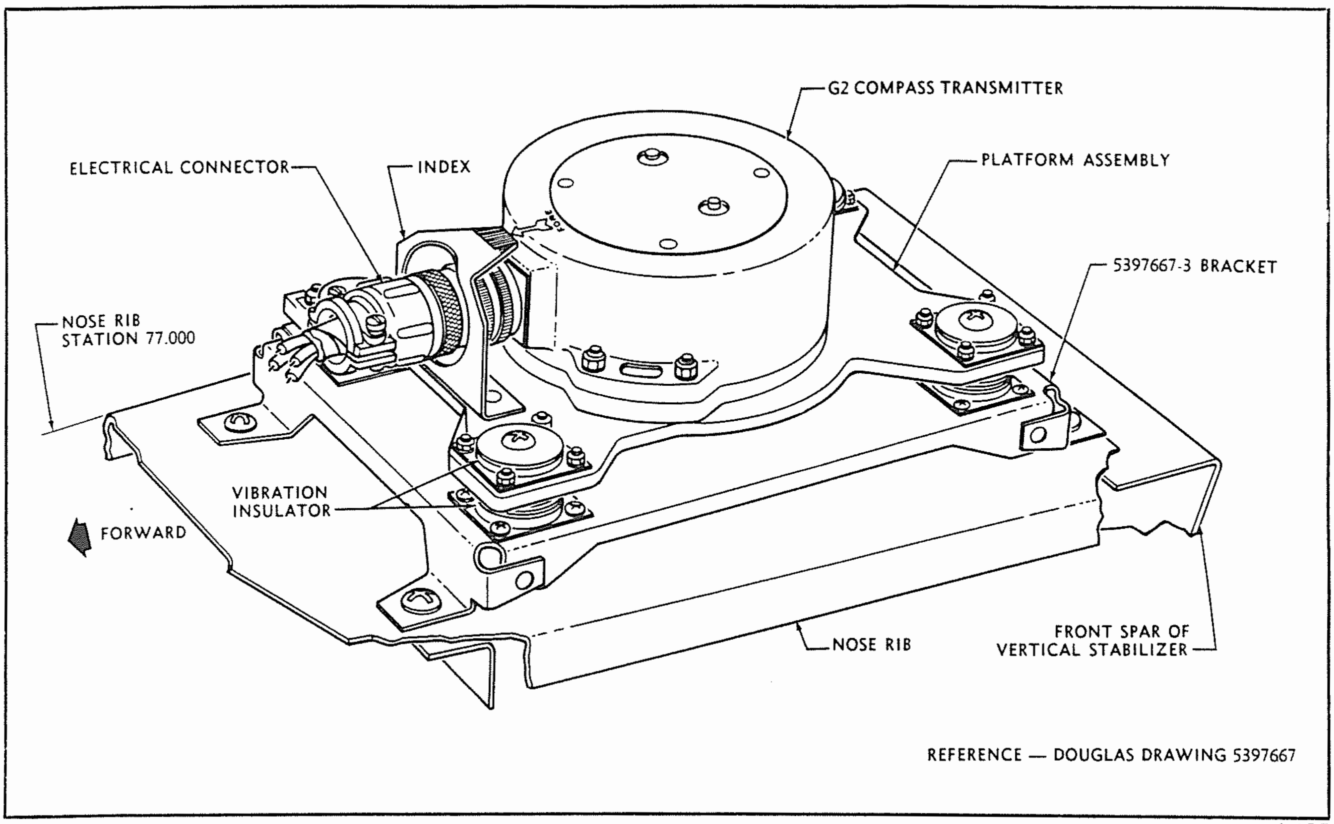


Figure 6-5. G-2 Compass Transmitter

lease; and a trim tab servo, trim tab servo pulley and friction release, located in the lower fuselage at station 156.

6-58. G-2 COMPASS TRANSMITTER. *(See figure 6-5.)* The G-2 remote compass transmitter consists of the magnetic element that aligns itself with the earth's magnetic field and a coil that electrically transmits this position to the master direction indicator, which is located on the main instrument panel. The G-2 remote compass transmitter is installed in the vertical stabilizer of the aircraft.

6-59. REMOVAL OF G-2 COMPASS TRANSMITTER.

a. Open the access door on the left side of the vertical stabilizer.

b. Disconnect the electrical lead.

c. Remove the transmitter by removing the three attaching bolts.

6-60. INSTALLATION OF G-2 COMPASS TRANSMITTER. Reverse the removal procedure and align for correct magnetic heading, if required.

6-61. MASTER DIRECTION INDICATOR. *(See figure 6-6.)* The master direction indicator is installed on the main instrument panel, directly in front of the pilot.

6-62. REMOVAL OF MASTER DIRECTION INDICATOR. For removal of the master direction indicator, see paragraph 6-16.

6-63. INSTALLATION OF MASTER DIRECTION INDICATOR. For installation of the master direction indicator, see paragraph 6-17.

6-64. COMPASS AMPLIFIER. *(See figure 6-7.)* The compass amplifier is installed on the left side of the aircraft, above the service door, between stations 47 and 64. The amplifier acts as a junction box for the other components of the system, as an amplifier and a rectifier for the detector signal from the master direction indicator, and as a 26-volt power source for the other units.

6-65. REMOVAL OF COMPASS AMPLIFIER.

a. Disconnect the electrical leads.

b. Remove the four bolts attaching the amplifier to the mount.

c. Remove the amplifier.

6-66. INSTALLATION OF COMPASS AMPLIFIER. Reverse the removal procedure.

(Continued on Page 473)

Notes:

1. Encircled numbers refer to cables listed in Cable Chart, see Sheet 4.
2. For cable assembly details, see sheets 4 and 5.
3. For Rigging and Adjustment Diagram, see Sheet 3.
4. Tension cables in accordance with adjustment procedure, see paragraph 6-70 C.
5. Tighten all nuts and bolts in accordance with torque values listed in Torque Value Chart.
6. Reference — Douglas drawings 5397005, 5398929, and 5399062.

A — Pulley Bracket — Station 10, see Sheet 2
B — Fairlead — Station 24, see Sheet 2
C — Fairlead — Station 156, see Sheet 2
D — Servo Installations, see Sheet 3

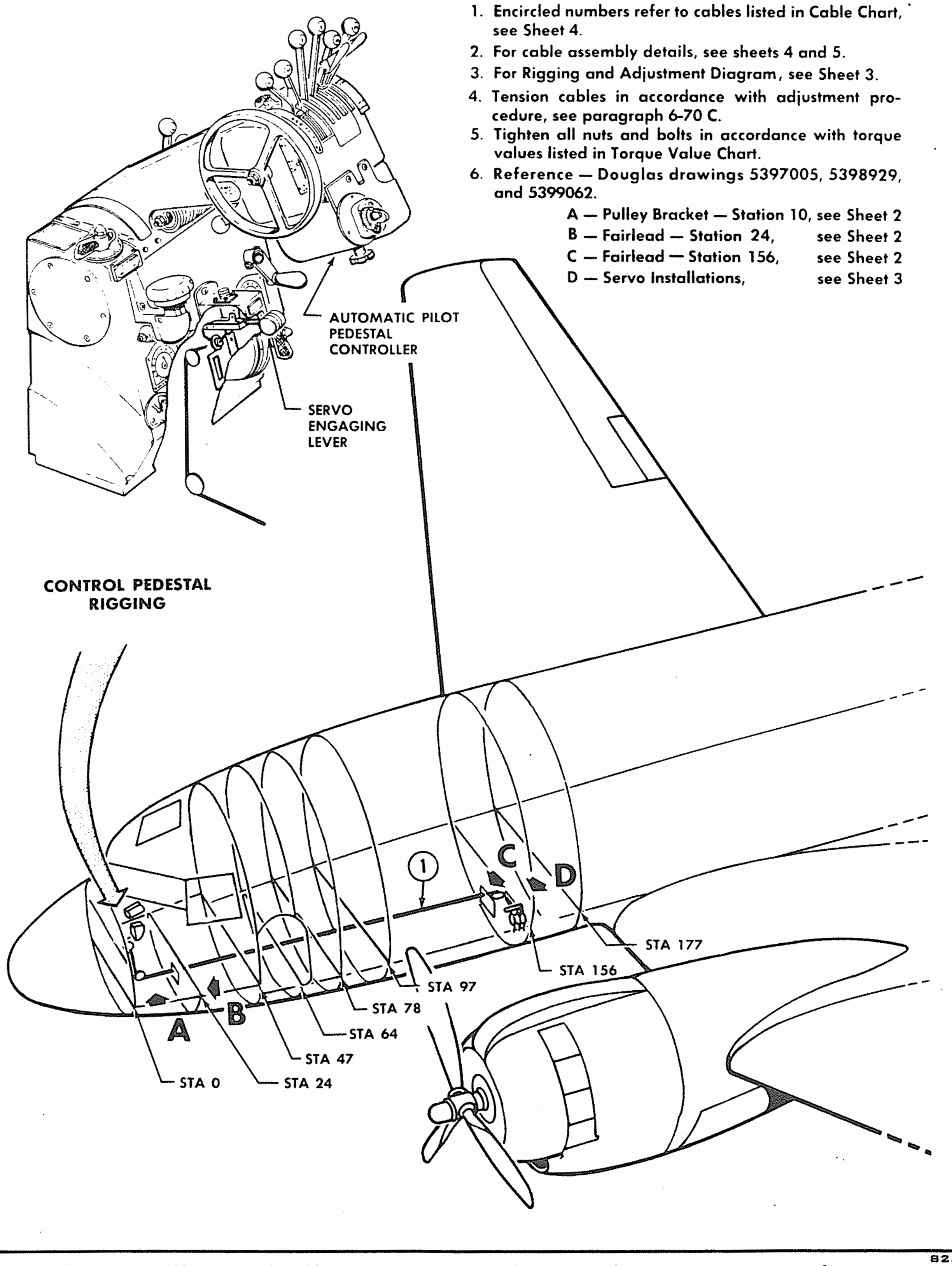
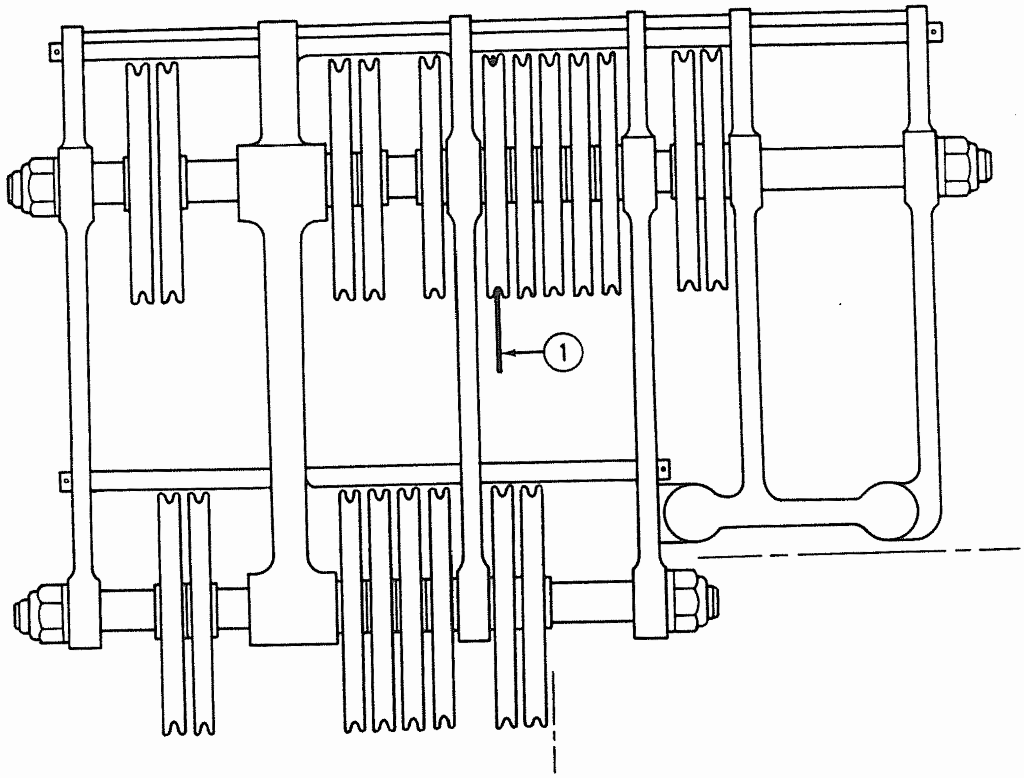
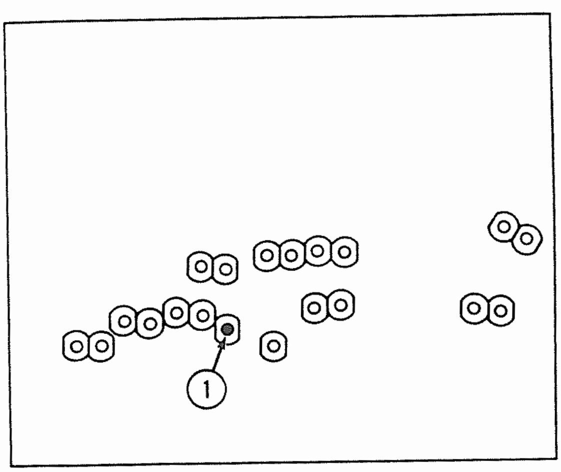


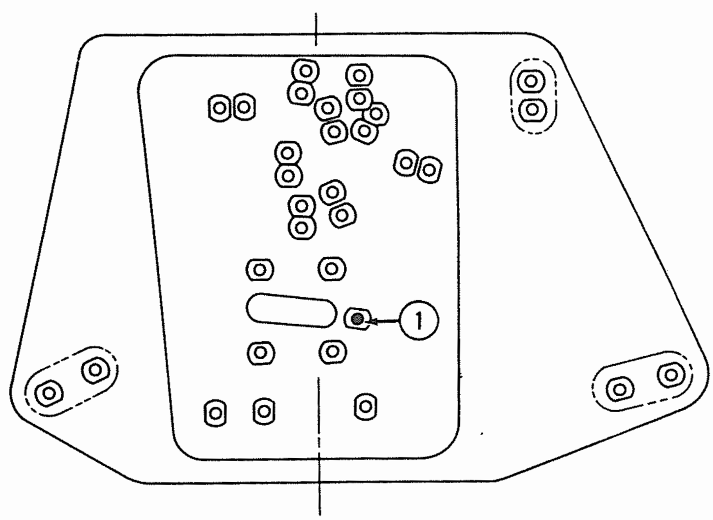
Figure 6-6 (Sheet 1 of 5 Sheets). Automatic Pilot Control System — Key Drawing



VIEW A — BOTTOM VIEW OF PULLEY BRACKET UNDER CONTROL PEDESTAL — STATION 10



VIEW B — LOOKING FORWARD AT FAIRLEAD — STATION 24



VIEW C — LOOKING FORWARD AT FAIRLEAD — STATION 156

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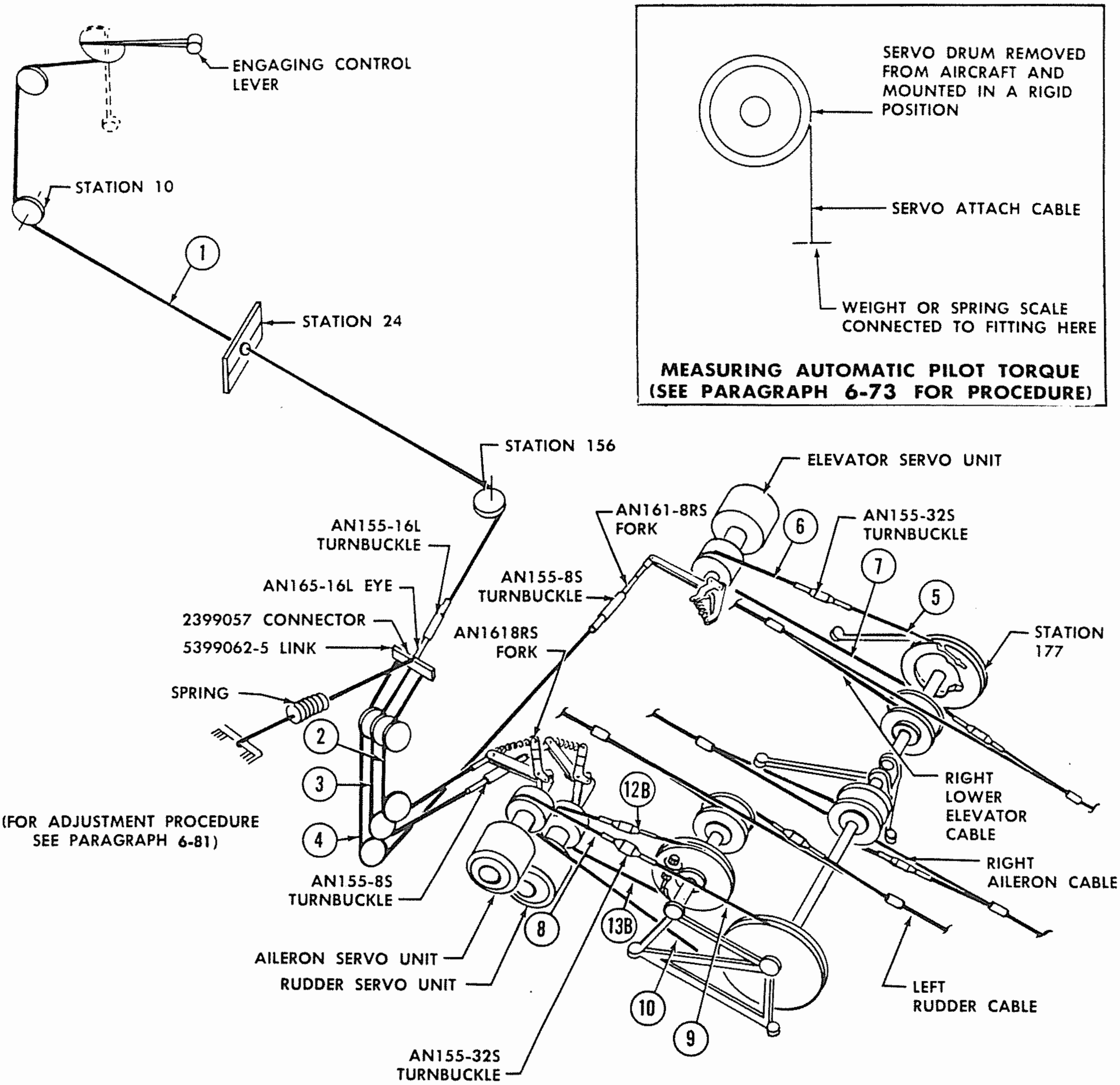
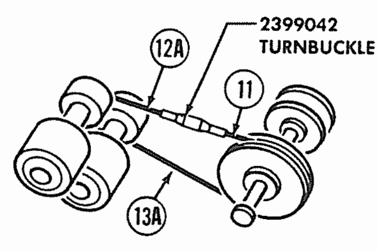
Figure 6-6 (Sheet 2 of 5 Sheets). Automatic Pilot Control System — Pulley Bracket, Station 10; Fairleads, Stations 24 and 156



VIEW D
AUTOMATIC PILOT SERVO UNITS AND CABLE HOOK-UPS



VIEW E
(AIRCRAFT A, B, 1 THROUGH 43, 45 THROUGH 59, AND 61 THROUGH 65)

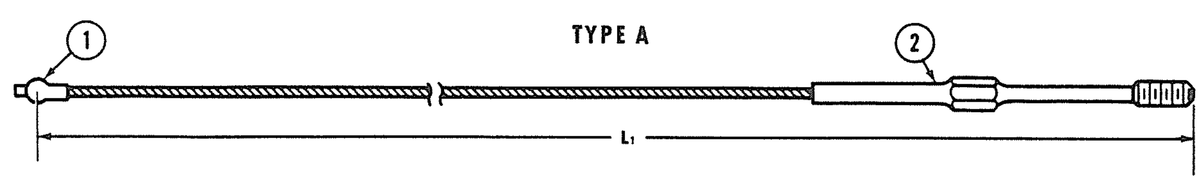
11,363

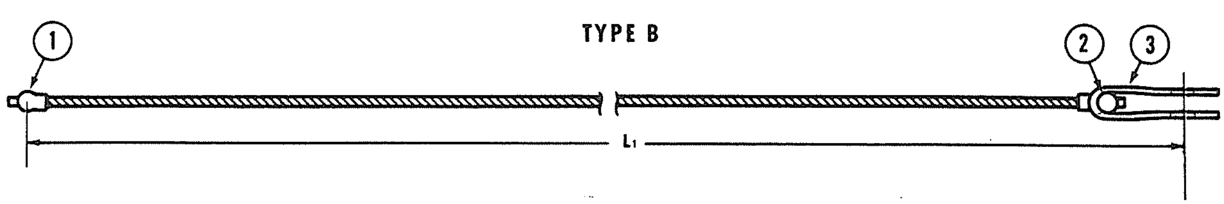
Figure 6-6 (Sheet 3 of 5 Sheets). Automatic Pilot Control System — Adjustment Diagram and Torque Measurement

AUTOMATIC PILOT CONTROL CABLE CHART

CABLE REF. NO.	DOUGLAS CABLE ASSEMBLY DRAWING NO.	NO. REQ.	TYPE	CABLE LENGTH (L_1)	CABLE SIZE	FITTINGS (1)	(2)	(3)	(4)
1	4398907-551 Engage-Disengage	1	A	145 1/2	3/32 dia 7 x 7 flex	AN668-3	AN669L3 LH		
2	4398907-549 Ail. Engaged	1	B	21 1/4	1/16 dia 7 x 7 flex	AN664-2	RA2487-2	2399061	
3	4398907-547 Elev. Engaged	1	C	30 7/8	1/16 dia 7 x 7 flex	RA2487-2	AN669S2 LH		
4	4398907-545 Rud. Engaged	1	C	22 1/2	1/16 dia 7 x 7 flex	RA2487-2	AN669S2 LH		
5	4398907-533 Elevator	1	D	18 1/2	1/8 dia 7 x 19 flex	AN664-4	AN669S4 LH		
6	4398907-535 Elevator	1	E	28	1/8 dia 7 x 19 flex	AN664-4	AN669S4 RH		
7	4398907-537 Elevator	1	F	47 1/2	1/8 dia 7 x 19 flex	AN664-4	AN664-4		
8	4398907-539 Aileron	1	E	26 3/4	1/8 dia 7 x 19 flex	AN664-4	AN669S4 RH		
9	4398907-541 Aileron	1	D	25 1/2	1/8 dia 7 x 19 flex	AN664-4	AN669S4 LH		
10	4398907-543 Aileron	1	F	54 1/2	1/8 dia 7 x 19 flex	AN664-4	AN664-4		
*11	4398907-527 Rudder	1	F	18	1/8 dia 7 x 19 flex	AN664-4	AN664-4		
*12A	4398907-529 Rudder	1	F	24 1/4	1/8 dia 7 x 19 flex	AN664-4	AN664-4		
†12B	2485463 Rudder	1	G	51 1/8	1/8 dia 7x19 flex	AN664-4	AN666-4 RH		
*13A	4398907-531 Rudder	1	F	44	1/8 dia 7 x 19 flex	AN664-4	AN664-4		
†13B	4480852-505 Rudder	1	F	45 3/4	1/8 dia 7 x 19 flex	AN664-4	AN664-4		

*Aircraft A, B, 1 through 43, 45 through 59, and 61 through 65.
†Aircraft C, D, 44, 60, 66, and subsequent.





1,477
Figure 6-6 (Sheet 4 of 5 Sheets). Automatic Pilot Control System — Cable Chart and Cable Assemblies

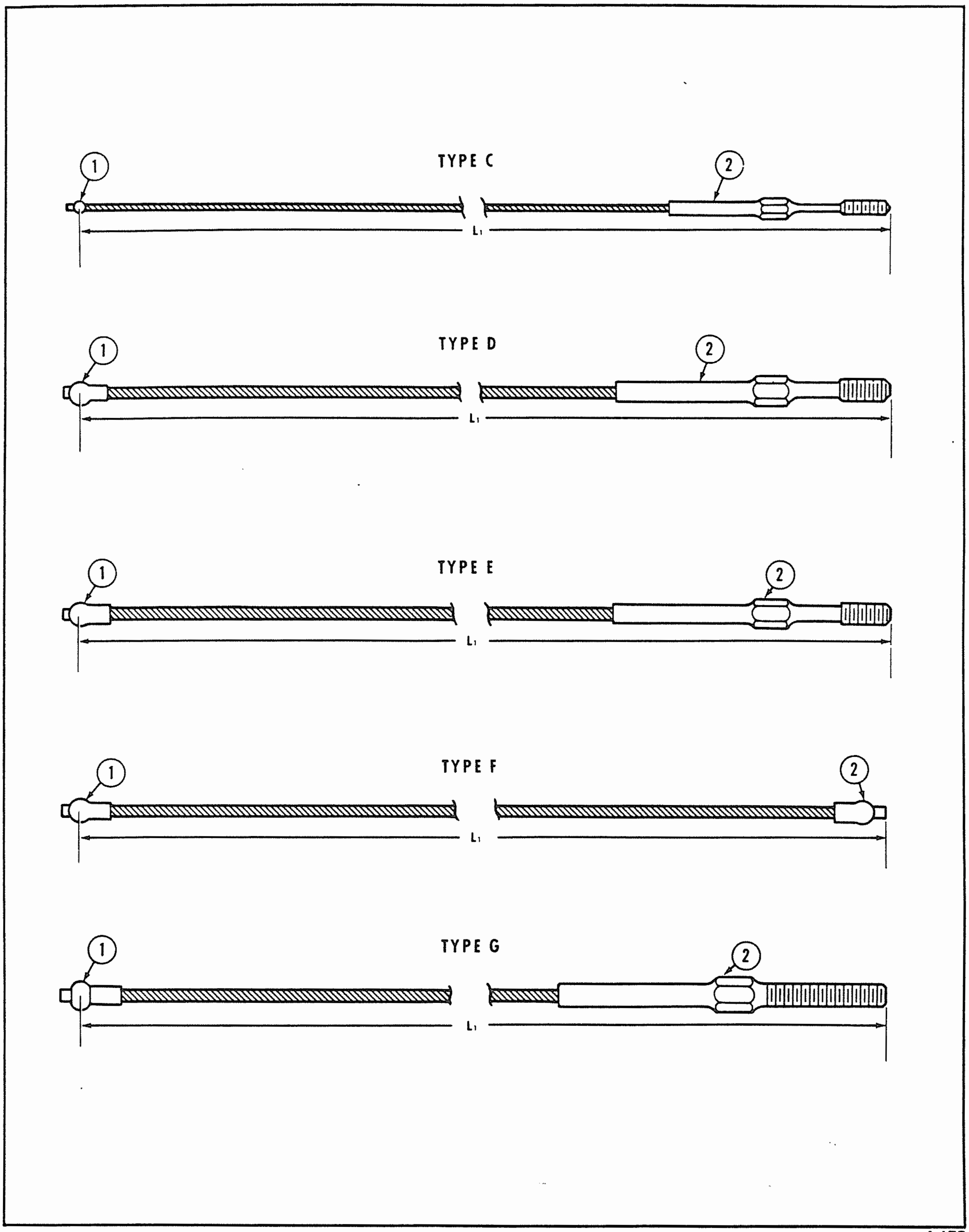


Figure 6-6 (Sheet 5 of 5 Sheets). Automatic Pilot Control System — Cable Assemblies

(Continued from Page 467)

6-67. BAROMETRIC ALTITUDE CONTROL UNIT (AIRCRAFT C, D, AND 59 THROUGH 96). *(See figure 6-8.)* The barometric altitude control unit, located in the fuselage nose section between stations −19 and 0, may be operated in conjunction with the automatic pilot system to maintain the aircraft at any desired barometric pressure altitude. The altitude control switch is mounted adjacent to the selector switch on the pedestal controller, and must be depressed manually when operation of the unit is desired. Disengagement is obtained by manual release of the switch, or through the actuation of an interlock circuit when the selector switch is turned to the APPROACH position.

6-68. REMOVAL OF BAROMETRIC ALTITUDE CONTROL UNIT (AIRCRAFT C, D, AND 59 THROUGH 96).

a. Open the access door on the nose of the aircraft.

b. Disconnect the electrical and static air connections.

c. Remove the screws and washers attaching the control unit to the supporting bracket.

d. Remove the control unit.

6-69. INSTALLATION OF BAROMETRIC ALTITUDE CONTROL UNIT (AIRCRAFT C, D, AND 59 THROUGH 96). Reverse the removal procedure.

6-70. SERVO UNITS. *(See figure 6-9.)* The aileron, elevator, and rudder servo units are installed in the lower part of the fuselage at station 15. The servo units provide the mechanical means for movement of their respective control surfaces.

6-71. REMOVAL OF SERVO UNITS.

a. Disconnect the cables between the servos and the differential pulley assembly.

b. Disconnect the automatic pilot engaging control cables from the servo units.

c. Disconnect the electrical leads.

d. Remove the attaching bolts and remove the servo units.

6-72. INSTALLATION OF SERVO UNITS.

a. Check the automatic pilot torques (see paragraph 6-73).

b. Install the servos with the attaching bolts.

c. Connect the electrical leads.

d. Attach the automatic pilot engaging control cables to the servo levers.

e. Connect the cables between the servos and the differential pulley assembly.

f. Adjust the automatic pilot engaging and servo drive controls (see paragraph 6-81).

g. Safety the turnbuckles.

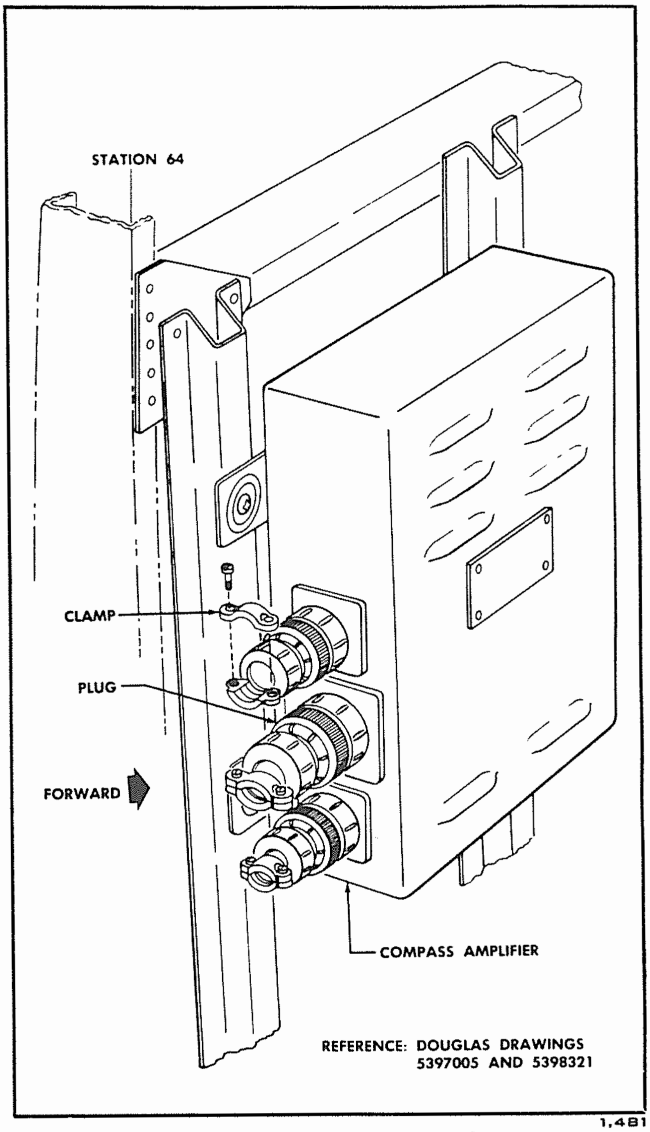


Figure 6-7. Compass Amplifier

6-73. MEASURING AUTOMATIC PILOT TORQUE. *(See figure 6-6.)*

a. Remove the servo to be checked from the aircraft.

Note

Whenever a servo is replaced, its torque must be checked before installing.

b. Mount the servo conveniently in a rigid position, with the servo cable attached and wound on the drum.

c. Connect the servo electrically to the aircraft's automatic pilot system with a jumper cable.

d. For the aileron and rudder servos, proceed as follows:

e. Suspend a 60-pound weight from the fitting on the end of the servo cable.

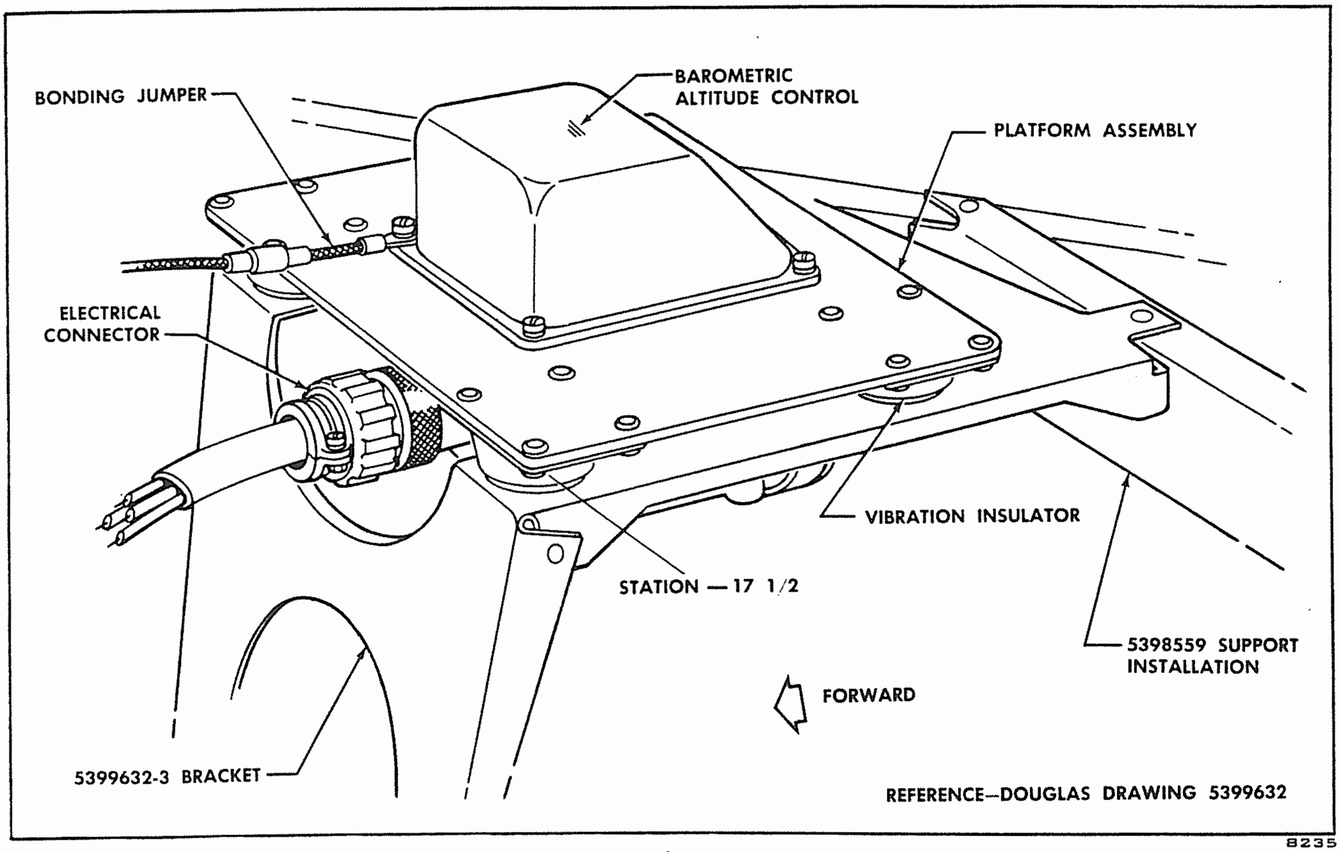


Figure 6-8. Barometric Altitude Control Unit (Aircraft C, D, and 59 through 96)

f. Apply a full automatic pilot signal to the servo by the controller in the cockpit. The servo must lift the weight. This corresponds to a 120 inch-pound minimum torque on the servo.

g. Add 30 pounds to the 60-pound weight, to make a total of 90 pounds.

h. Apply a full automatic pilot signal to the servo by the controller in the cockpit. The servo must not lift the weight. (Movement of the load a small distance, due to inertia build-up from the servo in taking up cable slack, may be ignored.) This corresponds to a 180 inch-pound maximum torque on the servo.

i. For the elevator servo, proceed as follows:

j. Suspend a 50-pound weight from the fitting on the end of the servo cable.

k. Apply a full automatic pilot signal to the servo by the controller in the cockpit. The servo must lift the weight. This corresponds to a 100 inch-pound minimum torque on the servo.

l. Add 25 pounds to the 50-pound weight, to make a total of 75 pounds.

m. Apply a full automatic pilot signal to the servo by the controller in the cockpit. The servo must not lift the weight. (Movement of the load a small distance, due to inertia build-up from the servo in taking up cable slack, may be ignored.) This corresponds to a 150 inch-pound maximum torque on the servo.

Note

The weights referred to in steps e, g, j, and l, preceding, may be replaced by a spring scale.

6-74. PEDESTAL CONTROLLER. *(See figure 6-6.)* The pedestal controller enables the pilot to put the aircraft through a variety of maneuvers without disengaging the automatic pilot system. By operating the pitch trim wheel, the aircraft can be made to climb or dive. Operation of the turn control knob automatically produces a turn which, for the rate selected, is correct in both bank and pitch. Effective on aircraft C, D, 44, 60, and 76 through 96, a bank angle limiter will limit the bank angle of the aircraft to approximately 45 degrees. Operation of the auto-pilot controller will not be effective beyond this bank angle limit. The bank angle limiter will also lock the bank trim knob in the neutral position to prevent changes in heading.

6-75. REMOVAL OF PEDESTAL CONTROLLER.

a. Disconnect the electrical plug.

b. Remove the three attaching screws and remove the controller.

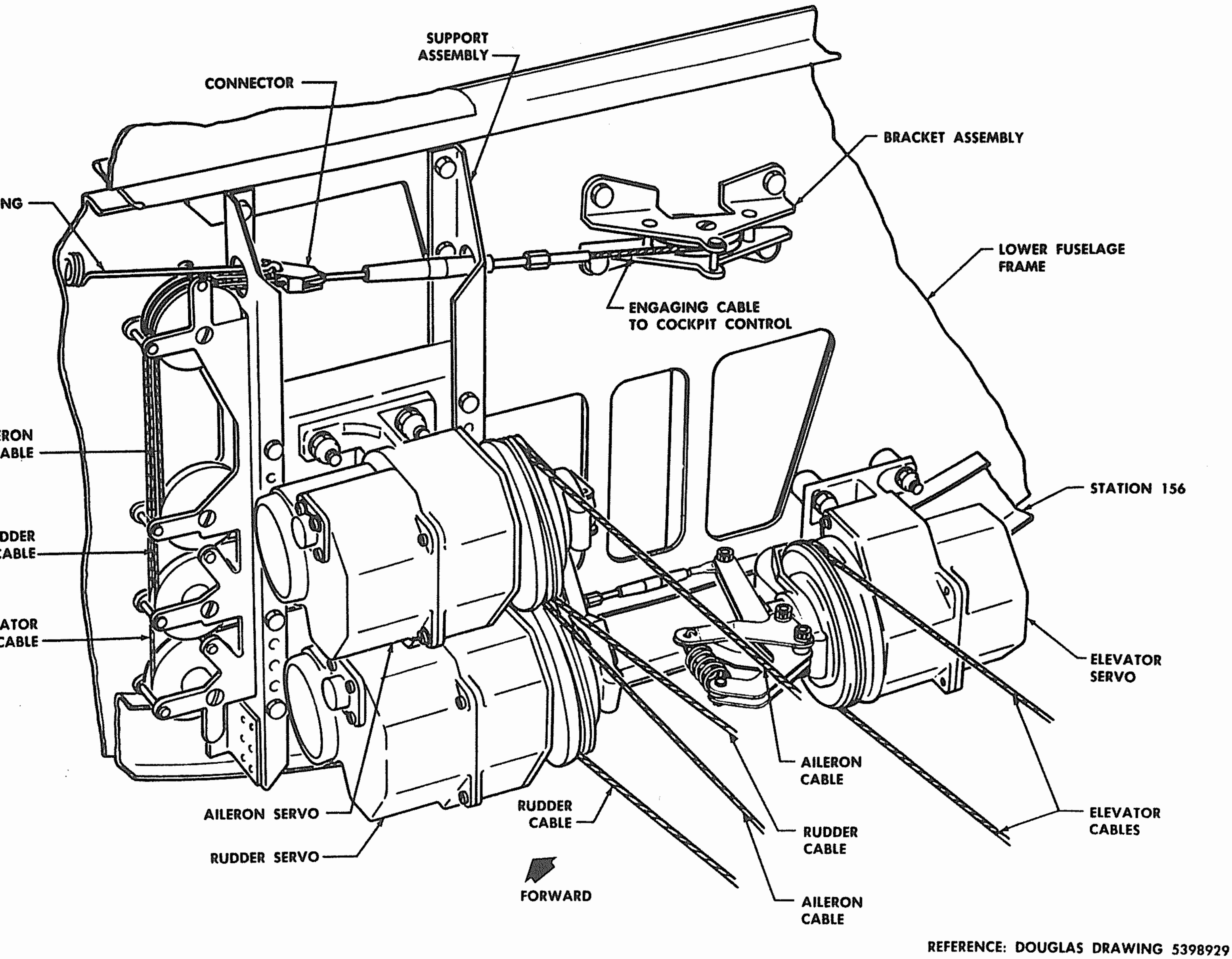


Figure 6-9. Servo Units

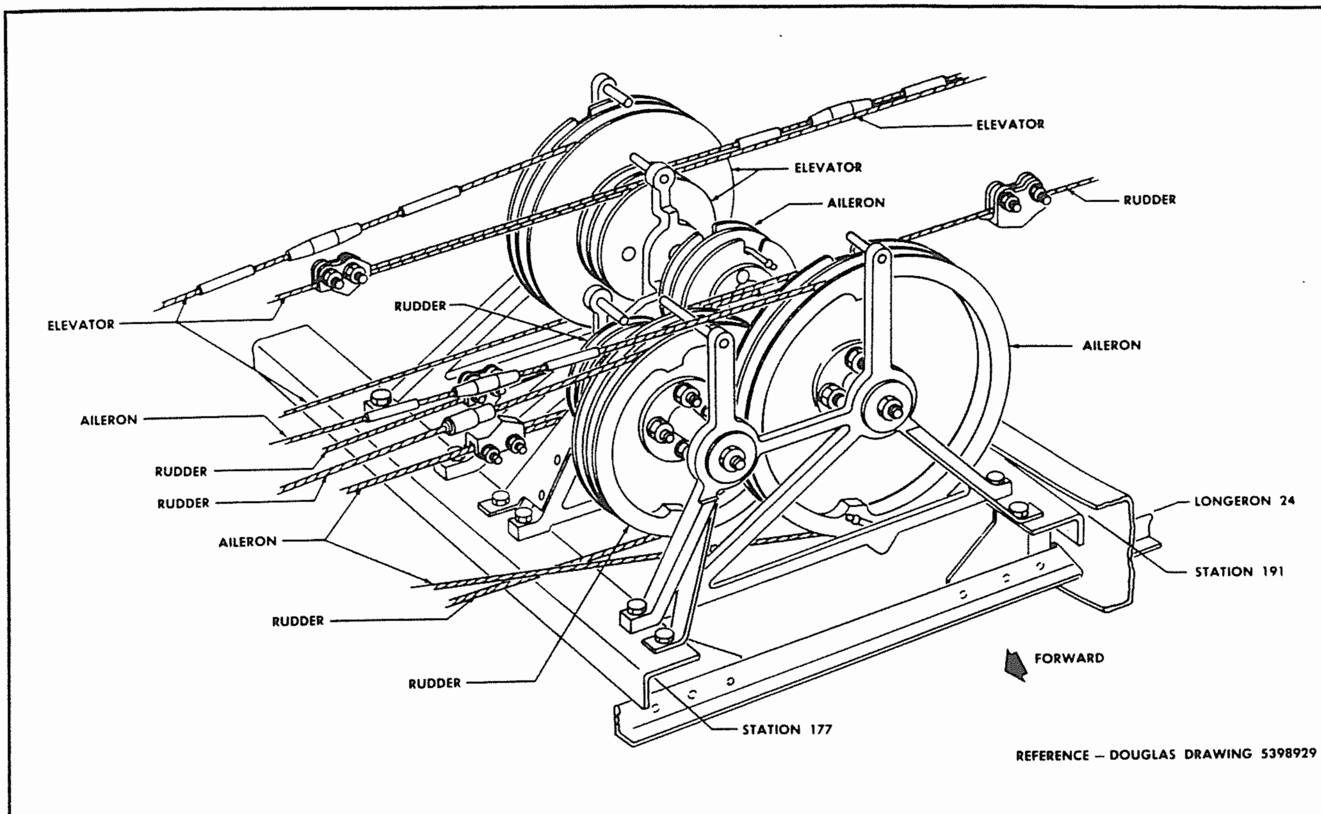


Figure 6-10. Differential Pulley Assemblies (Aircraft A, B, 1 through 43, 45 through 59, and 61 through 65)

6-76. INSTALLATION OF PEDESTAL CONTROLLER.

a. Install the controller with the three attaching screws.

b. Connect the electrical plug.

6-77. AUTOMATIC PILOT ENGAGING AND SERVO DRIVE CONTROLS. *(See figures 6-6, 6-10, and 6-11.)* The automatic pilot engaging and servo controls consist of a single, spring-return cable system extending from the engaging control lever on the control pedestal aft to station 156, where it is connected to the three cables leading to the aileron, rudder, and elevator servo lever assemblies. Raising the control lever causes the servos to engage. Servo operation is transmitted to the flight control cable systems by servo drive cables connected to the differential pulley assemblies.

6-78. AUTOMATIC PILOT ENGAGING AND SERVO DRIVE CONTROL CABLES. *(See figure 6-6.)* The automatic pilot control cables consist of a master servo engaging cable extending from the control lever on the control pedestal aft to a connector at station 156; three individual servo engaging cables that transmit the movement of the master cable to the aileron, elevator, and rudder servos; and three servo drive cables that extend between the servos and the differential pulley assemblies.

6-79. REMOVAL OF AUTOMATIC PILOT ENGAGING AND SERVO DRIVE CONTROL CABLES.

a. Remove access doors and floor panels as necessary.

b. Remove the safety wire, and disconnect the turnbuckle at station 156.

c. Disconnect the cable from the engaging lever at the control pedestal and thread the cable end.

d. Disconnect the three automatic pilot engaging cables from the connector at station 156.

e. Disconnect the three cables at the respective servo engaging levers.

f. Remove cable guard pins and fairlead grommets as necessary, and pull the cables out of the aircraft.

g. Remove the safety wire and disconnect the turnbuckle from each of the three servo drive cable assemblies.

h. Disconnect the servo drive cables from the servo pulleys, and remove the cables.

6-80. INSTALLATION OF AUTOMATIC PILOT ENGAGING AND SERVO DRIVE CONTROL CABLES.

a. Connect the servo drive cables to the servo pulleys.

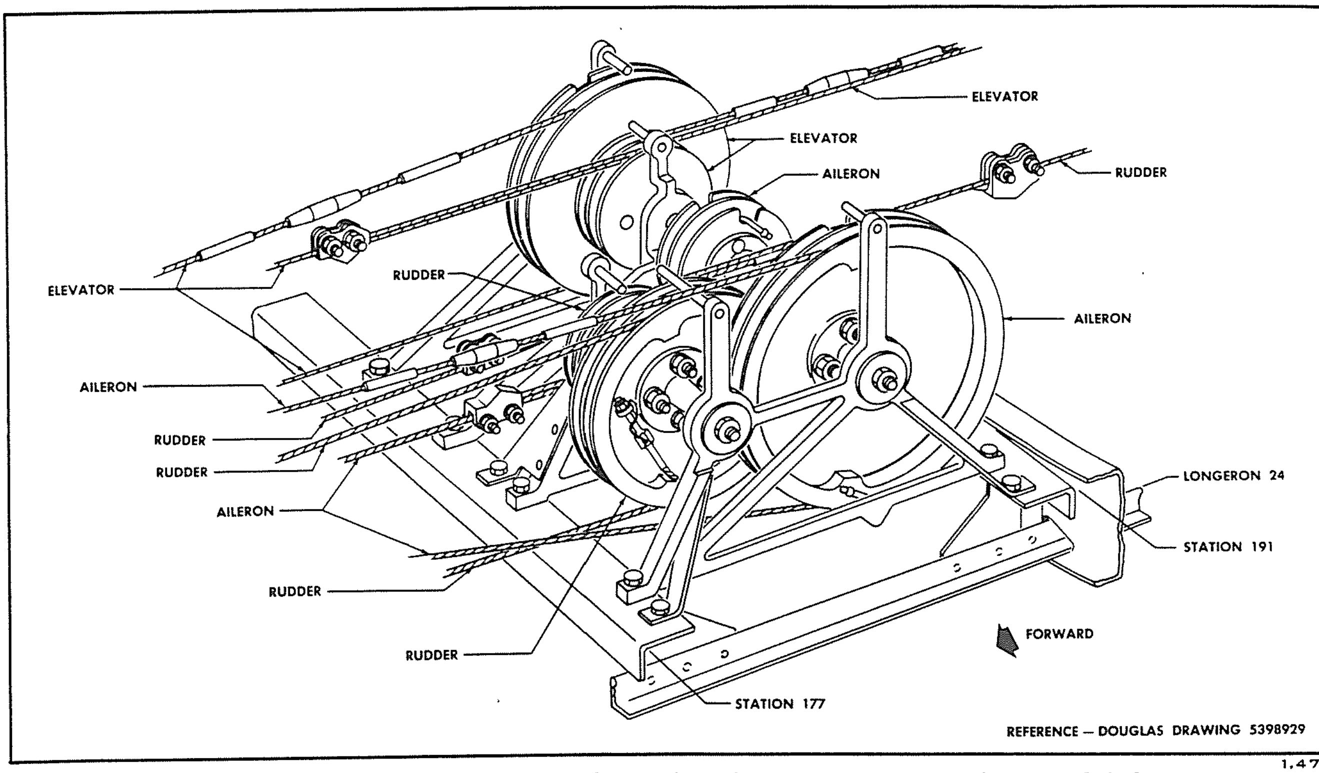


Figure 6-11. Differential Pulley Assemblies (Aircraft C, D, 44, 60, and 66 and Subsequent)

b. Attach the servo drive cables with turnbuckles.

c. Connect the three engaging cables to their respective servo engaging levers, and route them forward.

d. Connect the cables to the connector at station 156.

e. Route the forward automatic pilot engaging cable forward through the fairleads at stations 156 and 24, and the pulley bracket at station 10.

f. Connect the cable end to the engaging control lever at the control pedestal.

g. Connect the forward cable to the connector with a turnbuckle.

h. Adjust the automatic pilot engaging and servo drive controls (see paragraph 6-81).

i. Tension all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

j. Safety the turnbuckles.

k. Install the necessary guard pins and grommets.

l. Replace the access doors and floor panels in the aircraft.

6-81. ADJUSTMENT AND OPERATIONAL TESTS OF AUTOMATIC PILOT ENGAGING AND SERVO DRIVE CONTROLS.
(See figure 6-6.)

a. Lock the aileron, elevator, and rudder control surfaces in neutral.

b. Tension the aileron, rudder, and elevator servo jumper cables to one-half the rig load stated in the Cable Rigging Tension Chart, figure 2-9.

c. Adjust the cables attached to the servo drums, in accordance with the Cable Rigging Tension Chart, figure 2-9. Safety the drum cable turnbuckles.

d. Set the automatic pilot engaging control lever, located on the control pedestal, in the DISENGAGED position; adjust the turnbuckle forward of the connector so that the cable between the connector and the aileron servo is just slack.

e. Adjust the cables, between the connector and the elevator and rudder servos, with the turnbuckles at the respective engaging levers until the cables are just slack.

f. Move the automatic pilot engaging control lever, on the control pedestal, to the ENGAGED position. The engaging levers at each servo should have a travel of 1⅜ inches.

g. Safety all turnbuckles.

6-82. DIFFERENTIAL PULLEY ASSEMBLIES. *(See figures 6-10 and 6-11.)* The differential pulley assemblies are installed at station 177 and serve to reduce the cable travel transmitted from the servos to the main surface control cable.

6-83. REMOVAL OF DIFFERENTIAL PULLEY ASSEMBLIES.

a. Remove the automatic pilot servo drive cables (see paragraph 6-79).

b. Loosen the turnbuckles on the primary control cable jumpers.

c. Disconnect the jumper assemblies from the pulleys.

d. Remove the bolts and nuts attaching the pulley bracket assemblies to the aircraft structure.

e. Remove the pulley bracket assemblies.

6-84. INSTALLATION OF DIFFERENTIAL PULLEY ASSEMBLIES.

a. Install the pulley bracket assemblies on the aircraft structure.

b. Connect the jumper cables to the pulleys. Rig the cables in accordance with the instructions contained in paragraphs 2-89, 2-139, and 2-188.

c. Install the automatic pilot servo drive engaging cables (see paragraph 6-80).

d. Tension all cables in accordance with the Cable Rigging Tension Chart, figure 2-9.

e. Safety all turnbuckles.

AN 01-40NK-2

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z
AIRCRAFT

SECTION VII
ELECTRICAL SYSTEM

THIS SECTION SUPERSEDES SECTION VII OF AN 01-40NK-2
DATED 15 MAY 1952 REVISED 1 NOVEMBER 1952

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AND THE CHIEF OF THE BUREAU OF AERONAUTICS

15 April 1953

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SECTION VII

ELECTRICAL SYSTEM

TABLE OF CONTENTS

SECTION VII

ELECTRICAL SYSTEM

7-1. ELECTRICAL SYSTEM.

7-2. DESCRIPTION. The electrical system is basically a 24- to 28-volt, direct-current, single-conductor system. D-c power is obtained from 2 engine-driven, 300-ampere generators and two 12-volt storage batteries connected in series *(see figures 7-1 and 7-2)*. Two separate inverters are operated by 24- to 28-volt d-c system to provide 3-phase, 115-volt, 400-cycle alternating current. Power distribution is provided by feed cables with the aircraft structure serving as a ground return for both the d-c and a-c electrical circuits. Circuit protection is provided by fuses, current limiters, and circuit breakers. Relays, circuit breakers, fuses, limiters, resistors, and other electrical devices are installed in junction boxes throughout the aircraft. Major control switches are located on accessible control panels for the pilot and co-pilot in the flight compartment.

7-3. Most of the automatic equipment and nearly all of the flight and navigational instruments in the aircraft are operated electrically; therefore, a complete understanding of the function of all electrical installations is imperative.

7-4. TROUBLE SHOOTING OF ELECTRICAL SYSTEM. Electrical failures which are apt to occur most frequently are listed in the following trouble shooting tabulation. If the cause of an operational failure cannot be found by using the tabulation, a complete overall check must be made of the units and circuits involved. All electrical troubles arise from four basic causes: a circuit may be shorted, grounded, open, or it may have faulty units.

Note

Where specific electrical trouble shooting is related functionally with the balance of a system, for example, engine electrical, additional trouble shooting procedures may be found in the section applicable to that system.

Trouble	*Probable Cause*	*Remedy*
	GENERATOR SYSTEM	
a. Generator operating	Faulty ammeter.	Replace ammeter.
a. *(Continued)* at normal rpm, but no indication on the ammeter.	Open field-circuit.	Check and correct faulty wiring.
	Generator field circuit breaker open.	Check wiring and reset circuit breaker.
	External wiring improperly connected.	Check with appropriate wiring diagram.
	Generator field demagnetized or magnetic polarity reversed.	Flash field in the proper direction (see paragraph 7-11).
	Shunt field coil assembly open, grounded, or shorted.	Replace generator.
	Shorted, grounded, or open armature.	Replace generator.
	Ammeter circuit open, shorted, or grounded.	Check the circuit.
	Broken inner shaft or failure in vibration damper assembly.	With engine at standstill, remove generator brush inspection band. Pull engine through by hand and check rotation. If armature does not rotate smoothly while engine is being pulled through, replace generator.
	Overvoltage relay open.	Reset overvoltage relay.
b. Generator operating at normal rpm, but high voltage output is low.	Faulty or improperly adjusted voltage regulator.	Adjust regulator (see paragraph 7-15) or replace the voltage regulator.
	Loose, dirty, or high-resistance connections.	Check and tighten connections.
	Binding, worn, improperly seated, or loose-fitting brushes.	Service brushes (see paragraph 7-9).
	Dirty commutator.	Replace generator.

Trouble	Probable Cause	Remedy
c. Generator operating at normal rpm, but high voltage is indicated.	Improperly adjusted voltage regulator.	Adjust voltage regulator (see paragraph 7-15).
	Defective voltage regulator.	Replace regulator.
	Short circuit between cables A and D on generator.	Check generator leads for short circuits.
d. Excessive arcing of generator brushes.	Brushes binding in holder.	Remove brushes and clean with naphtha, Specification TT-T-291A.
	Brushes worn or weak brush springs.	Replace generator brushes.
	Dirty or coated commutator.	Replace generator.
	Short circuit between terminals A and D on the generator.	Check generator leads for short circuit.
e. Generator commutator throwing solder.	Dirty commutator.	Replace generator.
	Brushes defective.	Reseat, clean, or replace brushes.
	Open or shorted armature.	Replace generator.
BATTERIES		
f. Battery will not hold charge.	Battery is worn out.	Replace battery or batteries.
	Charge rate not properly adjusted.	Adjust the voltage regulator (see paragraph 7-15).
	Battery standing idle too long.	Battery will require removal and recharging if left in an unused aircraft for one week or more, depending on the temperature. A fully charged battery will lose approximately one-half its charge in the following number of days when temperatures are as shown: 15.6°C (60°F) 90 days 26.7°C (80°F) 45 days 37.8°C (100°F) 14 days 49°C (120°F) 6 days
	Electrical equipment left on inadvertently.	Remove and recharge the battery.
	Short circuit or ground in wiring.	Check the wiring and correct the trouble.
	Broken cell partition.	This is usually noted by two or more adjacent cells running down, particularly if left standing a few days. Replace the battery.
g. Battery life is short.	Overcharging.	Overcharging of the battery will cause buckling of the plates, shedding of active plate material, oxidation of grids, overheating, excessive loss of water, and gassing. Correct the adjustment of the voltage regulator (see paragraph 7-15).
	Level of electrolyte below top of plates.	Keep electrolyte level ⅜ of an inch above top of protector plate.
	Frequent discharging of battery.	This is due to excessive use of the starter and other electrical equipment while the aircraft is on the ground and then recharging the battery while the aircraft is in the air. External power source is to be used whenever possible to conserve the batteries.
	Sulphated plates.	Sulphating of the battery plates is caused when the battery is left in a discharged or undercharged condition (½ charge or less) for a period of time, or when electrolyte is not maintained at its proper level. Remove the battery and replace with a fully charged battery.
	Improper storage of battery.	Dry batteries stored in a damp location, or wet batteries stored for too long a period without charging will cause the plates to sulphate.
h. Cracked cell jars.	Hold-down clamps loose.	Replace the battery and secure the hold-down clamps.
	Frozen battery (see step l, following).	Remove and replace battery with fully charged battery (see step l).
i. Compound on top of battery melts.	Charging rate too high.	Replace the battery and adjust the voltage regulator (see paragraph 7-15).
	Electrolyte on top of cells, caused by overfilling or improper operation of the battery compartment ventilat-	Check the battery compartment ventilating pipes and clear the pipes if necessary. Remove any electrolyte from the top of the battery and

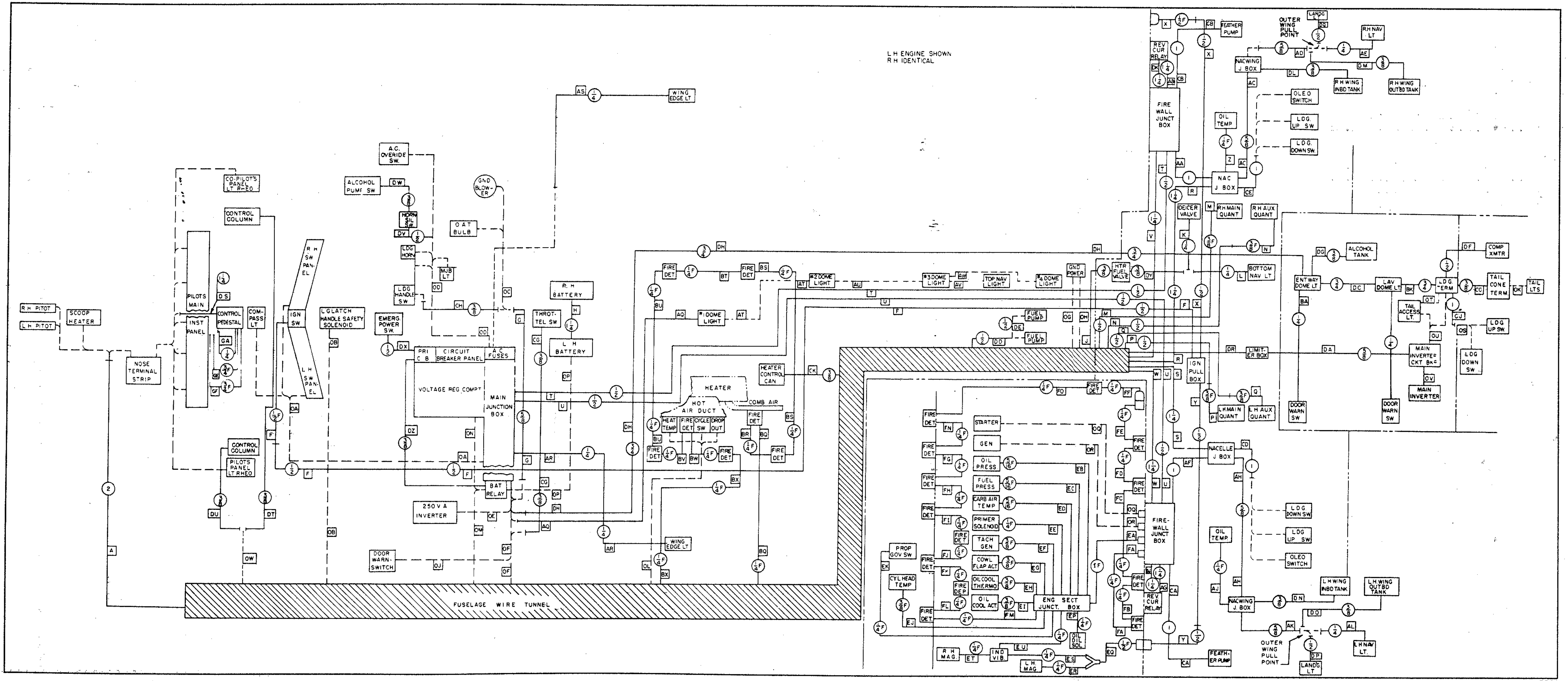

Figure 7-1. Electrical Conduit Block Diagram

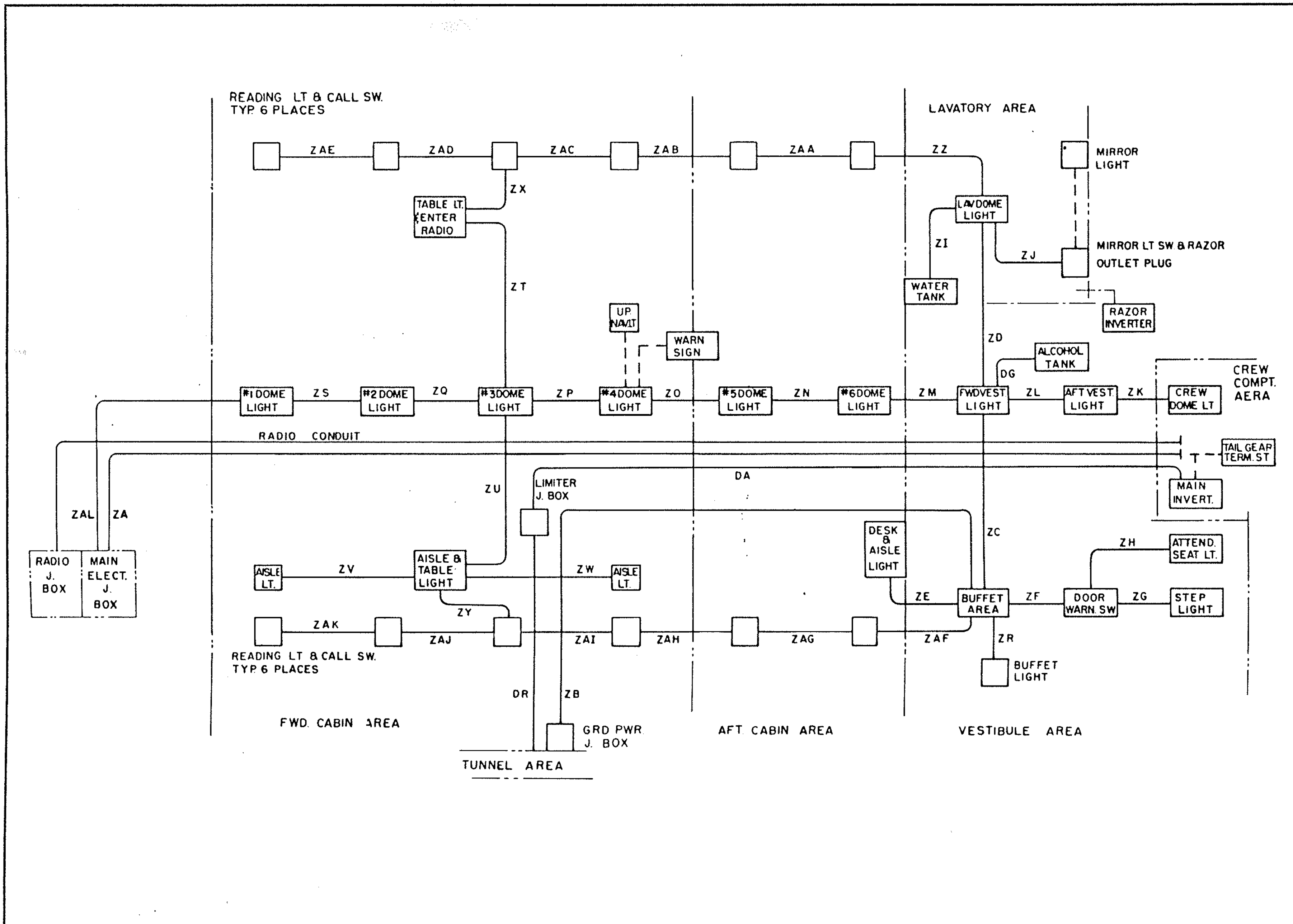

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Figure 7-2. Electrical Conduit Block Diagram (Aircraft A through D)

Trouble	*Probable Cause*	*Remedy*
i. *(Continued)*	ing system, may short-circuit the battery. The resulting heat will soften the battery compound.	neutralize with sodium bicarbonate. After neutralizing, wash off the top of the battery with clear water and recharge the battery.
j. Electrolyte runs out of vent caps.	Excessive water added to battery.	Remove excess water with self-leveling syringe.
	Excessive charging rate.	Adjust the voltage regulator.
k. Excessive corrosion inside container.	Battery overcharged.	Adjust the voltage regulator.
	Spillage, caused either by overfilling the battery or, in some remote cases, by cracked or melted sealing compound.	Remove excessive electrolyte. Reseal if necessary.
	High charging rate.	Adjust the voltage regulator.
l. Battery freezes.	Battery discharged (see the following table).	Remove and replace with fully charged battery.
	Water added to battery and battery not charged immediately.	*Water never should be added to the batteries in freezing weather when the batteries are to be left standing for any period before charging.* Approximately ½-hour charge will mix the added water with the electrolyte. The freezing point will then be in accordance with the table following. If the battery is allowed to become discharged in cold weather, freezing will result, due to the low specific gravity of the electrolyte.
	Extreme cold.	In cases of extreme freezing, it is usually necessary to replace the battery. In cases of only partial freezing, thawing out the battery in a warm room may save the battery. Check the battery for specific gravity before replacing it in the aircraft.

Trouble	*Probable Cause*	*Remedy*
l. *(Continued)*		

Specific Gravity	*Freezing Point*
1.300	−71.0°C (−95°F)
1.275	−62.0°C (−80°F)
1.250	−52.0°C (−62°F)
1.225	−37.2°C (−35°F)
1.200	−26.6°C (−16°F)
1.175	−20.0°C (− 4°F)
1.150	−15.0°C (5°F)
1.125	−10.6°C (13°F)
1.100	− 7.2°C (19°F)

Trouble	*Probable Cause*	*Remedy*
m. Battery polarity reversed.	Battery connections reversed at charger.	Slowly discharge the battery completely; then recharge the battery correctly.
	Battery connections reversed on the aircraft.	Slowly discharge the battery completely; then recharge the battery correctly.
n. Battery consumes excessive water.	Charging rate too high.	Adjust voltage regulator (see paragraph 7-15).
	Electrolyte runs out of vent caps.	Level of electrolyte too high. Adjust level to ⅜ of an inch above protector plate.
o. Battery will not take a charge.	Battery worn out.	Replace with fully charged battery.
	Battery badly sulphated, caused by standing idle in discharge condition.	Replace with fully charged battery.
INVERTERS		
p. With either inverter switch ON the inverter fails to operate.	Open or grounded power lead.	Check voltage at inverters.
	Faulty starting relay.	Remove and replace relay.
	Open switch lead.	Check wiring and switch.
	Internal fault in inverter.	Replace inverter.
q. No frequency regulation.	Faulty rectifier in control panel.	Remove and replace inverter.
	Faulty wiring.	Check wiring.
	Faulty carbon pile.	Remove inverter for replacement of carbon pile.
r. Low inverter voltage and high input current.	Short in inverter output leads.	Check external wiring.
	Short in inverter stator.	Remove and replace inverter.
s. Worn, scored, or pitted commutator.	Open or short circuit.	Remove and replace inverter.
	Faulty brushes or brush tension.	Remove and replace inverter.

Trouble	Probable Cause	Remedy
t. Improper voltage regulation.	Faulty voltage regulator.	Remove and replace inverter.
	Faulty rectifier in control panel.	Remove and replace inverter.

MAGNETOS

u. For trouble shooting of magnetos, see paragraph 5-156.

STARTER INDUCTION VIBRATOR

v. For trouble shqoting of the starter induction vibrators, see paragraph 5-156.

FUEL SYSTEM PRIMING CIRCUIT

w. For trouble shooting of the primary circuit, see paragraph 5-28.

FUEL BOOSTER PUMPS

x. For trouble shooting of the fuel booster pumps, see paragraph 5-28.

COWL FLAP ACTUATOR CIRCUIT

y. Cowl flaps fail to move when selector switch is placed in either the OPEN or CLOSE positions.	Circuit breaker open.	Check circuit breaker on circuit breaker panel.
	Faulty selector switch.	Remove and replace selector switch.
	Faulty external wiring.	Check and correct wiring.
	Faulty actuator.	Remove and replace actuator.
z. Cowl flaps fail to position correctiy.	Faulty external wiring.	Check wiring and limit switches.
aa. Cowl flaps fail to operate in POSITIONING.	Faulty selector switch.	Remove and replace selector switch.
	Faulty external wiring.	Check and correct wiring.
	Faulty actuator.	Replace actuator.

OIL DILUTION CIRCUIT

ab. For trouble shooting of the oil dilution circuit, see paragraph 5-108.

OIL COOLER DOOR ACTUATOR CIRCUIT

Trouble	Probable Cause	Remedy
ac. Oil cooler door actuator motor will not operate with selector switch in AUTOMATIC.	Circuit breaker open.	Reset circuit breaker.
	Faulty oil cooler door thermostat assembly.	Remove and replace thermostat.
	Short, open, or ground in AUTOMATIC circuit wiring.	Check and correct wiring.
	Faulty selector switch.	Remove and replace selector switch.
	Faulty actuator motor.	Remove and replace actuator assembly.
ad. Oil cooler door actuator motor will not operate with selector switch placed in either the OPEN or CLOSE position.	Faulty extend or retract limit switch.	Remove and replace actuator.
	Short, open, or ground in MANUAL circuit wiring.	Check and correct wiring.
	Faulty selector switch.	Remove and replace selector switch.
	Faulty actuator motor.	Remove and replace actuator assembly.

PROPELLER ELECTRICAL SYSTEM

ae. Propeller fails to feather.	Circuit breaker open.	Reset circuit breaker.
	Batteries low.	Recharge batteries.
	Faulty feathering pump.	Check feathering pump and if necessary replace with serviceable unit.
	Propeller feathering switch will not remain engaged.	Check wiring or low-pressure setting of cut-out switch. Check propeller feathering switch and replace if faulty.
	Propeller fails to remain feathered. Push-button fails to disengage.	Reset cut-out switch.
	Sheared coupling in feathering pump.	Replace coupling.
	Restricted oil supply to propeller feathering pump.	Check feathering pump inlet pipe for foreign material; bleed pipe.
	Faulty propeller feathering relay.	Replace relay.
	Excessive leakage, due to:	
	Improper distributor valve installation.	Check for tightness or interference.
	Faulty or incorrect seals between distributor valve and propeller shaft and between the governor base and engine mounting pad.	Install new seals.

FUEL AND OIL PRESSURE INDICATING SYSTEMS

af. For trouble shooting of the fuel and oil pressure indicating systems, see paragraph 6-7, step p.

Trouble	*Probable Cause*	*Remedy*

BULB-TYPE TEMPERATURE INDICATING SYSTEMS

ag. For trouble shooting of the bulb-type temperature indicating systems, see paragraph 6-7, step t.

THERMOCOUPLE-TYPE TEMPERATURE INDICATING SYSTEMS

ah. For trouble shooting of the thermocouple-type indicating systems, see paragraph 6-7, step x.

FUEL QUANTITY INDICATING SYSTEM (LIQUIDOMETERS)

ai. For trouble shooting of the Liquidometer-type fuel quantity indicating system, see paragraph 6-7, step z.

TACHOMETER SYSTEM

aj. For trouble shooting of the tachometer indicating system, see paragraph 6-7, step r.

CARBURETOR AND PROPELLER ANTI-ICING

Trouble	*Probable Cause*	*Remedy*
ak. Alcohol anti-icing pump fails to operate.	Circuit breaker OPEN.	Close circuit breaker. Replace if faulty.
	Open, ground, or short in wiring.	Check wiring circuit.
	Faulty anti-icing pump.	Remove and replace pump assembly.
	Faulty anti-icing switch.	Remove and replace switch.

CABIN HEATER AIRSCOOP AND PITOT ANTI-ICING CIRCUIT

Trouble	*Probable Cause*	*Remedy*
al. Anti-icing elements fail to function.	Circuit-breaker-to-heater element OPEN.	Reset circuit breaker.
	Open, ground, or short in wiring.	Check and correct wiring.
	Faulty control switch.	Replace switch.
	Faulty cabin heater airscoop relay.	Replace relay.
	Faulty heating element.	Replace heating element.
am. Ammeter fails to indicate amperage drain of elements.	Faulty ammeter.	Replace ammeter.
	Open, ground, or short in wiring.	Check and correct wiring.
	Faulty ammeter selector switch.	Replace selector switch.
	Faulty shunt.	Replace shunt.
	Blown fuse in ammeter circuit.	Correct circuit fault and replace fuse.
	Open circuit in heating element.	Replace heating unit.

SURFACE DE-ICER DISTRIBUTION PUMP CIRCUIT

Trouble	*Probable Cause*	*Remedy*
an. Surface de-icer fails to operate.	Circuit breaker OPEN.	Reset circuit breaker.
	Surface de-icer valve motor fails to operate, due to:	
	Faulty switch in distribution valve motor assembly.	Replace faulty switch.
	Faulty distribution valve motor.	Replace faulty distribution valve and motor assembly.

FUEL AND OIL PRESSURE WARNING CIRCUIT

Trouble	*Probable Cause*	*Remedy*
ao. Fuel or oil pressure warning light does not operate or gives false warning.	Lamp burned out.	Replace lamp with one that has been tested. (Lamp may be tested by means of the push-to-test button.)
	Open or faulty wiring.	Check and correct wiring.
	Faulty switch.	Replace switch.
	Loose pressure fitting.	Check all pressure fittings and tighten as necessary.
	Leak in connecting tube.	Replace tube.

FIRE WARNING CIRCUIT

Trouble	*Probable Cause*	*Remedy*
ap. One or both lamps fail to illuminate when test switch is operated.	Lamps burned out.	Replace lamps with lamps that have been tested.
	Open, shorted, or grounded circuit.	Check and correct wiring.
	Faulty sensitive relays or heater element in control box.	Replace relay or heater element.
	Defective thermoswitch.	Replace thermoswitch.

POSITION LIGHT CIRCUIT

Trouble	*Probable Cause*	*Remedy*
aq. Position light flasher fails to operate.	Circuit breaker OPEN.	Reset circuit breaker.
	Open or faulty wiring.	Check and correct wiring.
	Motor contactors worn.	Replace flasher assembly.
	Defective motor.	Replace flasher assembly.
	Faulty position switch on electrical panel.	Replace switch.

Trouble	Probable Cause	Remedy
ar. White position light circuit ON continuously while flasher motor is operating.	Incorrect micro switch adjustment.	Move micro switch out from gear box cover plate.
as. White position light fails to function each time flasher motor is stopped.	Incorrect micro switch adjustment.	Move micro switch in toward gear box cover plate.
at. Position lights flicker.	Burned flashing switch contacts in flasher.	Replace contacts.
	Loose connections in lamp circuits.	Replace, tighten, or resolder parts responsible for loose connetcions.
au. Red and white tail position lights flash unequal lengths.	Incorrect contact adjustment of flasher switches.	Readjust screw contacts.
av. Flashing cycles too long.	Brushes defective in flasher motor.	Replace brushes.
	Faulty flasher motor.	Replace flasher assembly.
aw. Radio interference caused by position light flasher.	Poor electric bond between position light flasher and metal structure of the aircraft.	Clean contacting surfaces between position light flasher and mounting surface.
	Faulty condensor or choke.	Replace faulty condensor or choke.
LANDING LIGHT CIRCUIT		
ax. Landing lamp does not illuminate.	Circuit breaker OPEN (two circuit breakers per lamp —landing lamp position control and lamp relay circuit breaker, and landing light circuit breaker).	Reset circuit breaker.
	Burned out lamp.	Replace lamp (see paragraph 7-119).
	Loose or open connection in circuit.	Check circuit and repair wiring.
	Faulty relay or switch.	Replace faulty relay or switch.
ay. Landing light inner canopy jams into outer canopy or does not retract to flush position.	Incorrect landing light limit switch adjustment.	Adjust landing light (see paragraph 7-118).
HEATING AND VENTILATING ELECTRICAL SYSTEM		
az. For trouble shooting on the heating and ventilating electrical system, see paragraph 4-4.		

7-5. ELECTRICAL SERVICE AND MAINTENANCE PRECAUTIONS. Experienced aircraft maintenance personnel are often reluctant to attempt major repair without specific electrical experience, because they are unfamiliar with the possibilities of damage to the electrical system in case of error and because of personal danger due to shocks or burns. There is little, if any, danger involved in aircraft electrical service. However, observe the following major precautions:

a. Never work on "hot" electrical circuits while wearing any metal articles, such as rings, watches, or metal wrist bands. The 24- to 28-volt basic electrical system does not create sufficient pressure to cause shock. Therefore, work on "hot" circuits can be done safely if no metal equipment is allowed to form a direct short or ground. The low 24- to 28-volt pressure will not force amperage through bodily resistance, but if the electrician wears a metal ring, which might become shorted between a "hot" lead and ground, the battery system (if ON) would be capable of delivering high amperage through the low resistance of the metal band. This could turn the metal white hot and cause a serious and painful burn. In the same manner, of course, allowing a metal tool, such as pliers or a screw driver, to bridge between a "hot" lead and ground will generate sufficient heat to result in possible equipment damage.

b. Shocks may be received from high-tension devices, such as the magneto or induction vibrator, or inductive surges from solenoids or relays. Such shocks, while unpleasant, are not actually injurious.

c. Caution must be observed, however, when working with the higher generated voltages, such as those from the 115-volt tap of the a-c inverters and those within the cases of the radio and other electronic equipment.

d. Do not touch any radio antenna if a set is transmitting. Considerable power passes through a radio antenna during transmission and unpleasant shocks and serious burns can result.

7-6. DIRECT-CURRENT POWER SUPPLY. *(See figure 7-4.)*

7-7. GENERATORS. *(See figure 7-3.)* The generator system is the source of power for operating the electrical equipment on the aircraft and for charging the two batteries. The two generators are connected in parallel and are driven directly by the aircraft engines. Each generator circuit includes a voltage regulator, a reverse-

1. Ammeter
2. Ammeter Shunt
3. Generator (300-Amp)
4. Reverse Current Relay
5. Field Circuit Breaker
6. Generator Switch
7. Overvoltage Protector Relay
8. Voltage Regulator
9. Generator Auxiliary Relay
10. Cabin Heater Interlock
11. Dimming Switch
12. Generator Cut Out Warning Lights
13. Generator Warning Light Test Circuit Breaker (5 Amp)

ITEM 4
REVERSE CURRENT RELAY

All current from the generator passes through the main contacts of this relay. This relay will open when reverse current of 5 to 30 Amps minimum flows from the master bus to the generator. This prevents draining of the entire system as a result of a faulty generator or voltage regulator.

ITEM 5
GENERATOR FIELD CIRCUIT BREAKER

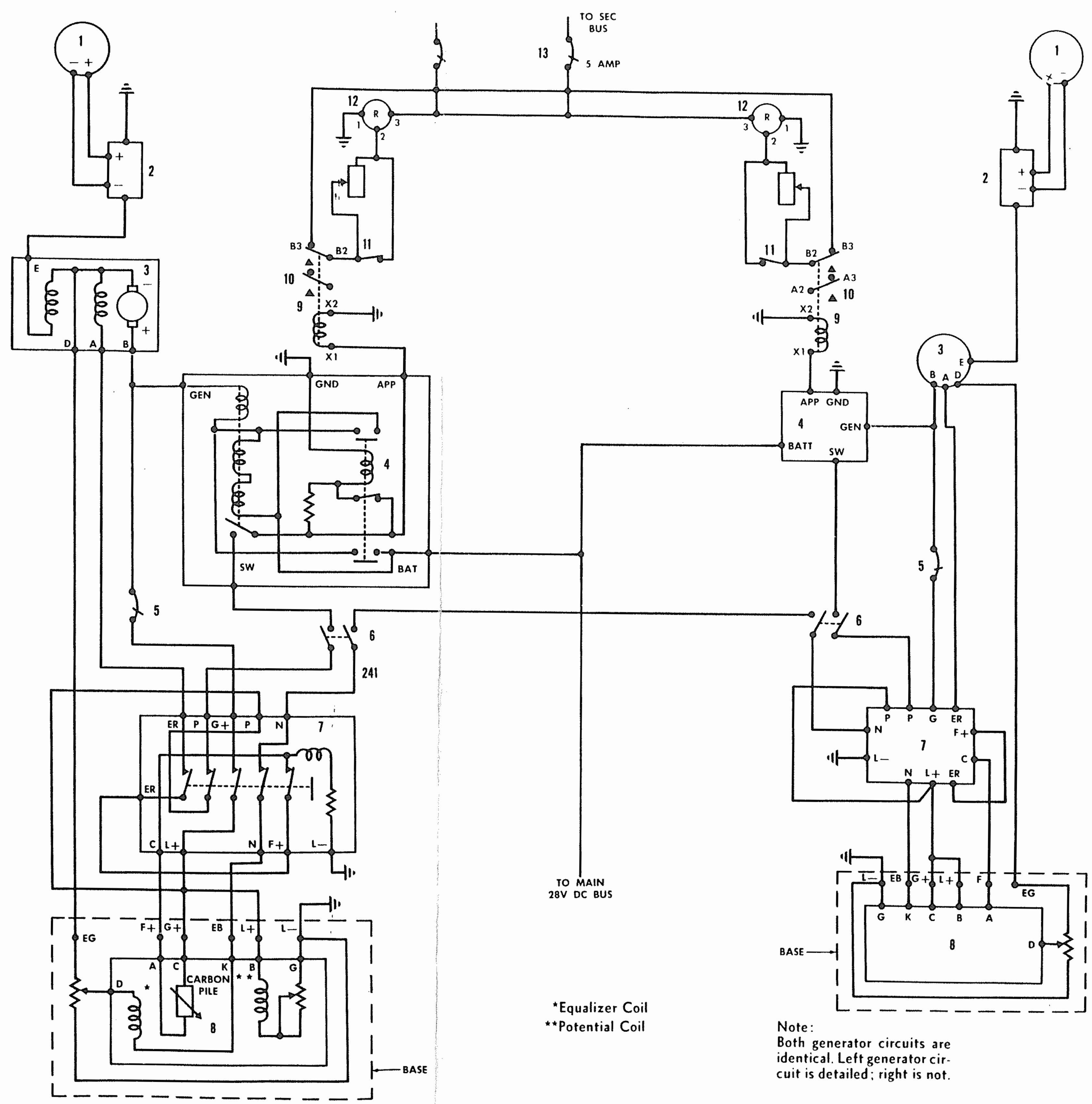
This is a trip-free circuit breaker. It should not be reclosed until the condition which caused it to trip has been corrected. Cause for tripping this circuit breaker is usually serious. It should never be reclosed except in case of emergency.

ITEM 7
OVERVOLTAGE PROTECTOR RELAYS

When the output voltage of the generator becomes abnormally high, this relay will open and disconnect the generator from the bus.

ITEM 8
VOLTAGE REGULATOR

The voltage regulator maintains a predetermined voltage for a varying current output of its generator to the master bus.



*Equalizer Coil
**Potential Coil

Note:
Both generator circuits are identical. Left generator circuit is detailed; right is not.

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Figure 7-3. Generator Control

ELECTRICAL POWER UNIT

Generators	(two)	AN3633-1, 300 A., 30 V.D.C., wide speed
Voltage regulator	(two)	E-1597-1
Reverse-current relay	(two)	A-750D Hartman
Battery	(two)	6FHM-13-1 Exide, 12 V., 88 A.H., wired in series
Inverter	(one)	E-1737-1, 1500 V.A. 3-phase 115 V.A.C. 400-cycle
Inverter	(one)	E-1617-1, 250 V.A. 3-phase 115 V.A.C. 400-cycle

REQUIREMENTS TO OPERATE

Load	*On Ground*	*In Flight*
1. Cabin ground blower (Ref. Note 3)	Both generators or ground power	Off at all times
2. Cabin airscoop heater	Off at all times	Operable by switch
3. A-C for Loran IFF *Automatic pilot (Ref. Note 7)	Ground power (test switch by-passes interlock)	Operable
4. Buffet and miscellaneous cabin equipment; installed on airplane factory serial numbers 43301 and 43302. (Ref. Note 6)	Ground power	Both generators

*26VAC for automatic pilot instruments is supplied by main inverter low-voltage transformer during interlock.

Notes:

1. Airplane condition—load computed with night icing in all columns.
2. Per cent of discharge—take-off and climb 20 per cent, cruise 15 per cent and landing 5 per cent.
3. Interlock circuit requires both generators operating or ground power connected before ground blower will function.
4. Operates 50 per cent "ON" under all short-time conditions, 5 minutes or less, 5 minutes out of 15 minutes for taxi, and 10 minutes out of 60 minutes for cruise.
5. Operates 1/6 (16.7 per cent) of the time for all short-time conditions, 5 minutes and under; 1 minute out of 15 minutes for taxi; 2 minutes "ON" out of 60 minutes for cruise.
6. Noted item will not function:
   On ground—
   Unless ground power is connected and "ON"
   In flight—
   Unless both generators are producing full voltage.
7. A-C load from 1500 V.A. inverter is reduced by an interlock on the ground. Interlock is bypassed when overload interlock override switch is actuated. In taxi column, following loads are removed by interlock relay: automatic pilot, 175 V.A.; IFF system, 225 V.A.; and LORAN receiver, 275 V.A. This reduces the d-c requirements of the inverter by approximately 29 amperes.
8. Generator output rating:
   135 amperes for ½ hour and ground cooling condition. Estimated output rating at 0°C, 350 amperes continuously.

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Figure 7-4 (Sheet 1 of 4 Sheets). D-C Power Electrical Load Chart

Equipment	Number of Units	Amperes per Unit	Operating Time—Min per Hour	Start and Warm Up: Amps	Start and Warm Up: Average Amps, Ground Power On: Prelim 1 Min	Start and Warm Up: Average Amps, Ground Power On: Start 2 Min	Start and Warm Up: Average Amps, Ground Power On: Run 3 Min	Taxi: Amps	Taxi: Average Amps: 1st 2 Min	Taxi: Average Amps: Total 15 Min	Take-Off and Climb: Amps	Take-Off and Climb: Average Amps: 1st 2 Min	Take-Off and Climb: Average Amps: Last 3 Min	Take-Off and Climb: Average Amps: Total 5 Min	Cruise: Amps	Cruise: Average Amps: Any 2 Min	Cruise: Average Amps: Total 60 Min	Landing: Amps	Landing: Average Amps: Last 2 Min	Landing: Average Amps: Total 5 Min
FLIGHT CONTROLS:																				
Automatic Pilot	1	4.00	60	4.00			4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00
Compass	1	1.50	60	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50
D-C turn-and-bank	1	0.75	60	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75
ANTI-ICING:																				
Pitot heaters	2	3.30	60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60	6.60
Scoop heater 51.00	1	51.35	30												51.35	51.35	51.35			
and relay 0.35																				
Alcohol pump motor	1	2.75	30												2.75	2.75	1.37	2.75	2.75	2.75
Boot motor	1	1.75	30												1.75	1.75	0.87	1.75	1.75	1.75
Alcohol quantity indicator	1	0.25	60	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25	0.25
ENGINE INSTRUMENTS:																				
Free air temperature indicator																				
Carburetor air temperature indicator		0.10	60	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30	0.30
Oil temperature																				
HEATING SYSTEM:																				
Ground flight relay 0.25																				
Ground blower relay 0.35		Noted	Blower grd only	58.75	58.75		58.75	58.75	58.75	58.75	6.15	6.15	6.15	6.15	6.15	6.15	6.15	6.15	6.15	6.15
Ground blower 52.00																				
Heater slave relay 0.25																				
Main fuel solenoid 0.30																				
Cycle solenoid 0.60																				
Fuel pump 2.00																				
Ignition coil 3.00																				
ENGINE CONTROLS:																				
Induction vibrator 3.00																				
Starter relay 2.00	2	Noted	0.50	205.00		102.50														
Starter 200.00																				
Primer	2	1.90	0.25	1.90		0.23														
Oil dilution	2	1.70	Last 2 min															1.70	1.70	0.68
Feather switch 0.75																				
and relay 2.00	2 each noted	Noted	15 sec								†202.75	26.09		10.13	†202.75	26.09	0.84	†202.75	26.09	10.13
Feathering motor 200.00																				
Cowl flaps	2	5.00	2	10.00		1.70	1.70	10.00	1.70	0.68	10.00	1.70	1.70	1.70	10.00	1.70	0.33	10.00	1.70	1.70
Oil cooler door motor	2	3.00	10	6.00		3.00	3.00	6.00	3.00	2.00	6.00	3.00	3.00	3.00	6.00	3.00	1.00	6.00	3.00	3.00
and thermo unit	2	0.25	60	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50	0.50
LIGHTING:																				
Panel floodlights	2	0.34	60	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68	0.68
Pilot's panel lights	28	0.04	60	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12
Center panel lights	20	0.04	60	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80	0.80
Co-pilot's panel lights	30	0.04	60	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12	1.12
Upper switch panel lights	24	0.04	60	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96

† Emergency items

For total see sheet 4

Figure 7-4 (Sheet 2 of 4 Sheets). D-C Power Electrical Load Chart

Equipment	Number of Units	Amperes per Unit	Operating Time—Min per Hour	Start and Warm Up: Amps	Start and Warm Up: Average Amps, Ground Power On: Prelim 1 Min	Start 2 Min	Run 3 Min	Taxi: Amps	Taxi: Average Amps: 1st 2 Min	Total 15 Min	Take-Off and Climb: Amps	Take-Off and Climb: Average Amps: 1st 2 Min	Last 3 Min	Total 5 Min	Cruise: Amps	Cruise: Average Amps: Any 2 Min	Total 60 Min	Landing: Amps	Landing: Average Amps: Last 2 Min	Total 5 Min
OPERATING CONDITIONS																				
LIGHTING: (Continued)																				
Radio panel lights	24	0.04	60	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96	0.96
Miscellaneous cockpit lights		Noted	60	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84	1.84
Two C-4 0.17 each																				
Two gooseneck 0.75 each																				
Dome lights	5	0.75	30	3.75	3.75	3.75	3.75	3.75	3.75	3.75					3.75	3.75	1.87			
Tail access	1	0.75																		
Lavatory dome	1	0.75	30	0.75	0.75	0.75	0.75								0.75	0.75	0.37			
Wing edge lights	2	1.95	2 and 10	3.90	3.90	3.90	3.90	3.90	3.90	3.90	3.90	3.90		1.56	3.90	3.90	0.65			
Landing lights 17.00 each and relays 0.35 each	2	17.35	10					34.70	34.70	20.82	34.70	34.70	34.70	34.70				34.70	34.70	34.70
Navigation lights		Noted	60	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59	3.59
Motor 0.50																				
Wing tip 0.75 each																				
Upper and lower 1.03 each																				
Tail 1.03 each																				
Main junction box panel lights	1	0.71	5	0.71	0.71	0.71	0.71													
POWER:																				
Battery charging and relay	See Note 2										66.00	66.00	44.00	52.80	49.50	49.50	14.85	16.50	16.50	13.20
Inverter—1500 V.A. noted	See Notes 7 and 8	Noted	60	97.35			97.35	68.35	68.35	68.35	97.35	97.35	97.35	97.35	100.35	100.35	100.35	97.35	97.35	97.35
Inverter— 250 V.A. 18.0																				
and control relay 0.35																				
FUEL SYSTEM:																				
Booster pumps	2	8.50	First 3 min	17.00	17.00	17.00					†17.00	17.00	17.00	17.00				†17.00	17.00	17.00
Fuel quantity		1.50	60	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50	1.50
RADIO:																				
ARR-15 norm.	1	3.10	60	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10	3.10
chan.		8.50	5 sec	8.50	0.70										8.50	0.70	Neg			
BC-348	1	2.00	60	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00
ARC-27 norm.	1	17.00	60	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00	17.00
xmtr		3.00	30 sec	3.00	1.50		0.50	3.00	1.50	0.10					3.00	1.50	Neg	3.00	1.50	0.10
ART-13 norm.	1	11.00	60	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00	11.00
xmtr		21.00	15 sec												21.00	2.62	Neg			
A1A-2A	1	4.00	60	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00	4.00
ARC-5 (one receiver)	1	2.00	60	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00
ARN-7	2	0.60	60	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20	1.20
ARN-8	1	2.00	60	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00
APN-1	1	2.80	60	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80	2.80
ARN-14	1	5.00	60	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00
ARC-1 norm.	1	7.00	60	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00	7.00
xmtr		8.00	15 sec												8.00	1.00	Neg			
ARN-18	1	2.25	60	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25	2.25
APX-6	1	1.40	60	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40	1.40
AM-40/A1C	1	2.00	60	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00	2.00

† Emergency items

For total see sheet 4

Figure 7-4 (Sheet 3 of 4 Sheets). D-C Power Electrical Load Chart

				OPERATING CONDITIONS																
				Start and Warm Up				*Taxi*			*Take-Off and Climb*				*Cruise*			*Landing*		
					Average Amps				*Average Amps*			*Average Amps*				*Average Amps*			*Average Amps*	
					Ground Power On															
Equipment	*Number of Units*	*Amperes per Unit*	*Operating Time—Min per Hour*	*Amps*	*Prelim 1 Min*	*Start 2 Min*	*Run 3 Min*	*Amps*	*1st 2 Min*	*Total 15 Min*	*Amps*	*1st 2 Min*	*Last 3 Min*	*Total 5 Min*	*Amps*	*Any 2 Min*	*Total 60 Min*	*Amps*	*Last 2 Min*	*Total 5 Min*
WARNING LIGHTS:																				
Generator inoperative	2	0.17	Emergency	0.34	0.35	0.17														
Inverter inoperative	2	0.17	Emergency	0.34	0.17	0.17														
Door open	1	0.17	Emergency	0.17	0.17															
Oil pressure	1	0.17	Emergency	0.17	0.17	0.17														
Fuel pressure	1	0.17	Emergency	0.17	0.17	0.17														
Engine fire warning	4	0.17	Test 0.25	0.68	0.17															
Heater fire warning	2	0.17	Test 0.25	0.34	0.08															
Generator failure warning	2	0.17	Emergency	0.34	0.17															
Warning light dimming relay	1	4.00	As required	4.00	Neg															
ALIGHTING GEAR:																				
Position indicator	1	0.75	60	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75	0.75
Isolation relay	1	0.20	Grd only	0.20	0.20	0.20	0.20	0.20	0.20	0.20	0.20	Neg						0.20	0.20	0.20
Unlock solenoid	1	2.50	30 sec								2.25	0.56		Neg				2.25	0.56	Neg
Warning horn	1	2.00	Emergency																	
Horn silence relays	2	0.25	Emergency																	
AIRCRAFT FACTORY SERIAL NUMBERS 43301 AND 43302																				
Toilet dome	1	0.62													0.62	0.62	0.62			
Bench light ▲	1	0.62						0.62	0.62	0.62	0.62	0.62	0.62	0.62				0.62	0.62	0.62
Razor inverter ▲	1	5.00													5.00	2.50	2.50			
Dome lights	6	0.62		3.72	3.72	3.72	3.72	3.72	3.72	3.72										
Reading lights	20	0.80	50 per cent	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00	8.00
Entertainment radio ▲	1	9.00		9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00	9.00
Night lights ▲	4	0.19		0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76	0.76
Table lights	2, 1	2.50, 0.62		5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00	5.00
"No smoking—fasten belt" sign	6	0.57		3.42	3.42	3.42	3.42	3.42	3.42	3.42	3.42	3.42	3.42	3.42				3.42	3.42	3.42
Shaving lights ▲	2	0.75													1.50	1.50	1.50			
Buffet light ▲	1	0.75		0.75	0.75	0.75	0.75								0.75	0.75	0.75			
Entrance door light	1	0.75		0.75	0.75	0.75														
Baggage compartment dome lights	2	0.62		1.24	1.24	1.24									1.24	1.24	1.24	1.24	1.24	1.24
Aft compartment dome	1	0.62		0.62	0.62	0.62		0.62	0.62	0.62										
Hot cup ▲	2	20.00	30												40.00	40.00	20.00			
Hot plate ▲	2	25.00	30												50.00	50.00	25.00			
Hot water heater ▲	1	20.00	50 per cent	20.00	10.00	10.00	10.00	20.00	10.00	10.00	20.00	10.00	10.00	10.00	20.00	10.00	10.00	20.00	10.00	10.00
			Total*		175	218	261		266	249		308	277	287		321	269		258	252
			Total**		214	258	297					345	314	324		492	397		296	290
			Total* plus †									351	294	314		347	270		301	279
			Total** w/o ground power		131	234	215		283	266		††324	††293	††304		377	327		275	269

▲ See Sheet 1, note 6

*Aircraft C and D and 1 through 96
**Aircraft A and B
†Emergency items
††With one generator out

Figure 7-4 (Sheet 4 of 4 Sheets). D-C Power Electrical Load Chart

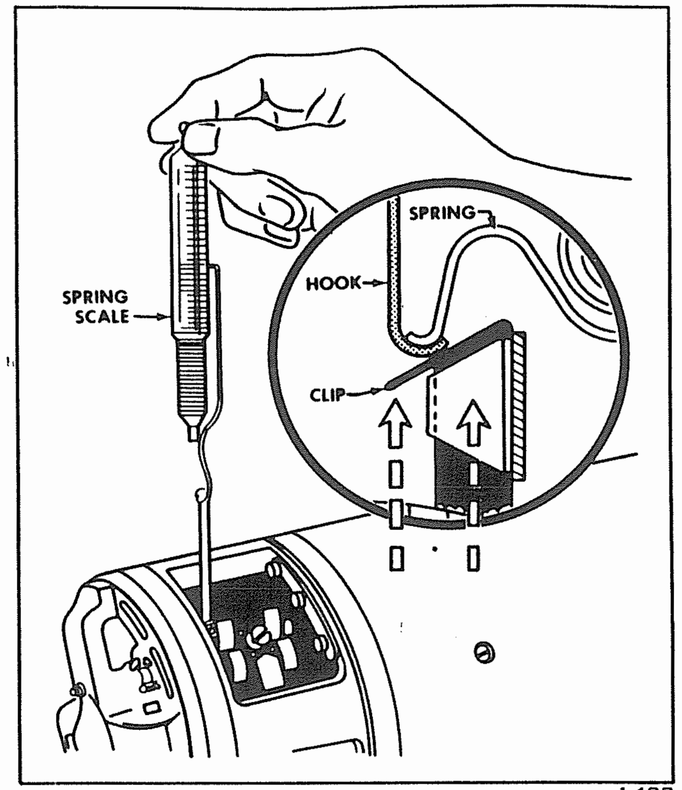


Figure 7-5. Testing Generator Brush Spring Tension

current relay, an over-voltage relay, and a generator ammeter. The voltage regulators are used to maintain constant voltage output of the generator and to equalize the load of the two generators. The voltage drop across the generator compensating winding is used as the means of controlling generator load paralleling. The reverse current relay operates both to close the main lead circuit when the generator voltage is ½-volt higher than the bus voltage, and to disconnect the generator when its voltage drops below bus voltage and reverse current occurs. The over-voltage relay drops the generator from the circuit when the generator output exceeds 30 volts. The generator is a 300-ampere wide-speed-range type, operating between 3000 and 8000 rpm and capable of continuous full rated load over the entire speed range. Vent holes are provided at the drive end of the generator and a blast tube at the opposite end provides cooling. A vibration-damping coupling is provided between the engine driving member and the generator armature to reduce rotational shocks.

7-8. GENERATOR INSPECTION AND MAINTENANCE.

a. Detach the brush inspection band and air inlet cover.

b. Wipe out the inside surfaces and see that there are no dents or cracks in the cover or obstructions to the free passage of cooling air.

c. Check for loose parts, broken screws, loose bearing retaining nuts, defects in the cross connector, and cracks or dents in the brush rigging, adapter, drive-end bearing support, and terminal block.

d. Examine the generator for the presence of brush and copper particles and engine oil. Absence of such matter usually indicates proper generator operation. Presence of brush and copper particles may indicate unsatisfactory condition of brushes and commutator. Presence of engine oil may indicate that the engine oil is leaking into the generator through the back head ball bearing.

7-9. MINOR REPAIR AND REPLACEMENT OF GENERATOR BRUSHES.

a. Check the brushes for free fit.

b. Use a continuity light to test for grounds. Touch one test prod of the continuity light to the front head and touch each brush box in succession with the other test prod. If a brush box is grounded, the lamp will illuminate.

Note

The generator must be isolated before making this test. The simplest way to isolate the generator is by disconnecting the firewall plugs.

c. Clean binding brushes and corroded brush boxes by wiping them with a cloth moistened in unleaded gasoline. *Do not* use carbon tetrachloride, as its use will cause rapid brush wear.

d. Check the length of each generator brush. Replace worn brushes before their maximum wear limit is reached to insure satisfactory operation until the next brush inspection period. When new brushes are being installed, make certain not to twist the brush leads. New brushes should be run-in with the generator on the bench test stand, turning over at approximately 3000 rpm.

e. Test the tension of the brush holder springs. If these springs are weak or misshapen, or if they are found to be blued (indicative of overheating and probable loss of temper), they should be replaced. Use an instrument scale to measure the tension of any accessible brush spring *(see figure 7-5)*. With the bottom surface of the brush spring arm, (which normally rests on the top of the brush clip), raised ⅛ inch above the top of the brush box, the tension should be approximately 23 ounces maximum per brush for a full brush, and 15 ounces per brush for a worn brush. These tolerances are important for high-altitude operation.

7-10. MINOR REPAIR AND REPLACEMENT OF GENERATOR WIRING CONNECTIONS.

a. Inspect all connections to be sure that they are clean and tight.

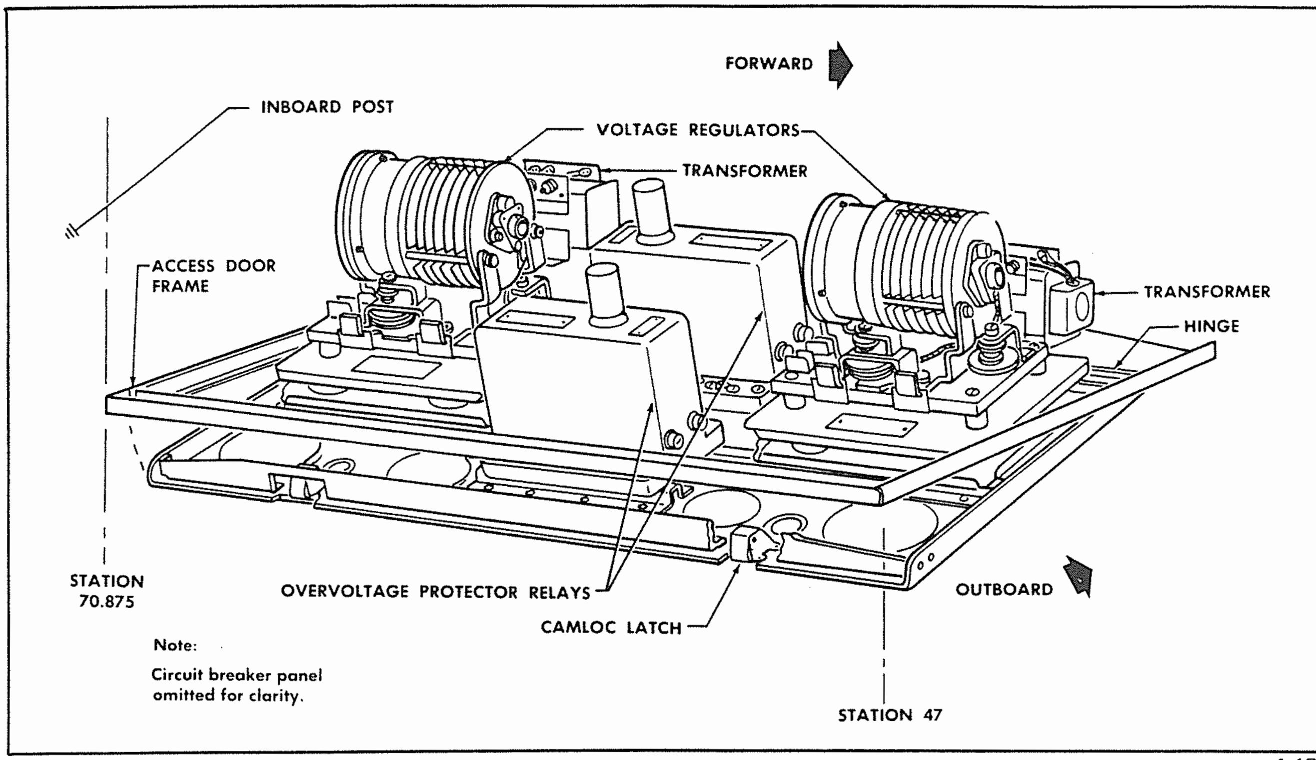


Figure 7-6. Voltage Regulator Compartment

b. Replace any wiring that has scuffed, frayed, or burned insulation.

c. Inspect the terminal posts for stripped threads and for burning.

d. Resolder or replace loose or broken terminals.

7-11. FLASHING GENERATOR. The aircraft's generators are self-exciting because of a low permanent residual magnetism in the field poles. This magnetism is sometimes lost entirely or becomes reversed in polarity, which is indicated on the ammeter by no output or a reverse output from the generator. In this case, it is necessary to renew the residual magnetism or to correct its polarity by sending a flow of current through the generator field windings in the proper direction (from terminal A to terminal E or D). Fundamentally, therefore, positive potential is applied to field terminal E or D of the generator. This is most easily accomplished in the following manner:

a. Turn the generator switch to the OFF position.

b. Remove the corresponding voltage regulator from its mounting base.

c. Connect a precision portable voltmeter, having a 30-volt range, into the voltmeter jacks provided at the regulator.

d. Connect one end of a wire to the positive bus.

e. Operate the corresponding engine at approximately 1200 rpm.

f. Observe the voltmeter while striking terminal F (or F plus if so designated) of the voltage regulator base with the opposite end of the wire, which should now be charged from the main bus. The voltmeter should indicate generator voltage output immediately after flashing (reinstall the voltage regulator). If the voltmeter shows no reaction, the failure of the generator to charge is not due to loss of residual magnetism but is caused by some other circuit defect.

g. Occasionally a generator will not build up voltage when brought up to speed, and it is commonly believed that it has "lost its residual." This of course, is possible, but not necessarily the only reason for such a condition. For example, if the aircraft is being operated in areas of high humidity and temperature, as in some tropical regions, and at relatively low altitudes, with low generator load, the commutator may develop too much film, especially with high-altitude brushes. If this should occur, the brush contact resistance may increase sufficiently to prevent build-up after the generator has stood idle for a few hours. Flashing the field will usually cause the generator to operate properly again. However, the trouble may recur if the generator is not made to carry more load when running.

7-12. REMOVAL OF GENERATOR.

a. Remove the accessory cowling.

b. Disconnect the plugs from the firewall.

c. Remove the generator blast tube.

d. Loosen the generator mounting nuts.

e. Disconnect the wiring from the terminal block.

f. Remove the nuts, starting at the bottom.

Note

Because of its weight, two men are required to remove a generator.

g. Remove the generator.

h. Install the drive pad cover over the engine opening.

7-13. INSTALLATION OF GENERATOR.

a. Remove paint, grease, and dirt from the generator flange to provide electrical bonding contact.

CAUTION

While the generator is being mounted, it must be continuously supported by a sling with its shaft parallel to the studs and in line with the driving element. Because of its weight, two men are required to install a generator.

b. Place the generator on the mounting studs.

c. Tighten the generator mounting nuts to 275- to 300-inch-pounds torque and safety them.

d. Connect the wiring to the generator terminal block.

e. Install the blast tube on the generator blast tube adapter and tighten the screws.

f. Connect the generator plugs at the firewall.

7-14. GENERATOR VOLTAGE REGULATORS. *(See figure 7-6.)* Before attempting to adjust the voltage regulators with the generator switches in the OFF position, check the voltage range of each regulator. With the adjusting knob rotated in the full clockwise position, the regulator voltage should indicate 30 volts. With the adjusting knob rotated in the full counterclockwise position, the regulator voltage should indicate 25 volts. If there are 29 volts or less with the adjusting knob in full clockwise position, or if there are 26.5 volts or more in the full counterclockwise position, replace the regulator. Each voltage regulator must be adjusted individually.

7-15. ADJUSTMENT OF GENERATOR VOLTAGE REGULATOR.

Note

Before starting the engines make the following adjustment to each voltage regulator: the slider arm of the paralleling potentiometer, mounted on the regulator base, should be adjusted one quarter of the distance up from the end of the potentiometer that connects to the EG terminal.

a. Turn all generator switches on the forward overhead electrical panel to their OFF positions.

b. Start the engines and allow them to warm up to normal operating temperatures. Run them for 20 to 30 minutes at approximately 1000 rpm to allow the voltage regulators to warm up and stabilize.

Note

During this period, operation of the voltage regulators may appear to be erratic even though the regulators are adjusted properly. Therefore, make no adjustments of the regulators during this warm-up period.

c. Plug a 0 to 30-volt voltmeter (accurate to 1/2 of 1 per cent) into the jacks provided at the regulator.

d. Place the battery selector switch in PLANE BATTERY position.

e. Increase the speed of the respective engine for the generator and voltage regulator being checked to 1600 rpm. Operate the other engine at idling speed.

f. Stabilize the voltage regulator by closing the generator switch for the regulator circuit being checked (leave the other generator switch OFF). Throw a heavy load of 150 to 200 amperes on and off the system several times.

g. With all loads off and both generator switches OFF, adjust the voltage regulator until the test voltmeter reads 28.0 volts. (Turn the adjusting knob on the voltage regulator clockwise to raise the voltage and counterclockwise to lower the voltage.)

h. Repeat steps c through g, preceding, for the other voltage regulator.

i. After completing the adjustments in steps a through h, preceding, the system is ready for operation. Disconnect the ground power supply from the aircraft, and place the generator switches in the ON position.

j. Increase and synchronize the speed of both engines to 1600 rpm, and apply a maximum load of approximately 200 amperes to the system.

k. Check ammeter readings for both generators to determine if each generator is carrying its share of the 200 ampere load plus or minus 10 per cent of the total load. For example: add both ammeter readings and divide the total by 2 to obtain the load for each generator; divide the total ammeter reading by 20 to obtain the tolerance. If the proper division of current is not maintained, the voltage regulator rheostat for the generator having the least current should be turned very slightly in a clockwise direction. Recalculate the generator carrying load with the ammeter total load to determine if the generator falls within its proper limits. Repeat adjustment of the regulator, if necessary.

l. After completing all adjustments, check the main bus voltage. The allowable voltage is between 27.75 and 28.25.

m. Increase the speed of both engines to as high a value as safety will permit, and check the division of

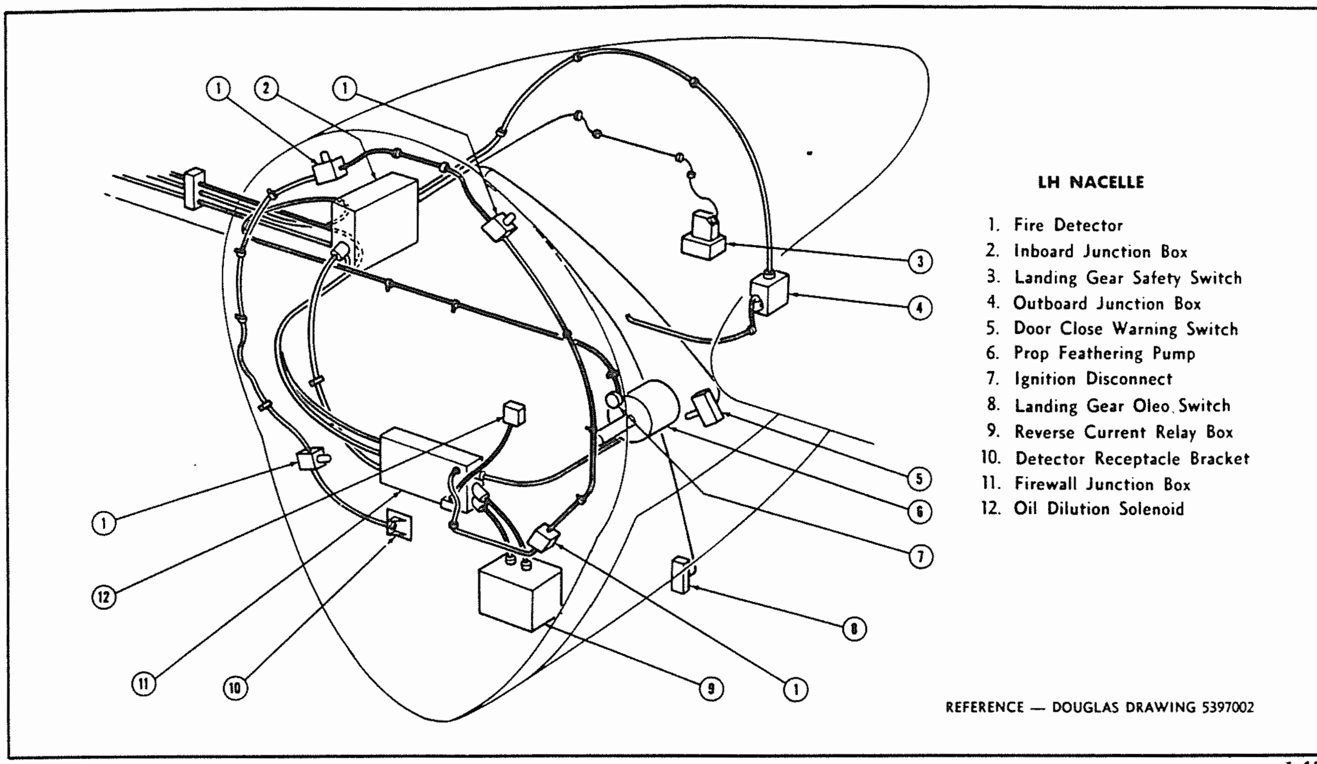


Figure 7-7. Firewall Electrical Equipment

current as the system loads are applied. The difference between the maximum and minimum readings should not exceed 30 amperes between generators at any value of system load.

7-16. ADJUSTMENT OF GENERATOR VOLTAGE REGULATORS FOR PARALLELING. For parallel operation, it is particularly important that the regulators be adjusted to the same voltage (after allowing time for warm-up) and that their voltage characteristics be matched as nearly as possible over the speed and load range of the generator. The parallel voltage drop is obtained from the generator compensating field and is adjusted by the paralleling resistor on the control panel. Before adjusting for parallel operation, both regulators must be adjusted as described in paragraph 7-15. Then proceed as follows:

a. Operate both engines at 1200 rpm.

b. Place both generator switches on the overhead electrical panel in the ON positions and allow time for the regulators to warm to operating temperatures.

c. Apply a d-c load of approximately 75 to 100 amperes to the circuit.

d. Check the ammeter reading for each generator. The readings should be steady and each generator should be carrying half the load. A variation of 10 per cent, plus or minus, is allowable. For example, suppose the total load (obtained by adding both ammeter readings) is 100 amperes. Each ammeter should then read about 50 amperes, and neither should read lower than 45 amperes nor higher than 55 amperes.

e. If the load distribution is not within 10 per cent of the ideally equal division of load, note which generator is carrying the greater load.

f. Adjust the equalizer rheostat to increase or decrease the load carried by that generator. Turning the knob clockwise will decrease the load and turning it counterclockwise will increase the load.

Note

Adjustment of the equalizing resistor of any one of the control panels will alter the load carried by each generator in the system; therefore, carefully observe the readings of both generator ammeters as the equalizer resistor is being adjusted.

g. Continue the equalizer rheostat adjustments until proper division of load is obtained (always select for adjustment the control panel equalizer resistor of the generator carrying the greater load).

h. After all adjustments have been made, a final check of the bus voltage should be made from the positive bus to ground. The bus voltage should be between 27.75 and 28.25 volts.

CAUTION

All voltage adjustments must be made with a voltmeter calibrated to an accuracy of at least ½ of 1 per cent.

i. Recheck division of current as the system loads are applied. Maximum and minimum generator ammeter readings should be within the allowable limits.

j. If, after the above paralleling adjustment has been made, it becomes necessary to replace a defective regulator, it is only necessary to adjust the new regulator to 28 volts. No change in equalizer resistor adjustment should be necessary. If, however, a generator or its wiring is changed, it may be necessary to readjust the equalizer resistor.

7-17. REMOVAL OF GENERATOR VOLTAGE REGULATORS. *(See figure 7-6.)*

a. Release the 2 latches on the bottom of the voltage regulator compartment door assembly between station 47 and station 70 to gain access to the left and right engine voltage regulators.

b. Remove the voltage regulator by releasing the two clips in the base assembly.

7-18. INSTALLATION OF GENERATOR VOLTAGE REGULATOR. *(See figure 7-6.)* Reverse the removal procedure.

7-19. DIFFERENTIAL REVERSE-CURRENT RELAY. *(See figure 7-7.)*

7-20. CHECK OF DIFFERENTIAL REVERSE-CURRENT RELAY. To check the operation of the reverse-current relay in the aircraft, proceed as follows:

a. Obtain two precision-type d-c voltmeters, 0 to 30-volt scale, of known accuracy and calibration.

b. Locate the two voltmeters so that they and the appropriate generator ammeter are visible at the same time.

c. By means of suitable wires, connect voltmeter No. 1 to the main bus and to structure ground.

d. Connect voltmeter No. 2 to one terminal of the generator field circuit breaker of the generator system being checked, and to structure ground.

e. Reduce the load on the main bus to as low a value as possible, and still obtain positive results.

f. Place the power selector switch in the BATTERY position and place both generator switches OFF.

g. With the engines idling, note the voltage reading on voltmeter No. 2. It should be lower than the voltage shown on voltmeter No. 1. If not, reduce the speed of the appropriate engine further until the generator voltage is lower than the bus voltage.

h. Turn the generator switch ON and slowly increase engine speed, observing the two voltmeters and the ammeter. At the instant that the reverse current relay closes, as indicated by a positive current reading on the ammeter, note both voltages. The reverse current relay should close when the generator voltage is between a minimum of 0.3 or a maximum of 0.6 volts higher than the bus voltage.

i. The bus voltage is subject to variation, due to load demands and the state of charge of the battery, but in all cases the relay should close when the generator voltage exceeds the bus voltage by a minimum of 0.3 or a maximum of 0.6 volts. If the relay cannot be made to fulfill the above requirements, replace the reverse current relay, as the relay cannot be adjusted in the aircraft.

j. After the above test has been performed satisfactorily, slowly decrease engine speed until the relay opens. Note the reverse current which is indicated just before the ammeter needle returns to zero. This value should be between a minimum of 15 amperes and a maximum of 40 amperes. If the reading does not fall between these limits, remove and replace the reverse current relay.

Note

With the wide-speed-range generators, the minimum idling speed of the engines may be too high to allow sufficient reverse current to trip the relay if the battery is the only source of reverse current. If this occurs, increase the speed of the other engine and close its respective generator switch, noting that the relay for this generator closes. This will maintain the bus voltage at a higher level than will the battery alone and should provide sufficient reverse current to trip the relay being tested.

7-21. ADJUSTMENT OF DIFFERENTIAL REVERSE-CURRENT RELAY. The differential reverse-current relays are bench set and the adjustment screws are sealed with wax. The entire unit is enclosed in a dustproof case and cannot be adjusted in the field. In the event of faulty operation of this type of reverse-current relay, remove the relay and replace it with a serviceable unit.

7-22. REMOVAL OF DIFFERENTIAL REVERSE-CURRENT RELAY.

a. Remove the cover from the relay box assembly located on the aft outboard side of the firewall below the firewall junction box.

b. With the generator main line and battery switches OFF, disconnect the wires from the terminals on the connector panel and bus bar from the relay.

c. Remove the attaching screws from the relay base to remove the relay from the box assembly.

7-23. INSTALLATION OF DIFFERENTIAL REVERSE-CURRENT RELAY. *(See figure 10-21.)* Reverse the removal procedure.

7-24. OVERVOLTAGE PROTECTOR RELAY. The overvoltage protector relay is located in the voltage regulator compartment above the flight compartment entrance door.

7-25. OVERVOLTAGE PROTECTOR RELAY CHECK-OUT PROCEDURE.

Note

With the generator system adjusted as outlined in paragraph 7-15, the following test should be made while the aircraft is on the ground to ascertain if the overvoltage protector relay is functioning correctly.

a. Note the position of the reset button in each of the two overvoltage protector relays, in order to determine during the following test whether the reset button trips or stays in.

b. After the engines and the voltage regulators have been operated a minimum of 15 minutes, place both generator switches in the ON position, and adjust the engine speed to the lowest rpm possible to connect the generators to the bus, connection being indicated by the ammeters showing a positive current flow.

c. Momentarily connect a jumper wire in such a manner as to short terminals T and L of the overvoltage protector relay. Shorting of these terminals should cause the reset button on the relay to trip, returning the corresponding ammeter for the tripped relay to zero, thus indicating that the generator for that circuit has been disconnected.

Note

The jumper wire used for the preceding test must not be connected to the relay terminals after the test has been completed and should only be used with the permanent connections intact on the overvoltage protector relay terminals.

d. Observe the position of the reset button on the relay to make sure that the relay has tripped. If the relay did not trip during the test, the relay is defective and must be replaced.

e. Before resetting the overvoltage protector switch, check the continuity between terminals C and L. The resistance between the 2 terminals should be approximately 65 ohms.

f. Repeat steps a through e for the second overvoltage regulator compartment door assembly between sta-

7-26. REMOVAL OF OVERVOLTAGE PROTECTOR RELAY.

a. Release the 2 latches on the bottom of the voltage regulator compartment door assembly between station 47 and station 70 to gain access to the left and right generator overvoltage relays.

b. Remove the wires from the overvoltage relay terminals.

c. Remove the overvoltage relay from the voltage regulator compartment by removing the four screws, washers, and nuts from the base of the relay.

7-27. INSTALLATION OF OVERVOLTAGE PROTECTOR RELAY. Reverse the removal procedure.

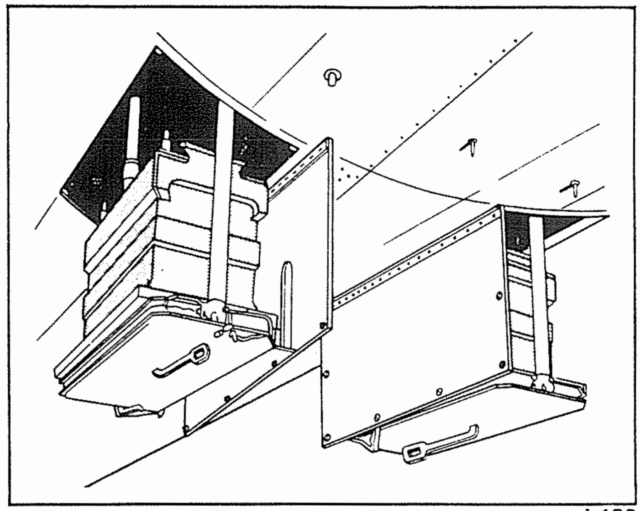


Figure 7-8. Battery Installation

7-28. GENERATOR MAIN LINE SWITCHES. Two generator main line switches, one for each generator circuit, are installed in the co-pilot's right overhead electrical panel *(see figure 7-17)*. The main line switch connects the generator and the battery circuit through the circuit breaker bus in the main electrical junction box.

7-29. REMOVAL OF GENERATOR MAIN LINE SWITCHES. It will be necessary to remove the right overhead electrical panel to gain access to the generator main line switches. For removal of the overhead electrical panels, see paragraph 7-91.

7-30. INSTALLATION OF GENERATOR MAIN LINE SWITCHES. *(See figure 10-21.)* Reverse the removal procedure.

7-31. MAIN GENERATOR AMMETER SHUNTS. Main generator ammeter shunts (AN3200-450) are installed on the bottom of both left and right nacelle firewall junction boxes to provide amperage load indications from the ground leg of the generator to the ammeter. The ammeter shunts are the external type, 50-millivolt, and rated at 450 amperes.

7-32. REMOVAL OF MAIN GENERATOR AMMETER SHUNTS.

a. Remove the firewall junction box cover.

b. Disconnect the wires from the two large and two small terminals of the shunt.

c. Remove the two attaching screws, washers, and nuts from the bottom of the junction box and shunt.

7-33. INSTALLATION OF MAIN GENERATOR AMMETER SHUNT. Reverse the removal procedure. When installing the wires on the shunt, note that the positive wire on the shunt is also the positive wire on the ammeter.

7-34. GENERATOR AMMETERS. Two generator ammeters are mounted on the right overhead electrical

panel to measure the load imposed on the direct-current system for each generator, from 0 to 450 amperes. Each ammeter is wired through the 450-ampere ammeter shunts mounted on the bottom of firewall junction boxes in each nacelle.

7-35. REMOVAL OF GENERATOR AMMETERS. It will be necessary to remove the right overhead electrical panel to gain access to the generator ammeters (see paragraph 7-91 for removal of the overhead electrical panels).

7-36. INSTALLATION OF GENERATOR AMMETERS. Reverse the removal procedure.

7-37. BATTERY SYSTEM. Two 12-volt, 88-ampere-hour batteries are connected in series and installed in the bottom of the forward fuselage, forward of station 97.5 *(see figure 7-8)*. The battery terminals are accessible through the flooring above the battery compartment. Each battery is mounted on spring-loaded platforms on each side of the bottom centerline of the aircraft. The batteries are accessible through individual access doors. To save weight, aircraft batteries have exceedingly small capacity; consequently, they must be used and serviced with great care. For example, if connected to a charging source only a fraction of a volt above normal, an aircraft battery will overcharge considerably in a short time. Conversely, if undercharged, the battery may have insufficient power when called into emergency service. These are some of the reasons why great stress has been laid on accurate and careful setting of the voltage regulators.

7-38. HYDROMETER TEST OF BATTERY SPECIFIC GRAVITY.

a. When testing the specific gravity of an aircraft battery, use an approved temperature-corrected hydrometer. Hold the hydrometer in a vertical position with the hose inserted in the cell to be tested. The hydrometer reading should be taken at eye level. Be sure that the float is free (not sticking to the side of the hydrometer). When filling the hydrometer, draw in *only* enough electrolyte to raise the hydrometer float from the plug upon which it normally rests. After the test, always return the electrolyte to the cell from which it was removed. The temperature-correction scale should be read for necessary addition or subtraction to adjust specific gravity reading to temperature variation.

CAUTION

The sulphuric acid in the electrolyte is extremely injurious to the skin and to clothing. If acid is spilled, it should be neutralized with sodium bicarbonate (baking soda), or boric acid, and washed with water. If acid is spilled on the aircraft structure, neutralize with sodium bicarbonate or boric acid and flush with clear water.

b. A temperature-compensated hydrometer should be used. This type of hydrometer contains a scale that indicates the temperature corrections to be made for various conditions of charge. With the electrolyte in the battery at a temperature between 18.3°C and 35°C (65°F and 95°F), the specific gravity of the electrolyte in a fully charged battery is approximately 1.275. If, however, the temperature of the battery is between 18.3°C and 35°C (65°F and 95°F), no correction need be made, since the correction factor between these temperatures is so small as to be negligible. If a temperature-correcting hydrometer is not available, any hydrometer may be used and the following temperature corrections made for various electrolyte temperatures. Note that it is the temperature of the electrolyte that is critical and not the temperature of the surrounding air.

Electrolyte Temperature (Degrees F)	*Specific Gravity Correction (Points)*	
	Add to Reading	*Subtract from Reading*
140	24	—
130	20	—
120	16	—
110	12	—
100	8	—
90*	—	—
80*	—	—
70*	—	—
60	—	8
50	—	12
40	—	16
30	—	20
20	—	24
10	—	28
0	—	32
−10	—	36
−20	—	40
−30	—	44

*No Correction.

7-39. ADDITION OF WATER TO BATTERY. When it is necessary to add water to a battery, use clean, drinking water. Distilled water is preferable but not necessary, as the detrimental effect on aircraft battery life due to impurities in drinking water is negligible.

CAUTION

Never add anything but water to the battery. High specific gravity of the electrolyte in the battery indicates a high percentage of acid. However, this acid concentration is not the reason for the high state of charge of the battery, but merely the result of its having been charged. Therefore, if acid is added to the battery when it is in a discharged state, it will not result in charging the battery.

7-40. REMOVAL OF BATTERY.
(See figure 7-9.)

a. Turn the battery master switch OFF.

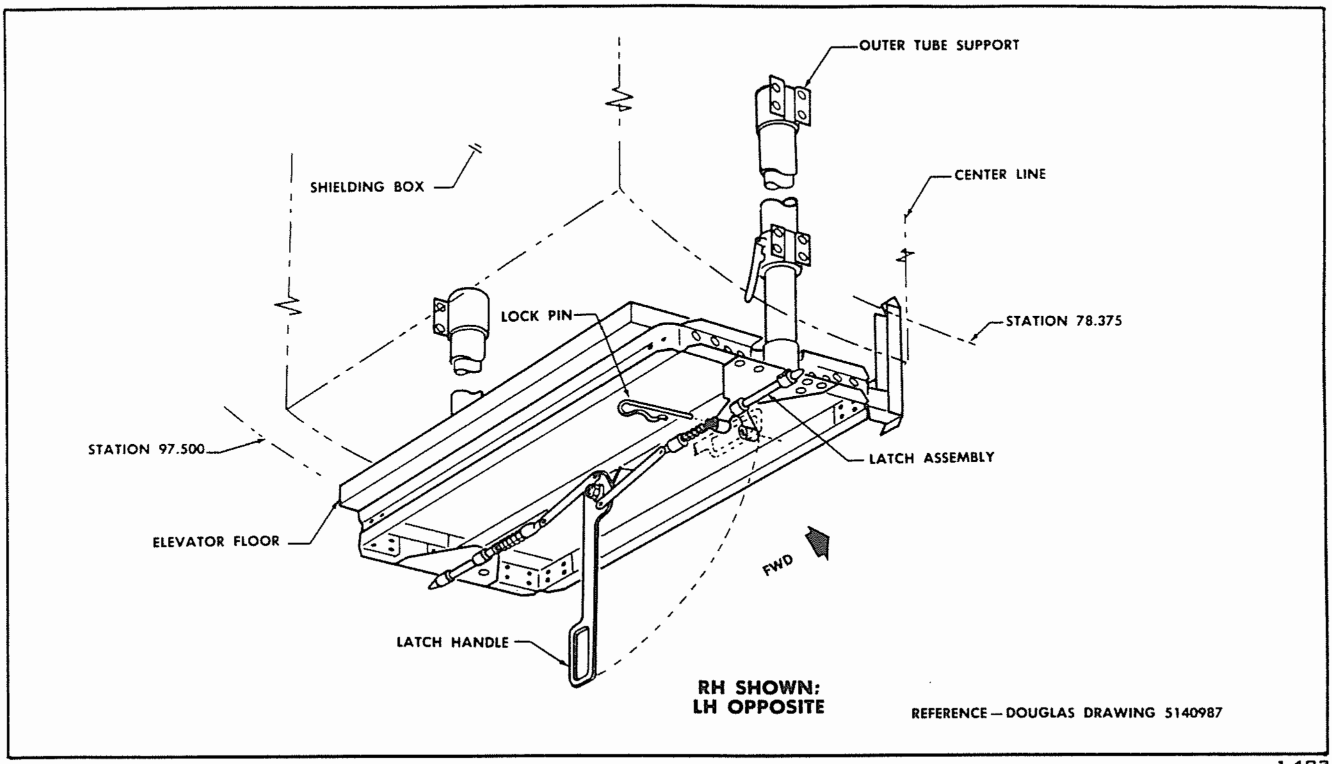


Figure 7-9. Telescopic Battery Support

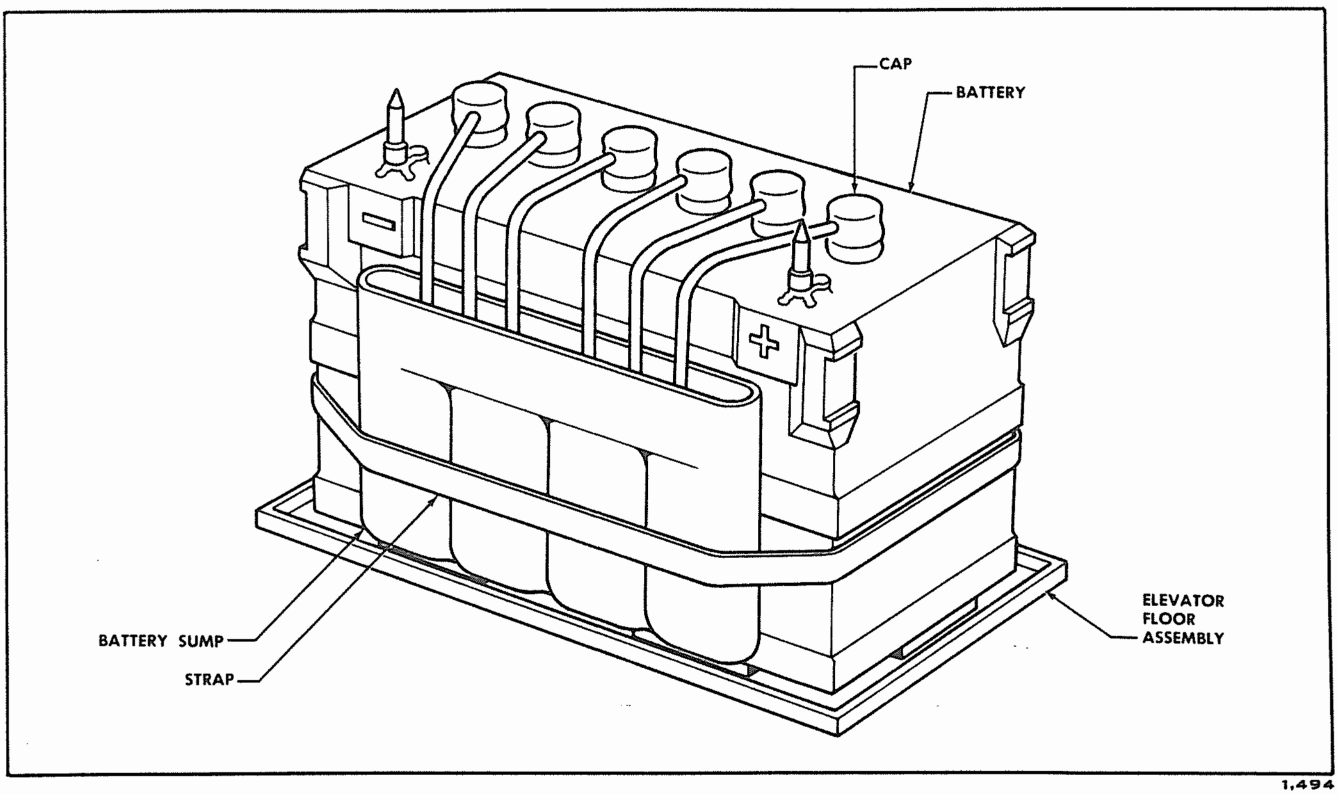


Figure 7-10. Battery Sump

b. Open the battery elevator access door by releasing the fasteners.

c. Pull out the elevator release handle locking pin.

d. Pull the release handle and the battery platform until the catch on the slide tube engages.

7-41. INSTALLATION OF BATTERY. Reverse the removal procedure.

7-42. BATTERY SUMP. *(See figure 7-10.)* A sump is strapped to the side of each battery. A rubber cap with an integral overflow tube is slipped over each cell cap on the battery and the overflow tubes discharge into the sump. With the exception of emptying the sump containers, there should be no maintenance required on the sumps and cap assemblies. *The presence of an appreciable amount of electrolyte in the sump is an indication of battery overcharging and should be investigated immediately.*

7-43. BATTERY CONNECTOR RELAY. One Type B-4 No. 94-32324-A battery connector relay is installed on the outboard side of the main electrical junction box. The relay is designed to provide a means of remotely controlling the battery circuit by the operation of the battery switch located on the pilot's left overhead electrical panel. The relay consists of a holding coil, which when energized or de-energized by the battery switch, causes the opening or closing of the main contacts of the relay connecting the batteries to the aircraft electrical system.

7-44. REPLACEMENT OF BATTERY CONNECTOR RELAY. When properly installed and operated, the battery connector relay should require little or no attention between overhaul periods. If the coils or contactors are found to be faulty, it will be necessary to replace the relay.

7-45. REMOVAL OF BATTERY CONNECTOR RELAY.

a. Turn off the battery switch in the cockpit.

b. Disconnect all wires from the relay.

c. Remove the attaching screws from the base of the relay and junction box.

7-46. TEST OF BATTERY CONNECTOR RELAY BEFORE INSTALLATION.

a. Place the relay in a horizontal position and connect the holding coil to a variable d-c supply.

b. Connect a 0 to 50-scale voltmeter across the coil terminals.

c. Connect a test lamp across the heavy switch terminals to indicate the closing of the main contacts.

d. Gradually increase the voltage until the main contacts close. This value must be 18 volts, or less, at room temperature.

e. Gradually reduce the voltage from the closing value and note the voltage at which the test light goes out, indicating the opening of the main contacts. This value must not be more than seven volts.

f. If the relay does not comply to the foregoing test, replace the relay with a serviceable unit.

7-47. INSTALLATION OF BATTERY CONNECTOR RELAY. *(See figure 10-14.)* Reverse the removal procedure.

7-48. BATTERY MASTER SWITCH. *(See figure 7-16.)* One battery master switch is installed on the pilot's overhead electrical panel. The switch operates the battery connector relay holding coil to connect the batteries to the electrical system. The switch is a three-position, selector-type switch that selects electrical power supply from either the aircraft batteries or the external battery cart, or it disconnects the electrical power supply.

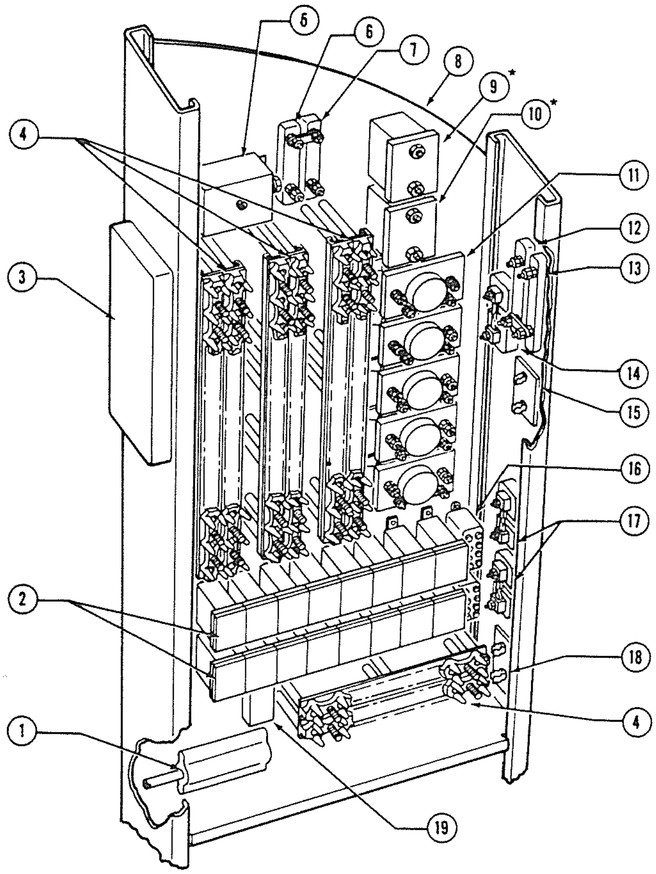
7-49. EMERGENCY POWER SWITCH. A guarded, two-position emergency power switch is mounted above the overhead radio control panel in the cockpit. The switch is guarded in the NORMAL position. In the event of electrical power failure at the main bus, the switch is thrown into the EMERGENCY position, thus disconnecting the battery from the main bus, and connecting the instrument panel flood lights, the cockpit portable lights, the main junction box flood light, and the co-pilot's turn-and-bank indicator directly to the battery. All power can be removed from the rest of the aircraft by turning off the generators.

WARNING

In the event of severe overload on items of electrical equipment, or of short circuit, causing overheating and power failure at the main bus, immediately turn ON the emergency power and turn OFF both generators and the aircraft's battery.

7-50. EXTERNAL POWER RECEPTACLE PROVISIONS. *(See figure 7-12.)* A battery cart receptacle (AN2552-A3) is provided on the under surface of the fuselage in a junction box between station 177.5 and 191. A ground power cart connector relay (AN3025-1) and a 5-ampere circuit breaker (AN3160-5) are also installed in the junction box to automatically connect the external power supply to the electrical system when the external power supply is plugged in. Before connecting the ground power cart plug to the receptacle, the battery master switch located on the pilot's upper overhead electrical panel should be placed in the OFF position.

7-51. MINOR REPAIR AND REPLACEMENT OF EXTERNAL POWER RECEPTACLE JUNCTION BOX. When properly installed and operated, the external power receptacle, the external power supply relay, and the 5-ampere circuit breaker in the junction box should require little or no attention between overhauls.



1. Warning Light Dimming Control Switch
2. Electro Switch Relay (16 Reqd)
3. AC Fuse Panel
4. Terminal Strip Panel
5. Master Radio Feed Filter
6. Secondary Bus FLLG 225 Amp Limiter (Stowage)
7. Radio Feed Master FLLC 100 Amp Limiter (Stowage)
8. Main Junction Box Aft Panel
9. * Throttle Servo Power Relay
10. * Automatic Approach Power Relay
11. AN3350-2 Relay (5 Reqd)
12. Radio Feed Master FLLC 100 Amp Limiter
13. Secondary Bus FLLG 225 Amp Limiter
14. Airscoop Heater Ammeter Shunt
15. Heater Airscoop Fuse
16. Inverter Warning Relay (2 Reqd)
17. Left and Right Pitot Heater Shunt
18. Heater Drop-Out Fuse
19. Capacitor

*AIRCRAFT C.D. AND 9 THROUGH 96

REFERENCE — DOUGLAS DRAWING 5397008

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Figure 7-11. Main Junction Box Electrical Equipment

a. In the event that the ground power electrical circuit does not function with the battery master switch placed in the GROUND POWER position and with the ground power supply plugged into the battery cart receptacle, check the 5-ampere circuit breaker (AN3160-5) adjacent to the receptacle. If the circuit breaker is defective, replace it with a serviceable circuit breaker.

b. Check the ground power cart connector relay (AN3025-1) by connecting a heavy jumper cable across the heavy contact terminals on the relay. With the master battery switch in GROUND POWER position, there should be electrical power from the battery cart. Replace the ground power cart connector relay with a serviceable unit if it is defective.

7-52. REMOVAL OF GROUND POWER SUPPLY RECEPTACLE JUNCTION BOX. *(See figure 7-12.)*

a. Remove the 20 attaching screws around the edge of the junction box.

b. Remove the bus and wires from the receptacle, relay, and circuit breaker.

7-53. INSTALLATION OF GROUND POWER SUPPLY RECEPTACLE JUNCTION BOX. Reverse the removal procedure.

7-54. ALTERNATING-CURRENT POWER SUPPLY. *(See figure 7-13.)* Two inverters are installed to provide a source of 400-cycle, a-c power. The main inverter supplies power to the pilot's instruments, a-c radio, and the automatic pilot. It is located on the left side in the tail section of the aircraft between stations 596 and 609. The main inverter is connected to the rear bus through a 125-ampere circuit breaker located above the main inverter at station 602. A-c power is supplied from the main inverter through two 15-ampere circuit breakers, located on a panel above the main inverter, forward of station 609. A transformer located in the voltage regulator compartment changes the voltage from 115 volts to 26 volts, 3-phase, at 400 cycles a-c for instrument loads. The auxiliary inverter is rated at 115 volts, 250 volt-amperes for either 3-phase or single-phase power. It is connected to the main bus in the main junction box through a 25-ampere circuit breaker located on the circuit breaker panel. The inverter is controlled by a relay located in the main junction box which is energized by the auxiliary in-

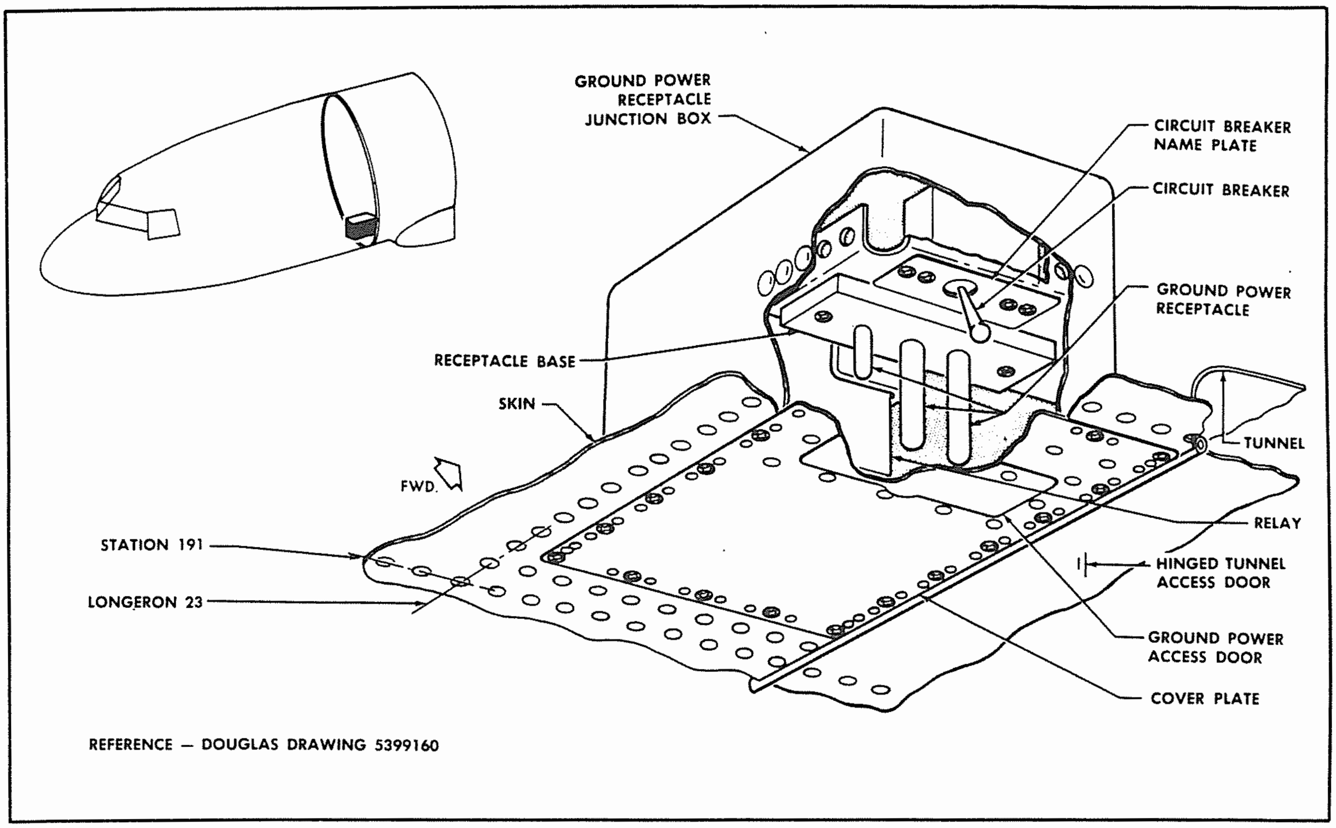

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Figure 7-12. External Power Receptacle Junction Box

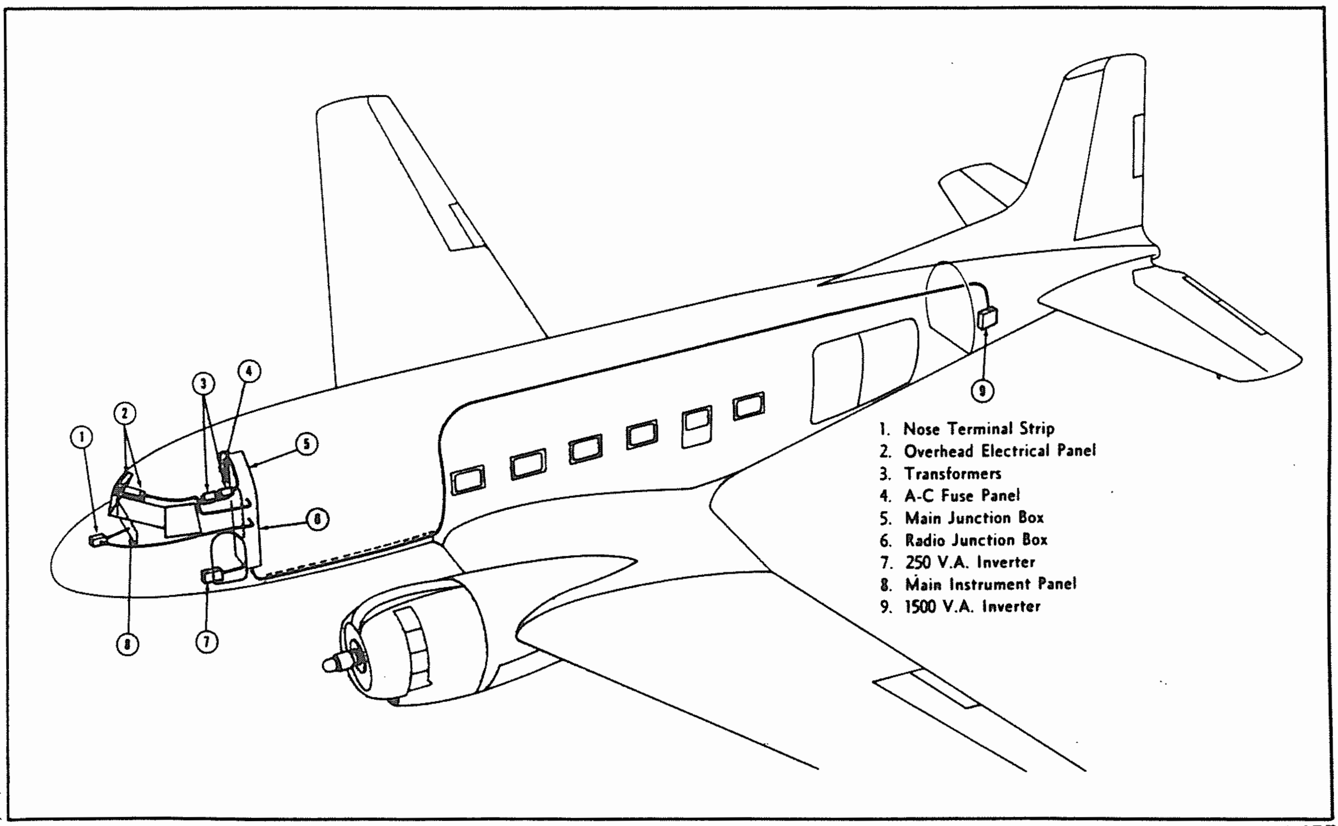

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Figure 7-13. A-C Power System

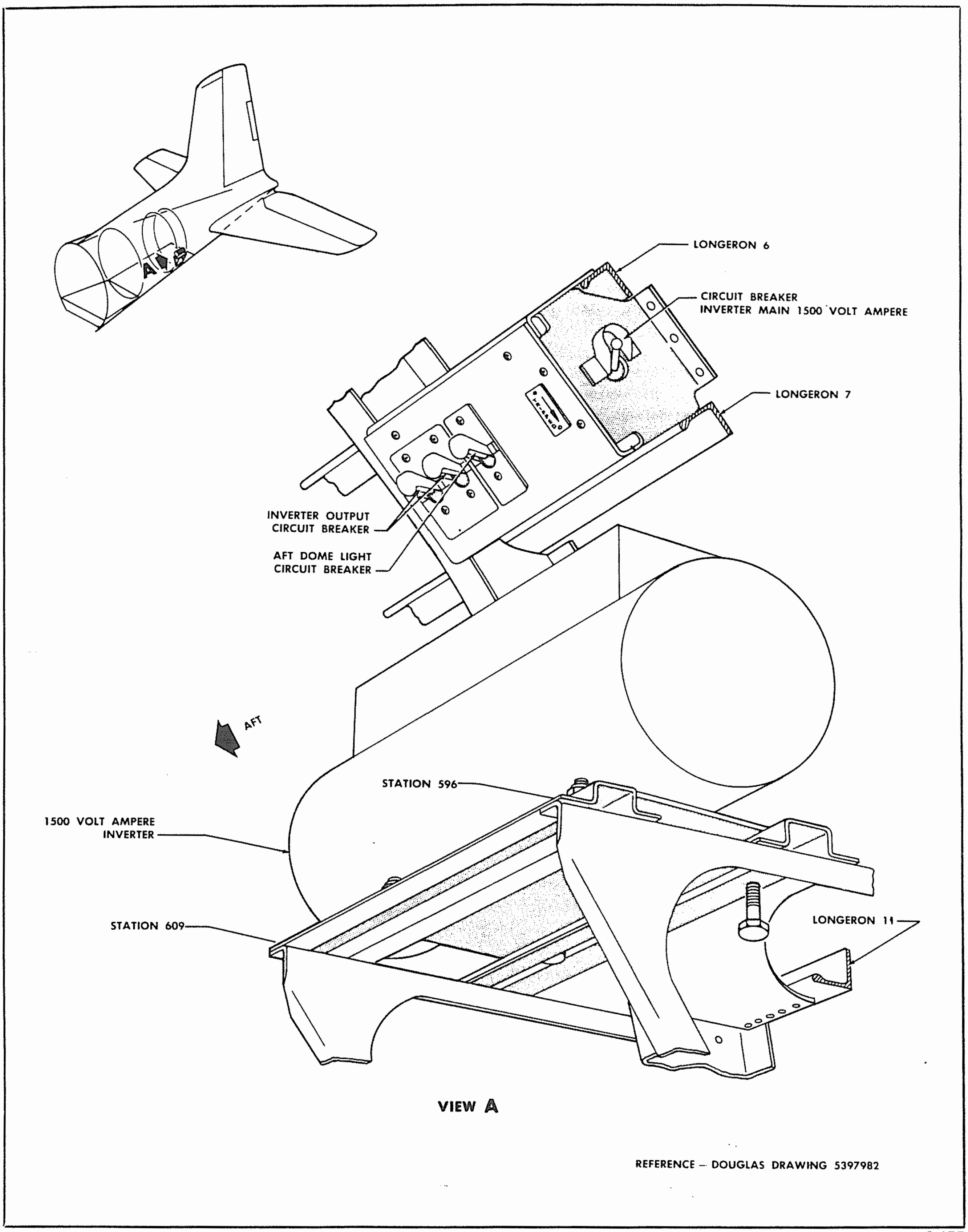

Figure 7-14. 1500-Volt-Ampere Inverter Installation

verter control switch located on the co-pilot's overhead electrical panel. The auxiliary inverter is located below the fuselage floor in the left side of the aircraft between station 47 and station 64. A-c power is supplied by the auxiliary inverter through two 2-ampere fuses located in the a-c fuse panel on the top inboard post of the main junction box. A transformer located in the voltage regulator compartment changes the voltage from 115 volts to 26 volts, 3-phase, 400-cycle, a-c power for instrument loads. Two inverter control switches are located on the co-pilot's overhead electrical panel, one for the main inverter ON and one for the auxiliary inverter ON. With both switches in the OFF position, the red inverter warning lights located on the pilot's and co-pilot's instrument panels should be ON, indicating no a-c power. With both switches in the ON (or up) position the indicator lights should be off and all a-c equipment should be energized. With the main inverter switch ON (or in the up position), and the auxiliary inverter switch in the TRANSFER (or down) position, the inverter warning light will be OFF and the radio equipment, automatic pilot, and all a-c power instruments should be energized. With the auxiliary inverter in the ON (or up) position, and the main inverter switch in the TRANSFER (or down) position, the inverter warning light should be OFF, all a-c radio equipment (except ARN-14 equipment) and the automatic pilot will be inoperative, the pilot's and co-pilot's a-c instruments, and the ARN-14 (VOR) equipment will be energized.

7-55. INVERTERS.
(See paragraph 7-54.)

7-56. MINOR REPAIR AND REPLACEMENT OF INVERTERS.

Note

No adjustment is to be made on the inverters other than the voltage output. To adjust the inverter voltage output, turn the adjusting screw located in the regulation control box on top of the inverter. Check only one inverter at a time.

a. With the d-c bus energized between 26 and 29 volts, and with all the load possible on one of the two inverters, measure the a-c voltage output. The a-c output should be between 110 and 120 volts. Check the warning light for the respective inverter to make certain that it is OFF.

b. Defective inverters are to be replaced with serviceable units.

7-57. REMOVAL OF MAIN 1500 VA INVERTER.
(See figure 7-14.)

a. Trip the 125-ampere circuit breaker located above the inverter.

b. Remove the cover from the terminal housing on the top aft end and forward end of the inverter.

c. Remove the five wires from the inverter terminal studs.

d. Remove the four mounting bolts, washers, and nuts in the base of the inverter.

7-58. INSTALLATION OF MAIN 1500 VA INVERTER. *(See figure 7-14.)* Reverse the removal procedure.

7-59. REMOVAL OF AUXILIARY 250 VA INVERTER.
(See figure 7-15.)

a. Remove the left floor panel between station 47 and the main junction box by removing the attaching screws and washers to gain access to the auxiliary inverter.

b. Remove the electrical disconnect plug from the inboard side of the inverter.

c. Remove the inverter and mounting channel from between lateral floor beams station 47 and station 64 by removing the 2 attach bolts and washers at each end of the channel.

7-60. INSTALLATION OF AUXILIARY 250 VA INVERTER. *(See figure 7-15.)* Reverse the removal procedure.

7-61. INVERTER TRANSFORMERS. Two inverter transformers are installed in the voltage regulator compartment to lower the voltage output of the inverter from 115-volt, 400-cycle to 26-volt, 400-cycle a-c power for the pilot's and co-pilot's horizon and turn-and-bank indicators.

7-62. MINOR REPAIR AND REPLACEMENT OF INVERTER TRANSFORMERS. Defective inverter transformers must be removed and replaced with serviceable units.

7-63. REMOVAL OF INVERTER TRANSFORMERS.

a. Lower the voltage regulator compartment access door to gain access to the two transformers located outboard over the generator overvoltage regulators.

b. Remove the wires from the six terminals on the transformers.

c. Remove the four transformer mounting screws, washers, and nuts.

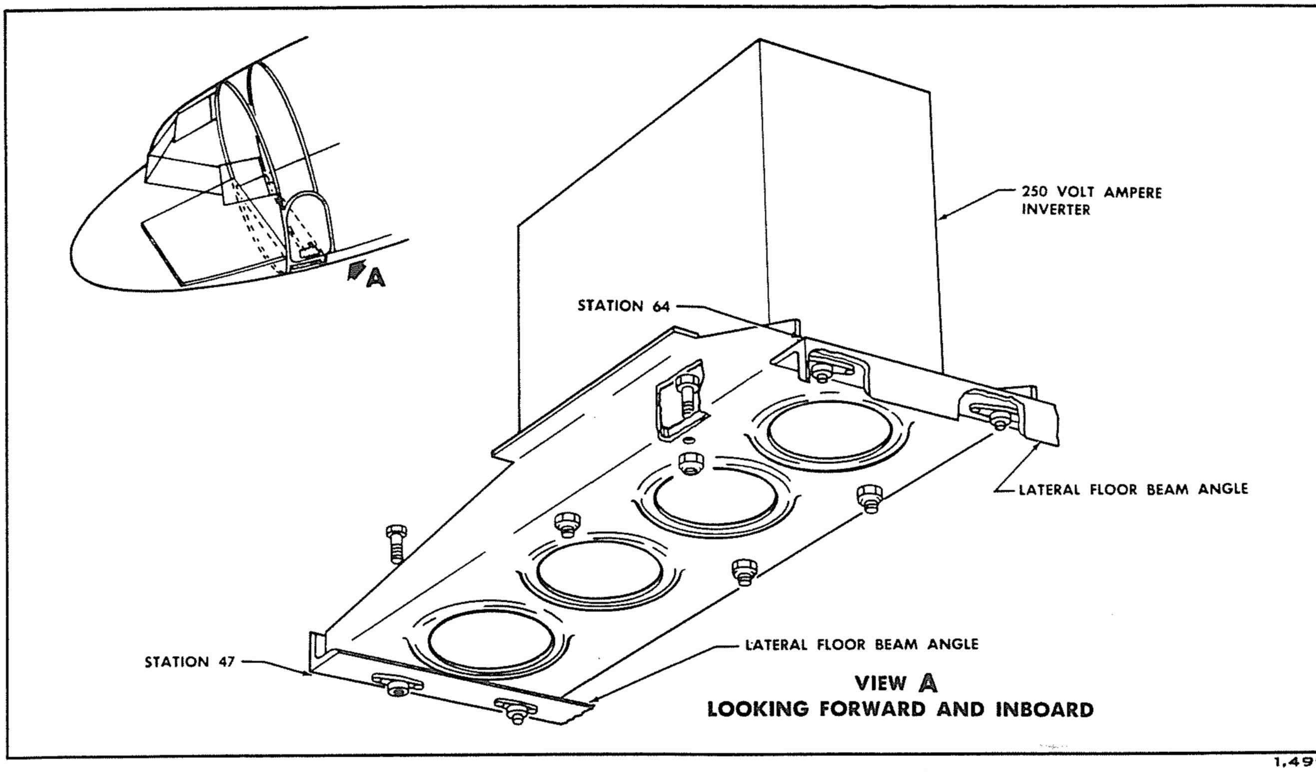

Figure 7-15. 250-Volt-Ampere Inverter Installation

7-64. INSTALLATION OF INVERTER TRANSFORMERS. Reverse the removal procedure.

Note

When installing the forward transformer, terminals 4 and 5 of the transformer must be at the aft end of the transformer. When installing the aft transformer, terminals 4 and 5 of the transformer must be at the forward end of the transformer.

7-65. INVERTER 115-VOLT A-C POWER TRANSFER RELAYS. Two inverter 115-volt a-c power transfer relays are installed in the main junction box and are identified as items R2-16 G-2 compass and R2-22 directional gyro. They transfer the 115-volt a-c, 400-cycle phase A and phase C power from the main inverter system to the auxiliary inverter for the directional Gyrosyn indicator and G-2 compass.

7-66. REMOVAL OF INVERTER 115-VOLT A-C POWER TRANSFER RELAYS.

a. Remove the wires from the eight terminals of the relays.

b. Remove the two screws and washers in the base of the relay.

7-67. INSTALLATION OF INVERTER 115-VOLT A-C POWER TRANSFER RELAYS. Reverse the removal procedure.

7-68. INVERTER 26-VOLT A-C POWER TRANSFER RELAYS. Two 26-volt a-c power transfer relays are installed in the main junction box to transfer the 26-volt 400-cycle a-c power from the main inverter system to the auxiliary inverter system for the pilot's horizon and turn-and-bank indicators, and the co-pilot's horizon indicator. The relays are identified as item R2-23 for the pilot's horizon and turn-and-bank relay, and as item R2-24 for the co-pilot's horizon indicator.

7-69. REMOVAL OF INVERTER 26-VOLT A-C POWER TRANSFER RELAYS.

a. Remove the wires from the eight terminals of the relays (item R2-23 or R2-24).

b. Remove the two screws and washers in the base of the relay and remove the relay.

7-70. INSTALLATION OF INVERTER 26-VOLT A-C POWER TRANSFER RELAYS. Reverse the removal procedure.

7-71. AUXILIARY INVERTER CONTROL RELAY. When the co-pilot's auxiliary inverter control switch

is placed in the ON (or up) position, the control relay is energized in the main junction box to make the circuit for the auxiliary inverter. When the auxiliary inverter control switch is placed in the transfer position, the control relay breaks the circuit to the auxiliary inverter and a-c power is supplied by the main inverter to the equipment that was being powered by the auxiliary inverter.

7-72. REMOVAL OF AUXILIARY INVERTER CONTROL RELAY.

a. Remove the wires from the three terminals of the relay.

b. Remove the two screws and washers in the base of the relay.

7-73. INSTALLATION OF AUXILIARY INVERTER CONTROL RELAY. Reverse the removal procedure.

7-74. A-C GROUND OVERLOAD INTERLOCK RELAY. On aircraft A, B, and 2 through 26, 2 a-c interlock relays are provided and identified as R2-27 and R2-28. On aircraft C, D, 1, and 27 through 96, relay R2-27 is deleted. Since both circuits function alike, the latter system will be discussed.

7-75. The a-c ground overload interlock relay (R2-28) is installed in the main junction box. The relay is actuated by the alighting gear safety switch. When this switch is depressed, the following loads are removed from the inverter system: the automatic pilot, the IFF (APX-6), and LORAN (APN-4). This reduces the d-c requirement of the inverter to a safe value for ground operation of the aircraft's generators.

7-76. GROUND INTERLOCK OVERRIDE SWITCH. The ground interlock override switch is located above and forward of the co-pilot's side window, and is provided with a guard. By use of this switch, the a-c ground overload interlock relay is deactivated when the aircraft is on the ground, permitting the use of the equipment noted in paragraph 7-75, powered by the aircrafts' generators.

WARNING

The total load on either generator must not be permitted to exceed 135 amperes during ground operation.

7-77. REMOVAL OF A-C GROUND OVERLOAD INTERLOCK RELAYS.

a. Remove the wires from the terminals on the relay.

b. Remove the two screws and washers in the base of the relay.

7-78. INSTALLATION OF A-C GROUND OVERLOAD INTERLOCK RELAYS.

a. Attach the relay to the junction box with two screws and washers.

b. Connect the wires to the terminals on the relay.

7-79. A-C POWER WARNING RELAYS. Two a-c power warning relays are installed in the main junction box and are identified as items R2-17. The relays are normally closed relays and are energized to break the 28-volt d-c circuit to the inverter warning lights on the pilot's instrument panel when either of the 2 inverters are operating and 26-volt a-c power is applied to the pilot's horizon indicators. When either of the 2 horizon indicators does not have a-c power applied, the warning relay for the respective instrument will be de-energized and the contacts will close to make the 28-volt d-c circuit to the warning light.

7-80. REMOVAL OF A-C POWER WARNING RELAYS.

a. Remove the wires from the terminals on the relay.

b. Remove the two screws and washers in the base of the relay and remove the relay.

7-81. INSTALLATION OF A-C POWER WARNING RELAYS. Reverse the removal procedure.

7-82. INVERTER WARNING LIGHTS. Two press-to-test inverter warning lights are installed on the pilot's instrument panel to indicate no a-c power to the respective horizon indicators from either the main 1500 va inverter or the auxiliary 250 va inverter. The inverter warning light located above the pilot's airspeed indicator will be ON when the main 1500 va inverter switch is OFF. The inverter warning light located above the co-pilot's gyro horizon indicator will be ON when the auxiliary inverter switch is OFF. When one inverter is operating and the switch for the other inverter is in the transfer position both warning lights will be OFF.

7-83. REMOVAL AND REPLACEMENT OF INVERTER WARNING LAMPS. If the lamp is defective, remove the red jeweled lens from the light assembly and replace the lamp with a new one (AN3121-313).

7-84. CIRCUIT PROTECTION. Circuit protection devices are placed in all circuits and are generally located in the circuit as near to the main distribution bus as is possible. Circuit breakers are used for protection against overloads or shorts for all d-c electrically operated equipment up to 125 amperes. Limiters are used as protection against excessive electrical loads or shorts over 150 amperes. Fuses and circuit breakers are used for protection against overloads or shorts in the a-c electrical circuit. Most of the circuit breakers are installed on a circuit breaker panel aft of station

47, behind and above the pilot's station. Fuses are installed in the fuse compartment near the top inboard side of the main junction box. A 100-ampere limiter is installed in the main junction box as a circuit protection for the radio d-c electrical equipment. A 225-ampere limiter is installed in the main junction box adjacent to the radio circuit 100-ampere limiter to provide circuit protection for the secondary bus. A 150-ampere limiter is installed aft of station 177.5 on left longerons No. 11 and No. 12 adjacent to the cabin window to provide circuit protection for the rear bus. A 150-ampere limiter is installed in both the left and right firewall junction boxes to provide circuit protection for the starter circuit in case of overload or short.

7-85. CIRCUIT BREAKERS. A circuit breaker is a device containing either a solenoid or an electrically heated thermal unit arranged to open switch contacts by the tripping of a spring. Time delay is inherent in the thermal device due to the time required to raise the temperature of the device. Dash pots or similar devices are added to solenoid types. The heater or solenoid is in series with the contacts. The device thus serves to protect the circuit in which it is connected from overload by opening its contacts when the current exceeds a certain amount for a period of time determined from the calibration curve for the unit.

7-86. TESTING CIRCUIT BREAKERS. No installation test of the circuit breakers is required, although a test may be made with the circuit breaker installed if it is suspected that a unit has not been tested, or that the calibration of the circuit breaker has changed due to severe overloads or other causes. To test an installed circuit breaker, proceed as follows:

a. Apply a load equal to 200 per cent of the rating. The breakers must trip automatically within the time limits specified in the table following.

b. All breakers not tripping within the times specified are to be rejected.

c. All tests shall be made with the breaker and surrounding air at 25° (±1°)C or 77° (±1.8°)F.

d. Heinemann circuit breakers are made to three calibration curves requiring different trip-time tests. This is shown by the second dash number of the part number. Thus: 1613-15-1 is a 15-ampere curve 1 circuit breaker and 1613-15-2 is curve 2.

Manufacturer and Type of Circuit Breaker	*Size Amperes*	*Minimum Seconds*	*Maximum Seconds*
All makes and types not otherwise listed	5 thru 35	10	40
All makes and types not otherwise listed	50 and larger	10	75
Heinemann Curve 1	All	10	40
Heinemann Curve 2	All	3	9

7-87. FUSES AND LIMITERS. A fuse or limiter consists of a section of fusible metal designed to melt at a predetermined current overload.

7-88. OVERHEAD ELECTRICAL PANELS. *(See figures 7-16 and 7-17.)*

7-89. PILOT'S OVERHEAD ELECTRICAL PANEL. *(See figure 7-16.)* The pilot's overhead electrical panel is located aft and above the left windshield to provide accessibility to the following electrical units: pilot's switch, radio control, and engine instrument panel light rheostats; magnetic compass light switch, left and right engine cowl flap control switches, cabin heater fire warning light, cabin heater master switch, cabin heater fire detector test button, left and right landing light switches, navigation light switch, wing leading edge light switch, and left engine propeller feathering switch. The panel is made of two separate materials: 0.040 sheet Alclad and 24S-TAL and 0.187 sheet transparent acrylic base. The electrical units are installed on the metal panel and the transparent panel covers the metal panel. Cut-outs are provided in the transparent panel for each of the electrical units. The transparent panel is covered with a layer of white translucent material and then with opaque covering, leaving the white surface exposed where letter marking are required. The opposite surface facing the metal panel is attached to the metal panel with 12 No. 1625C Wemac-type lamp assemblies. Miniature lamps direct the light either through the transparent material and across the instrument or under the white letters identifying the electrical unit.

7-90. CO-PILOT'S OVERHEAD ELECTRICAL PANEL. *(See figure 7-17.)* The co-pilot's overhead electrical panel is located aft and above the right windshield to provide accessibility to the following electrical units: right engine propeller feathering switch, starter safety switch, engine primer solenoid switches, oil dilution selector switches, engine starter switches, left and right engine fuel booster pump switches, aircraft power/ground-power switch, left and right generator control switches, main and auxiliary inverter selector switches, pitot and nose heater airscoop ammeter, nose heater airscoop switch, pitot head heater switch, pitot head heater and nose airscoop heater ammeter selector switch, and left and right generator ammeters. The panel is similar to the pilot's overhead electrical panel, and incorporates the same lighting design for the units on the panel (see paragraph 7-89).

7-91. REMOVAL OF OVERHEAD ELECTRICAL PANELS. Both the metal and transparent panels can be removed as a unit when it becomes necessary to gain access to the inside of the panel assemblies, although if it is necessary to replace a switch or instrument on the panel, removal of the transparent panel is required to gain access to the screws supporting the unit to be replaced.

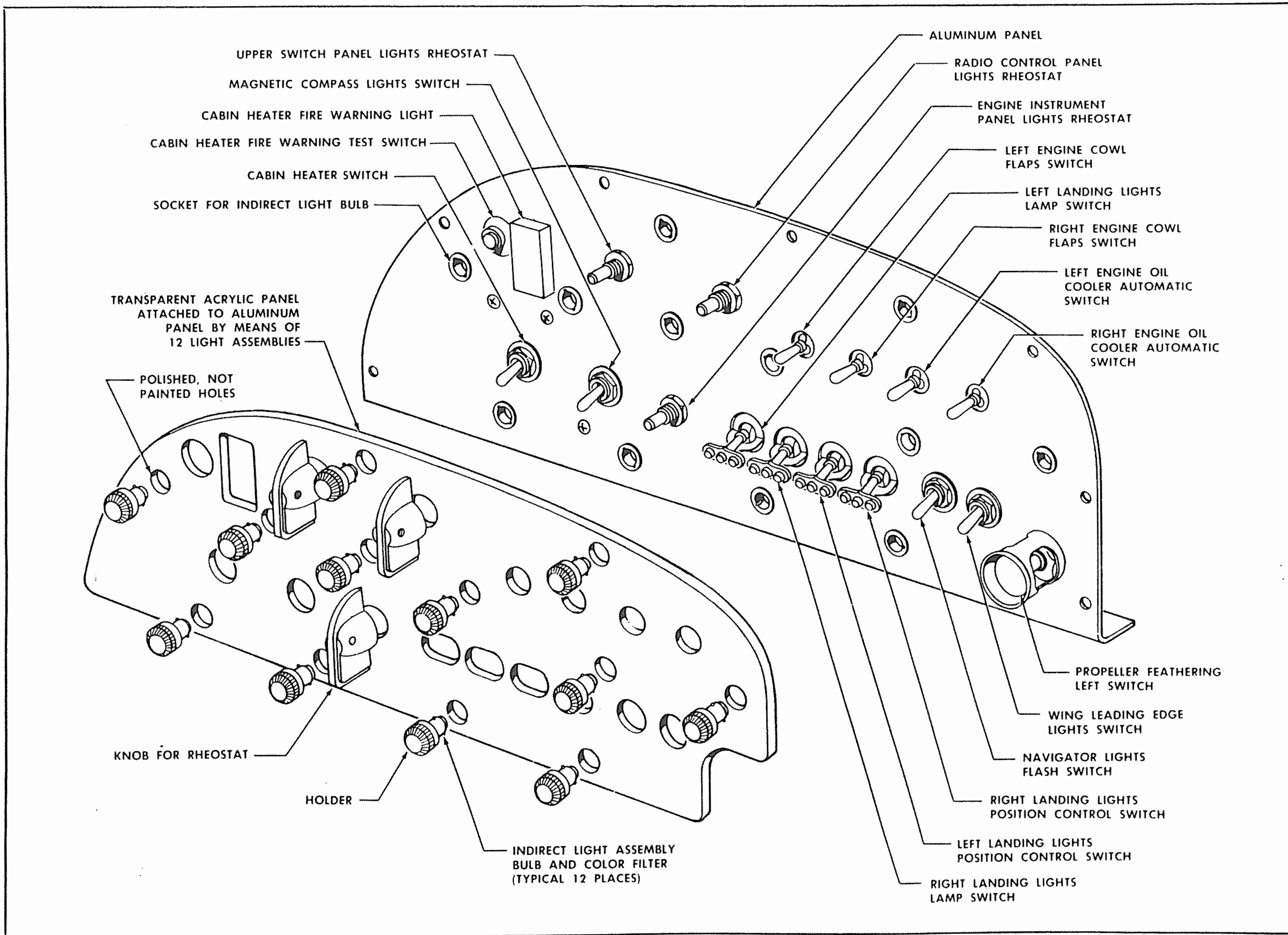

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Figure 7-16. Pilot's Overhead Electrical Panel Assembly

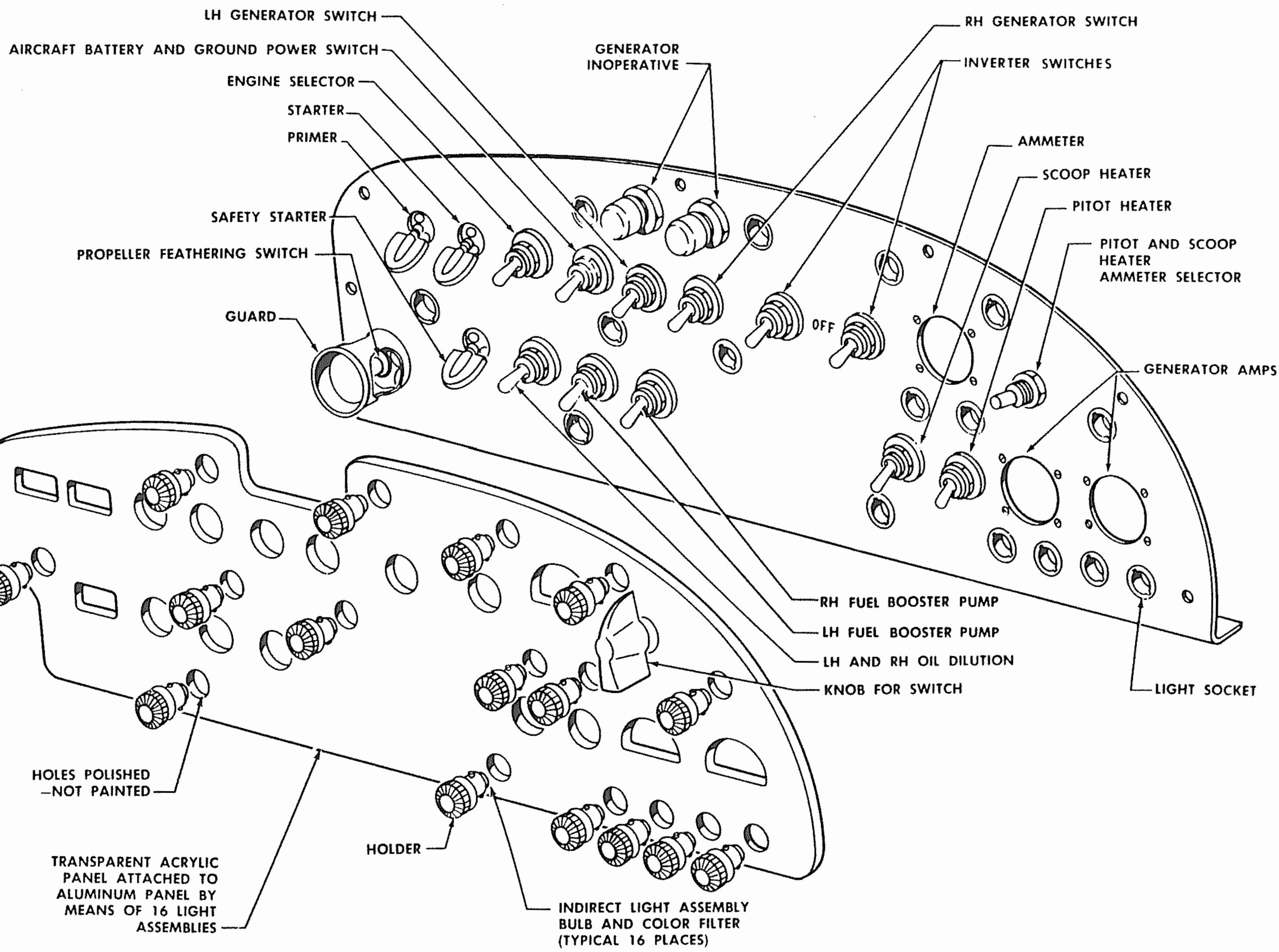


Figure 7-17. Co-Pilot's Overhead Electrical Panel

a. The panel may be removed as a unit by removing the screws around the edge of the panel and allowing the panel to be lowered from the box assembly.

b. To remove the transparent panel, remove the rheostat knobs from the rheostats, and remove the miniature-type lamp sockets that hold the two panels together.

c. To lower the panel from the box, remove the screws around the edge of the metal panel.

7-92. INSTALLATION OF OVERHEAD ELECTRICAL PANELS. Reverse the removal procedure.

CAUTION

Give the entire area a thorough inspection at the conclusion of service operations, checking for safety, security, cleanliness and loose items. When panels are reinstalled, check that all fasteners are secured.

7-93. CIRCUIT BREAKER PANEL. *(See figure 7-18.)* The circuit breaker panel is located immediately forward of the main electrical junction box, flush with the aisle. Protective transparent guards are provided to cover the circuit breaker panel to prevent accidental tripping of any of the breakers. The circuit breakers on the panel are supplied with power through two bus systems. The primary bus is unprotected and feeds the propeller feathering control and fuel booster pump emergency circuits. The secondary bus supplies power to the remainder of the electrical circuit breakers and is protected by a 225-ampere current limiter where it joins the main bus.

7-94. FUSE PANEL. *(See figure 7-19.)* The a-c fuse panel is located on the top inboard side of the main junction box and is protected with a Fiberglas laminated cover.

7-95. JUNCTION BOXES.

7-96. MAIN ELECTRICAL JUNCTION BOX. *(See figure 7-11.)* The upper half of the main junction box is used for the installation of equipment pertaining to the electrical system only. The lower half is the main radio junction box.

7-97. ENGINE SECTION JUNCTION BOXES. One junction box is mounted in each engine accessory section and is provided for the interconnection of the various electrical units of the engine.

7-98. FIREWALL JUNCTION BOXES. *(See figure 7-21.)* One firewall junction box and one reverse-current relay box are mounted on the aft side of each firewall. Each firewall junction box contains a starting relay, a propeller feathering relay, firewall disconnect plugs, a starter feed wire current limiter and the connecting bus bars.

7-99. NACELLE JUNCTION BOXES. *(See figure 7-21.)* Two junction boxes are mounted in each nacelle. One box, mounted on the inboard nacelle wall, contains a terminal strip for the nacelle wiring. The other box, mounted on the outboard nacelle wall, contains the terminal strip for the outer wing wiring disconnect.

7-100. FIREWALL IGNITION PLUGS. A separate plug and receptacle are provided on each firewall for the magneto ignition leads. A grounding receptacle is provided on the engine section junction box to ground the engine magneto when the firewall ignition plug is disconnected from the firewall.

7-101. CENTER WING IGNITION PULL BOX. *(See figure 7-20.)* The center wing ignition pull box is located in the forward part of the center wing section. The box may be reached by removing the fillet on the under side of the wing adjacent to the left wing leading edge.

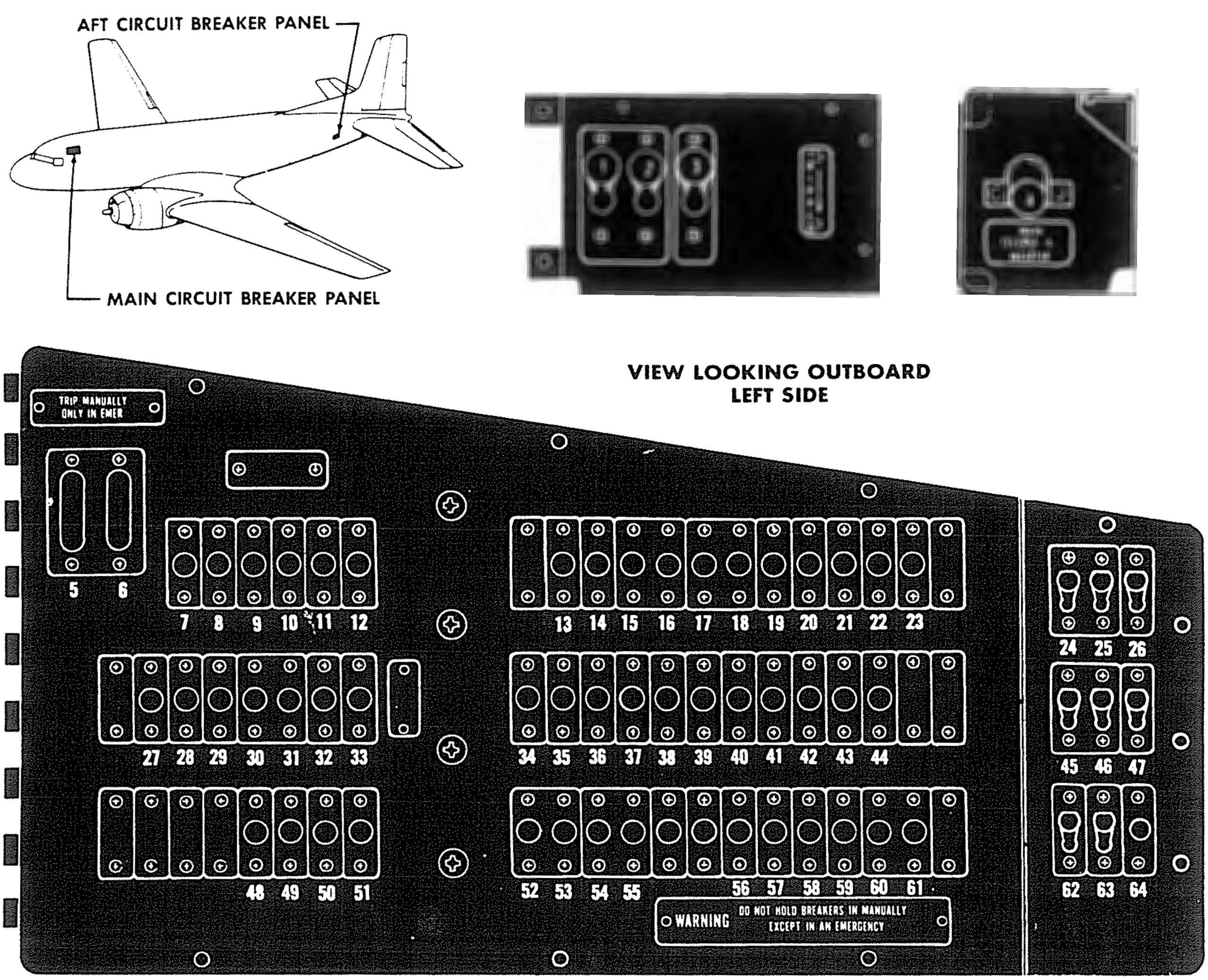
7-102. FUSELAGE ELECTRICAL TUNNEL. *(See figure 7-22.)* The fuselage electrical tunnel is located on the left side of the aircraft and extends from approximately station 40 to the wing leading edge. The tunnel serves as a junction point and wire break point for wiring routed between the fuselage and the wing or nose.

7-103. NOSE TERMINAL STRIPS. *(See figure 7-22.)* The nose terminal strips are located on the left side of the fuselage at station 19. These terminal strips serve as a break point for wiring to the instrument panel.

7-104. ENGINE PRIMER SYSTEM.

7-105. ENGINE PRIMER SOLENOID VALVE. For information on the primer solenoid valve, see paragraph 5-79 and figure 10-15.

7-106. ENGINE PRIMER SWITCHES. A single spring-loaded engine primer switch is located on the right overhead electrical panel for the control of both engine primer solenoid valves. Each of the engine solenoid valves are operated separately by the one engine primer switch through a selector switch.



1. 115 V.A.C.A.—Phase Inverter Output
2. 115 V.A.C.A.—Phase Inverter Output
3. Aft Dome Lights
4. Main Inverter
5. LH Generator Field
6. RH Generator Field
7. Heater Ground Control
8. Heater Control
9. Ground Blower
10. Inverter Warning Lights
11. 1500 V.A. Inverter Control
12. 250 V.A. Inverter
13. Cabin Dome Lights
14. Navigator Lights
15. RH Landing Lights Control
16. LH Landing Lights Control
17. Left Landing Light
18. Right Landing Light
19. Radio Panel Lights
20. Pilot Instrument and Center Instrument Panel Lights
21. Co-Pilot Instrument and Upper Switch Panel Lights
22. Miscellaneous Cockpit Lights
23. Wing Leading Edge Lights
24. LH Propeller Feathering
25. RH Propeller Feathering
26. Cockpit Lights (Turn-and-Bank Indicator)
27. Range Receiver
28. VOR Receiver
29. Glide Slope Indicator
30. Red ADF System
31. Green ADF System
32. Marker Beacon System
33. Altimeter
34. Landing Gear Position Indicator
35. General Warning Lights
36. Door Warning Light
37. Landing Gear Warning Light
38. Fuel and Oil Pressure Warning Light
39. Heater Fire Warning Light
40. Engine Fire Warning Light
41. G-2 Compass
42. Automatic Pilot
43. Temperature Instruments
44. Fuel Quantity Indicator
45. Left Fuel Booster Pump
46. Right Fuel Booster Pump
47. Cockpit Panel Flood Lights
48. IFF Power
49. Pilots HF Receiver
50. VHF Transmitter—Receiver
51. UHF Transmitter—Receiver
52. Left Oil Cooler Doors
53. Right Oil Cooler Doors
54. Left Cowl Flaps
55. Right Cowl Flaps
56. Engine Starter and Ignition
57. Alcohol Quantity Indicator
58. De-Icer Equipment
59. Airscoop Heater
60. Left Pitot Heater
61. Right Pitot Heater
62. D-C Turn-and-Bank Indicator
63. Cockpit C-4 Lights
64. Main J Box Flood Light

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Figure 7-18. Circuit Breaker Panel

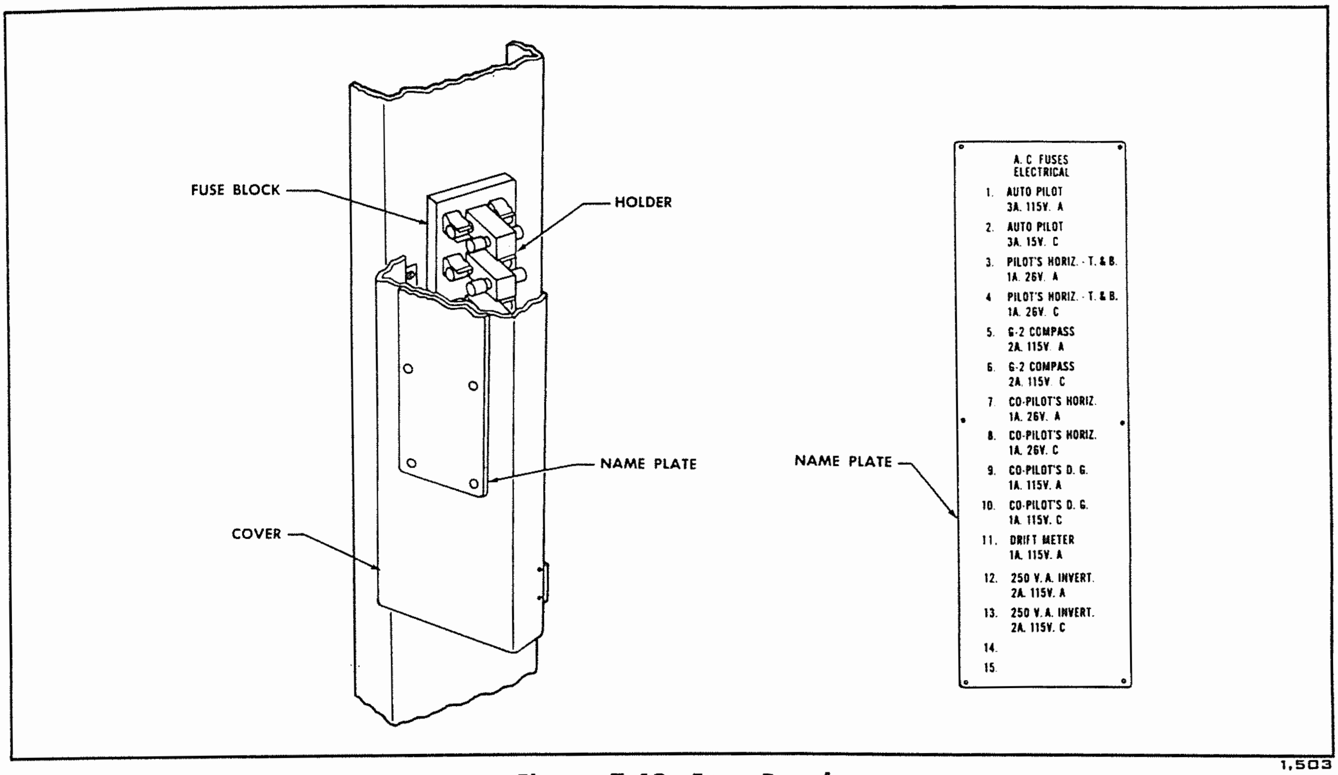


Figure 7-19. Fuse Panel

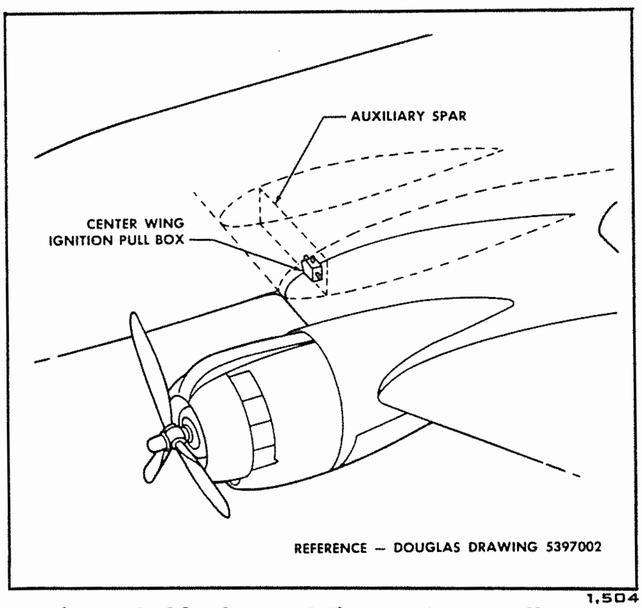


Figure 7-20. Center Wing Ignition Pull Box

7-107. ENGINE OIL DILUTION SYSTEM. One single-pole, double-throw switch is installed on the co-polit's overhead electrical panel to open either the left or right oil dilution solenoid valve located on the forward side of each firewall. The switch is a momentary-contact-type switch. D-c power is supplied for the oil dilution system through a 10-ampere circuit breaker located on the circuit breaker panel. For information on the oil dilution valve, see paragraph 5-145.

7-108. PROPELLER FEATHERING SYSTEM. One propeller feathering pump, driven by a 24- to 28-volt d-c electric motor, is installed on the outboard side of each nacelle aft of the firewall. Two propeller feathering switches are installed, one on each overhead electrical panel, for the operation of the pumps. The switches operate the pumps through relays mounted in the firewall junction boxes. A pressure cut-out switch, mounted on the base of each propeller governor, opens the propeller feathering circuit when the pressure in the propeller system reaches 400 pounds. For further information on the propeller feathering system, see paragraph 5-214 and figure 10-38.

7-109. FUEL BOOSTER PUMPS. Two fuel booster pumps are installed in the wing center section aft of station 191. The booster pumps are operated by 24- to 28-volt d-c motors. Operation of the fuel booster pumps is controlled by two switches located on the co-pilot's overhead panel. The switches are wired to electrical power through 15-ampere circuit breakers located on the circuit breaker panel. For additional information see paragraph 5-71 and figure 10-18.

7-110. AIRCRAFT LIGHTS. *(See figure 7-23.)* The aircraft lights, both inside the aircraft for the use of the crew members and passengers and outside the aircraft for navigation assistance, are described in the following paragraphs. All lamps on the aircraft are electrically operated on the 24 or 28-volt d-c system. A spare lamp box is installed on the forward face of the right bulkhead aft of the co-pilot's seat and con-

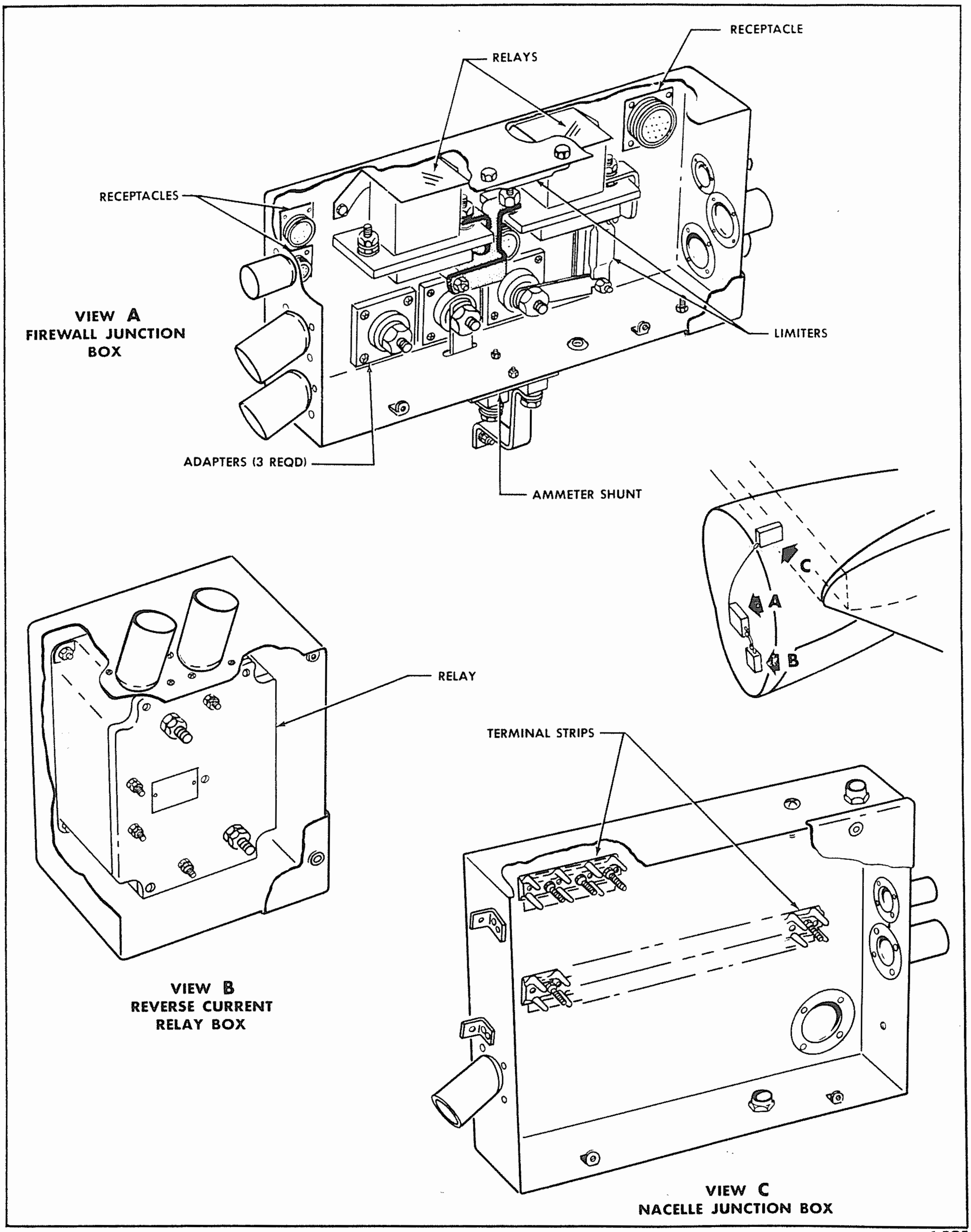


Figure 7-21. Firewall and Nacelle Junction Boxes and Reverse-Current Relay Boxes

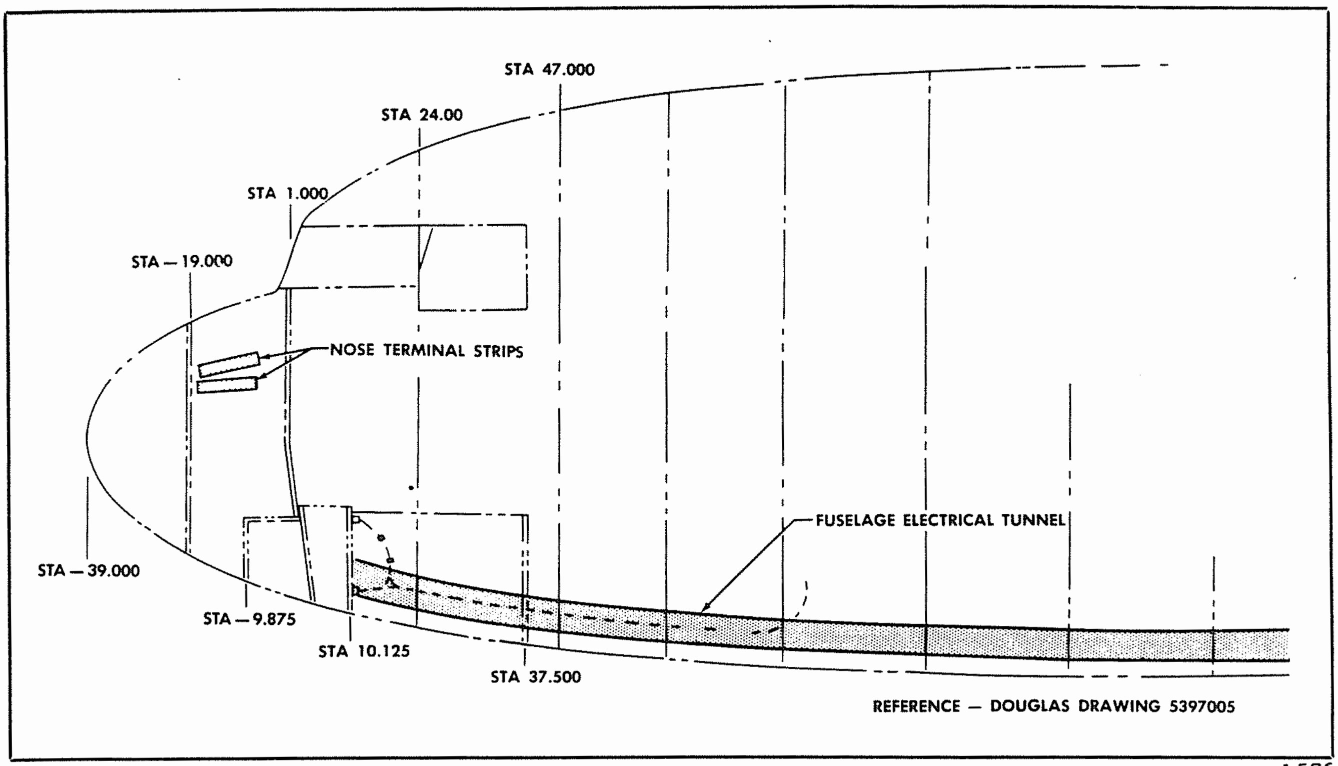


Figure 7-22. Fuselage Electrical Tunnel and Nose Terminal Strip

tains the necessary spare lamps for the flight compartment. One 20-watt lamp is installed, above and forward of the main junction box and across the aisle from the circuit breaker panel, in adjustable-type light assembly. The lamp is controlled by a 5-ampere circuit breaker located on the lower forward corner of the circuit breaker panel. See figures 7-23 and 10-24 for lamp location in the aircraft and to the respective wiring diagrams in Section X.

7-111. DOME LIGHTS. The four main cabin dome lights are controlled by a rheostat located on the ceiling at the forward end of the cabin. One dome light is installed in the top center of the fuselage near the main cabin door between stations 465 and 479, and is controlled by a rheostat located forward of the light assembly. One light assembly is installed in the lavatory compartment and one on the top of the tail section compartment. Both light assemblies are controlled by AN3022-2 switches located adjacent to the light assemblies. The entrance way, tail section, and lavatory lamps are electrically powered through a 5-ampere circuit breaker located in the tail section at station 596 above the 1500 va inverter. For dome light locations see figures 7-23 and 10-23.

7-112. REMOVAL OF DOME LIGHT ASSEMBLIES AND LAMPS.
(See figure 7-23.)

a. Remove the three screws in the rim of the light assembly and lower the rim with the lens.

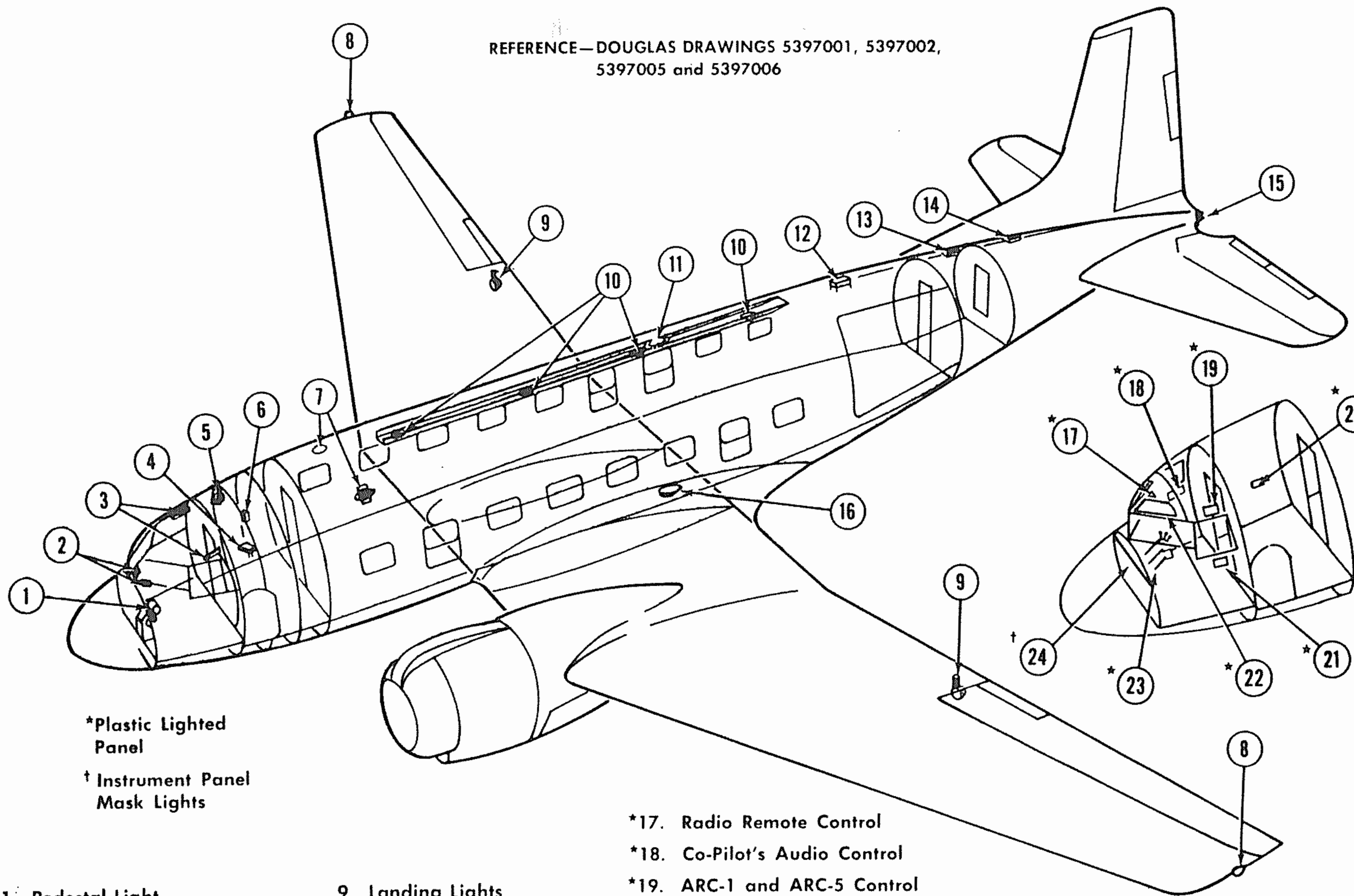
b. Remove the lamp by pressing the lamp in with a slight pressure and then turning the lamp counterclockwise approximately 1/8 of a turn to eject the lamp from the socket.

c. To remove the light assembly, remove the three additional screws from the rim of the reflector and disconnect the wires from the terminals.

7-113. INSTALLATION OF DOME LIGHT ASSEMBLIES AND LAMPS. Reverse the removal procedure.

7-114. FLIGHT COMPARTMENT LIGHTS. Two type C4-A extension lights are installed in the top of the flight compartment above the pilot's and co-pilot's seats. The type C4-A lights are controlled by a rheostat-type switch located on the aft end of the light assemblies. The size of the spot can be adjusted by a knob located on the side of the light assemblies forward or aft. A removable red lens is installed on the forward end of the light assemblies and can be removed from the light assemblies by depressing the two small catches on the outer rim of the red lens. Two AN3047-3A work table light assemblies are installed at the radio operator's table and the navigator's table. Each of the two light assemblies are controlled by rheostat-type switches located adjacent to the base of the light assemblies. In addition to the four flight compartment lights, a receptacle is installed in the top of the inboard side of the main junction box for the periscopic sextant. The 4 flight compartment lights and the periscopic sextant receptacle are electrically powered through a 10-ampere circuit breaker located

REFERENCE—DOUGLAS DRAWINGS 5397001, 5397002, 5397005 and 5397006



*Plastic Lighted Panel

† Instrument Panel Mask Lights

1. Pedestal Light
2. Instrument Panel Floodlights
3. Cockpit Lights (C-4)
4. Navigator's Table Light
5. Main Junction Box Floodlight
6. Radio Operator's Light
7. Wing Illuminating Lights
8. Wing Tip Position Lights
9. Landing Lights
10. Main Cabin Dome Lights
11. Upper Position Light
12. Entrance Dome Light
13. Lavatory Dome Light
14. Tail Access Dome Light
15. Tail Position Lights
16. Lower Position Light

*17. Radio Remote Control
*18. Co-Pilot's Audio Control
*19. ARC-1 and ARC-5 Control
*20. Radio Operator's Audio Control
*21. Pilot's Audio Control
*22. Overhead Electrical Control Panels
*23. Automatic Pilot Control (Aircraft C,D,9 and Subsequent)
†24. Main Instrument Panel

Figure 7-23. Location of Aircraft Lights

on the circuit breaker panel which is identified MISC. COCKPIT LIGHTS *(see figures 7-18 and 10-25).*

7-115. INSTRUMENT PANEL AND PEDESTAL FLOOD LIGHTS. Two red pilot's instrument panel flood lights are installed above the pilot's left and right instrument panels and one aft pedestal red light on the aft side of the pedestal. The pilot's left instrument panel flood light and pedestal aft flood light are controlled by a rheostat-type switch located on the side of the fuselage below the pilot's side window. The co-pilot's right panel flood light is controlled by the rheostat-type switch located on the side of the fuselage below the co-pilot's side window *(see figures 7-23 and 10-28).*

7-116. INSTRUMENT AND OVERHEAD PANEL SWITCH LIGHTS AND COMPASS LIGHT. The instruments on the pilot's, co-pilot's and center instrument panels are individually illuminated by special lighting fixtures, each of which contain two or more lamps. The small lamps are installed in a hood on the top of the fixture and are accessible for replacement by pulling the top hood away from the panel. Both inverter warning, heater inoperative warning, gear unsafe warning, fuel and oil pressure warning, and low limit warning lights on the instrument panels are AN3157-6 light assemblies. The pilot's and co-pilot's overhead electrical panels are illuminated through a transparent sheet over the face of the panels. For further information on the overhead panels, see paragraph 7-88.

7-117. NAVIGATION POSITION LIGHTS. One position light is installed in each wing tip *(see figure 7-23).* The lens for the left wing tip is red and the lens for the right wing tip is green. The lighting arrangement installed in the tail cone is comprised of two light assemblies, one with a red lens and one with a white lens. The white and red light assemblies are installed adjacent to each other in the end of the tail cone. One clear light assembly is installed in the top of the fuselage between stations 333 and 351 and 1 in the bottom of the fuselage, at station 302. A navigation light flasher is installed in the radio rack to flash all navigation lights on and off. A three-position switch (AN3023-1) on the pilot's overhead electrical panel is used to select the type of use required for the navigation lights. When the switch is placed in the steady position only the red and green wing tip lights and the white tail light will be illuminated. With the switch in FLASHER position, the top and bottom lights on the fuselage and the red tail light will flash alternately with the left and right wing tip and white tail light. Electrical power for the navigation lights is supplied through a 10-ampere circuit breaker identified as NAVIGATION LTS. on the circuit breaker panel *(see figures 7-18 and 10-31).*

7-118. LANDING LIGHTS. *(See figure 7-24.)* An electrically retractable landing light assembly is installed on the under side of both left and right outer wings forward of the trailing edge of the wings outboard of station 107. The landing light housing contains a gear mechanism, a motor, a motor limit switch, and a light switch. The lamps are the 28-volt, 600-watt, seal-beam type. Each landing light assembly is controlled by two single-pole toggle switches, located on the pilot's left overhead electrical panel. Two of the switches have three positions, EXTEND, OFF, and RETRACT. The other two switches for the landing light circuit located on the pilot's overhead electrical panel open and close the circuit to the lamp of each light assembly by means of a relay for each lamp circuit located in the main junction box. A small motor in the landing light assemblies drives the extension and retraction mechanism. Two limit switches are incorporated in each lamp assembly. The limit switch for the full-retract position is a fixed-position limit switch. The limit switch for the extended position is adjustable, and the landing lights are to be preadjusted prior to installation to 86 degrees from the full RETRACT position. The 86-degree adjustment will provide a center forward beam of 400 feet when the aircraft is in the normal landing attitude of approximately 6 degrees tail down. No adjustment is provided for lateral alignment of the beam, which must be obtained by accurate installation of the landing light assemblies. The filament shield on each landing light must be installed toward the pilot to eliminate glare. There are two field windings in the light unit drive motor which make reversing of the lamp assembly possible for UP or DOWN travel. When the landing light POSITION CONTROL switch is placed in the OFF position, the landing light will stop and remain in the position it was in when circuit was opened by the switch.

Note

Each landing light should be tested separately and the lamps should not be allowed to burn more than 15 minutes at a time while the aircraft is on the ground, as the absence of a cooling airstream will result in a dangerous overheat condition with resultant failure of the light. Disconnect the circuit to the lamps when alignment inspection is being made.

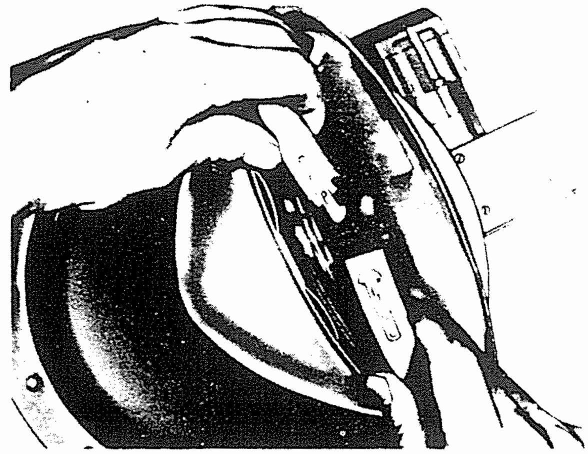
7-119. MINOR REPAIR AND REPLACEMENT OF LANDING LIGHT ASSEMBLY. *(See figure 7-24.)* Replace burned out lamp bulbs as follows:

a. Extend the landing light out two-thirds of the way.

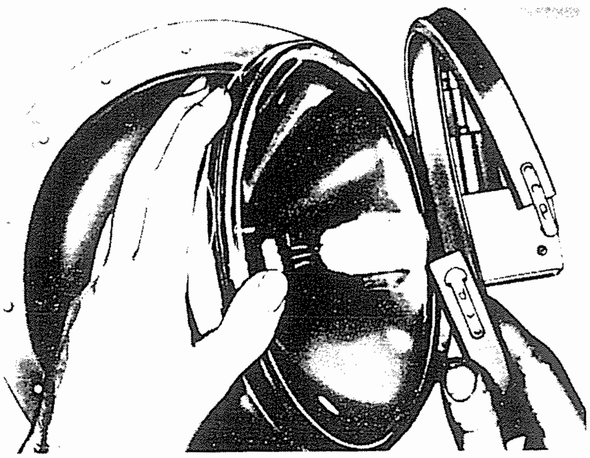
b. Lift the lamp retainer ring clip with a sharp-pointed screw driver, open the ring, and remove the ring clip or push it back over the lamp bulb and inner canopy far enough to allow the lamp bulb to come free of the canopy.

c. Lower the lamp bulb enough to gain access to the terminals and remove the wires from the terminals.

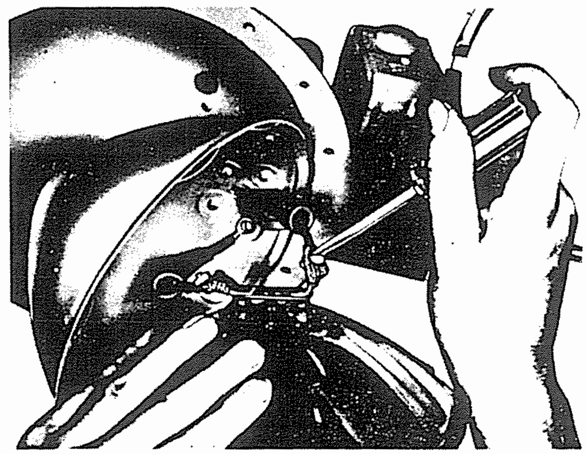
A — DISENGAGE LAMP RING CLIP



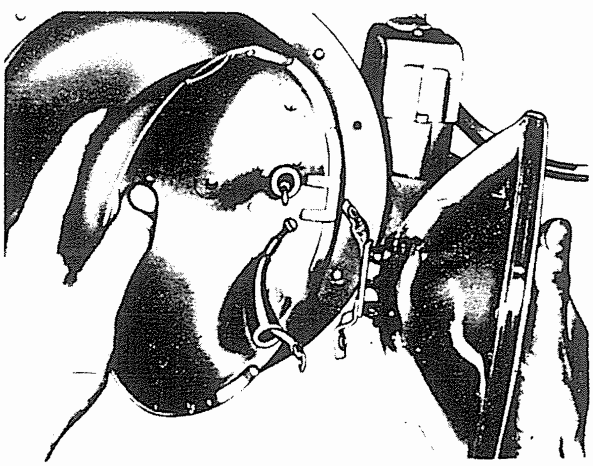
B — SPREAD LAMP RING



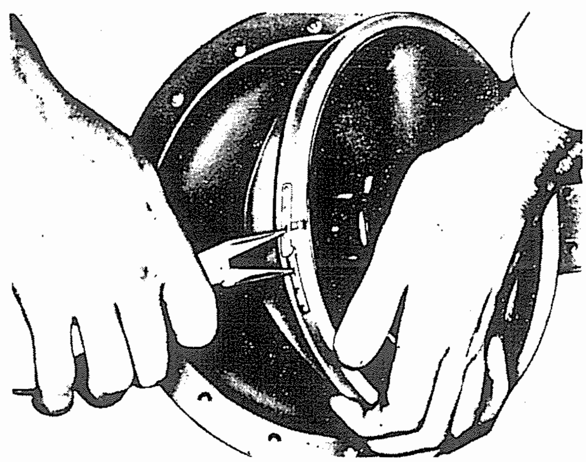
C — REMOVE LAMP RING



D — REMOVE LAMP TERMINAL SCREWS



E — REPLACE BURNED OUT LAMP



F — ENGAGE LAMP RING CLIP

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Figure 7-24. Landing Light Bulb Replacement

d. Connect the wires on the terminals of the replacement lamp.

e. Insert the lamp bulb into the inner canopy with the locating lugs on the lamp bulb properly seated in the slots provided for the lugs in the canopy.

Note

When installing a lamp bulb, the filament shield of the lamp bulb should be inboard toward the pilot.

f. Place the lamp bulb retainer ring on the inner canopy with the beveled edge inside toward the apex of the inner canopy.

g. Clamp the retainer ring in place and engage the clip. A pair of small nose pliers will aid in drawing the clip together and latching it in place.

7-120. REMOVAL OF LANDING LIGHT ASSEMBLY.

a. Remove the 15 screws that attach the outer canopy of the landing light assembly to the wing panels.

b. Remove the electrical wires.

c. Remove the light assembly.

7-121. RETRACTION ADJUSTMENT BEFORE INSTALLATION OF LANDING LIGHT ASSEMBLY. If the landing lights fail to retract properly, the following adjustments should be made before installing the light assembly:

a. Remove the four screws that hold the drive housing cover in place and lift off the cover to expose the limit switches which govern the extension and retraction of the landing light assembly.

b. Provide a source of power (24 to 28 volts) to operate the lamp during adjustment.

c. Adjust the points to a $1/16$-inch gap when the inner canopy is fully retracted. The retract points should be set to break just before the inner canopy reaches the flush position, because the high rpm of the driving motor causes a slight override, or coasting, after the contact points break.

Note

The retract points can be adjusted by either bending the strap spring or by slightly bending the point arm. Straightening the contact point arm will run the inner canopy deeper into the outer canopy. Bending the contact point arm inward toward the sliding contact will break the circuit earlier to restrict the degree of recessing of the inner canopy. Before the light is operated, check the spring tension of the contact point arm to determine that the points are making a good contact.

d. After adjustment, operate the landing light assembly very carefully the first few times. Operate the inner canopy from approximately a half-open position to within a half inch of the full flush position. If the inner canopy does not override the flush position, repeat the operation, successively running the light out one inch, then four inches, and then full travel.

7-122. EXTENSION ADJUSTMENT BEFORE INSTALLATION OF LANDING LIGHT ASSEMBLY. If the landing light assembly does not extend correctly, adjust the open points. Disconnect the landing light from the power supply for this adjustment.

a. Loosen the locking screw until the contact assembly will slide in the rack. If a greater opening is desired, move the contact assembly toward the canopy and away from the retract contact points. If less opening is desired, move the contact assembly toward the retract points.

b. Tighten the locking screw while holding the contact points in the desired position.

c. Follow the same procedure to check the operation of the open points as was given for the check-out of the retract points.

d. After adjustment of the contact open points blow all foreign material from the housing, replace the cover on the drive housing and install the four screws.

7-123. INSTALLATION OF LANDING LIGHT ASSEMBLY. Reverse the removal procedure.

7-124. WING LEADING EDGE ILLUMINATION LIGHTS. *(See figure 7-25.)* Two sealed-beam lamps are installed behind flush transparent-plastic windows between station 117 and station 136, near the top of each side of the fuselage, to illuminate the wing leading edges. The circuit for the lights consists of a single-pole, double throw, (AN3022-2) switch installed on the pilot's left overhead electrical panel, and a 10-ampere circuit breaker located on the circuit breaker panel.

7-125. REMOVAL OF WING LEADING EDGE ILLUMINATION LIGHT. *(See figure 7-25.)* Replace burned out lamp bulbs as follows:

a. Remove the six attaching screws around the edge of the transparent plastic window and attach ring.

b. Remove the screw, washer, and nut from the top and bottom of the lamp clamp and mounting clip.

c. Disconnect the wires from the terminals on the lamp *(see figure 10-32).*

d. Remove and replace burned out lamps by means of the four screws in the lamp clamp and mounting bracket.

7-126. INSTALLATION OF WING LEADING EDGE ILLUMINATION LIGHT. Reverse the removal procedure.

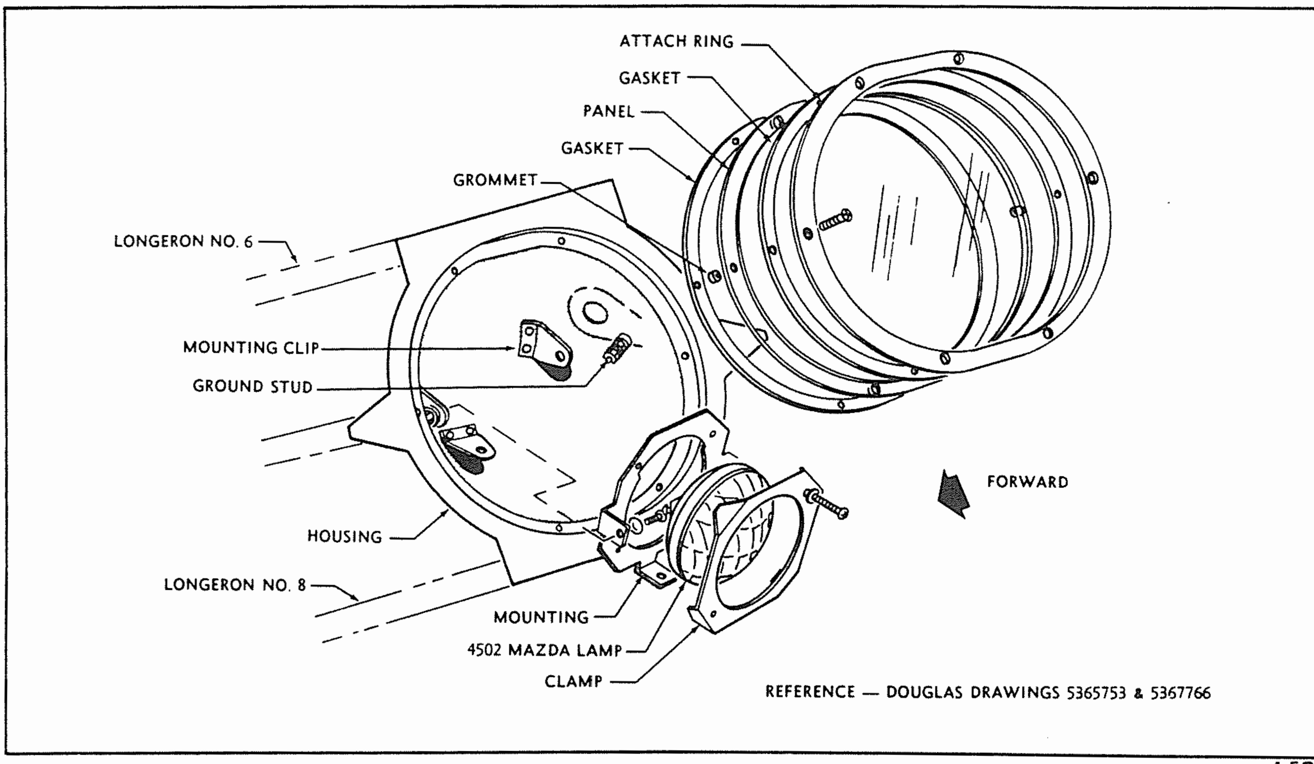

Figure 7-25. Wing Leading Edge Illumination Light Assembly

Note

Prior to installing the transparent plastic window, adjust the lamp with the top and bottom screw in the mounting bracket to direct the center of the lamp beam just over the top of the nacelle and along the leading edge of the wing.

7-127. COWL FLAP DOOR ACTUATOR MOTOR. An electrical positioning and operating mechanism is provided for the cowl flaps. The power unit is composed of an electric motor, gear box, and limit switches. Each cowl flap actuator is controlled by a four-position switch mounted on the left overhead electrical panel. In the EXTEND, CLIMB or RETRACT positions, the actuator will move the door until stopped by the particular limit switch. If the switch is moved to the OFF position, the actuator motor will stop without coasting. For additional information on the cowl flap actuator motor, see paragraph 5-222 and figure 10-13.

7-128. OIL COOLER AIR EXIT DOOR ACTUATOR MOTOR. An automatic control system is provided to maintain the temperature of the engine oil within a range of 170°F to 180°F. A temperature-sensing device is incorporated in the oil return pipe to the tank. It operates to increase or decrease the airflow over the oil cooler as required. The airflow at the oil cooler is controlled by a linear-type reversible actuator. Two switches mounted on the left overhead electrical panel enable the pilot to select any of the following oil cooler door conditions for each engine: OPEN, CLOSED, AUTOMATIC or OFF. For additional information on the oil cooler air exit door actuator motor, see paragraph 5-131 and figure 10-33.

7-129. ANTI-ICING AND DE-ICING SYSTEMS ELECTRICAL EQUIPMENT.

7-130. PROPELLER AND CARBURETOR ANTI-ICING PUMP MOTOR. The propeller and carburetor anti-icing pump motor is located at the alcohol supply tank, mounted under the cabin floor on the right side aft of the trailing edge of the wing. The motor is operated by a switch on the ant-icing control panel, to the right of the co-pilot's seat. For further information on the propeller and carburetor anti-icing pump motor, see paragraph 4-112 and figure 10-5.

7-131. SURFACE DE-ICER DISTRIBUTOR VALVE MOTOR. The distributor valve incorporates a shunt-wound motor and is connected to the motor drive shaft. The operation of the motor is controlled by the cable-connected ON-OFF control located on the right side of the control pedestal. For further information on the surface de-icer distributor valve motor, see paragraph 4-124 and figure 10-5.

7-132. PITOT TUBE AND AIRSCOOP HEATERS. Two electrically heated pitot tubes (AN5813-1) are installed in the nose of the aircraft at station —39. The pitot heaters are electrically controlled by a double-pole switch (AN3023-2) located on the co-pilot's overhead electrical panel. Cabin heater airscoop

de-icing is accomplished by a Calrod unit built into the airscoop, located on the nose section of the aircraft, which is electrically controlled by a single-pole, double-throw switch (AN3022-2) located on the co-pilot's overhead electrical panel. Current flow to each of the two pitot heaters and the airscoop heater is indicated on an ammeter located on the co-pilot's overhead electrical panel. An ammeter selector switch is installed adjacent to the ammeter to obtain individual heater current drain readings. For further information on the pitot tube and airscoop heaters, see paragraph 4-112 and figure 10-4.

7-133. CABIN HEATER SYSTEM ELECTRICAL COMPONENTS. The cabin heater output is controlled by cycling the fuel flow to the heater. A cycling switch operates two fuel solenoids electrically connected in parallel and contained in the heater pump and cycling solenoid assembly. Control of the heater cycling temperature is provided by a Fenwal thermoswitch. The normal operation of the heater is provided by a switch set at 54°C (265°F) and identified by a blue and white color code above the flange of the thermoswitch. If the normal switch fails to operate at the proper temperature, the drop-out switch, which is set at 148.9°C (300°F) and identified by a green and white color code above the flange of the thermoswitch, will close and blow the drop-out fuse, thus rendering the heater inoperative. For additional information on the cabin heater system, see paragraph 4-1 and figure 10-9.

7-134. CABIN HEATER PUMP AND CYCLING SOLENOID ASSEMBLY. The assembly is located on the aft, right side of the heater compartment. It contains a fuel filter, a fuel pressure regulator, a fuel pump, and two fuel cycling solenoid shutoff valves.

7-135. CABIN HEATER IGNITION UNIT. The ignition unit is located aft of and below the heater. It supplies high voltage to the spark plug in the heater. The unit consists of a coil, vibrator, resistor, condenser, radio noise filter, and a single-pole, double-throw switch.

7-136. CABIN HEATER GROUND BLOWER. The cabin heater ground blower is located in the ground blower ducting, forward of the battery compartment. A flapper valve is installed at the fuselage skin line to prevent reverse flow through the blower and the short air intake duct.

7-137. ELECTRICAL WARNING SYSTEMS. For information on the electrical components of the warning systems, see paragraph 6-44 and the appropriate wiring diagrams in Section X.

7-138. WARNING LIGHT DIMMING CIRCUIT. A warning light dimming relay is installed near the bottom inboard side of the main electrical junction box. Provisions have been made in the inboard post of the main junction box to insert a screw driver to manually adjust the relay in the event of a failure of the automatic feature. The dimming relay is operated by means of a switch which is integral with the pilot's instrument light rheostat located below the pilot's side window *(see figures 10-58, 10-59, and 10-60).*

Note

All warning lights are dimmed, except when the instrument light rheostat is in the OFF position.

AN 01-40NK-2

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z AIRCRAFT

SECTION VIII
RADIO AND RADAR

THIS SECTION SUPERSEDES SECTION VIII OF AN 01-40NK-2
DATED 15 MAY 1952 REVISED 1 NOVEMBER 1952

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15 April 1953

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SECTION VIII

RADIO AND RADAR

TABLE OF CONTENTS

SECTION VIII

RADIO AND RADAR

8-1. ELECTRONIC SYSTEM.

8-2. DESCRIPTION. *(See figure 8-1.)* Electronic equipment is installed in the aircraft to provide a means for communication, to aid in navigation, and to assist in guiding the aircraft during blind landing operations. Power for operation of the electronic equipment is provided by the 28-volt d-c battery system and by the 115-volt a-c inverter system *(see figure 8-2).* Remote radio controls are grouped on the pilots' overhead radio control panel with corresponding indicators and indicator lights. Audio selector panels are located below the pilot's and co-pilot's side sliding windows and on the right side of the fuselage at the radio operator's station. An interphone control box is located above the flight compartment entry door, aft of station 47, for the navigator *(see figure 8-3).* Antennas for the various radio units are attached to the fuselage exterior, as shown in figure 8-4. The location of the following radio equipment is shown in figure 8-1:

HF Communication Equipment
VHF Transmitter-Receiver
UHF Transmitter-Receiver
Interphone and Radio Control
Range Receiver
Marker Beacon Receiver
Visual Omni-Range (VOR) Navigation System (Complete Provisions)
Glide Path Receiver (Antenna and Coaxial Lead-In Only)
Identification Friend or Foe (IFF) Equipment (Complete Provisions)
Long-Range Navigation (LORAN) Equipment
Radio Compasses (Red and Green)

Note

Detailed information concerning each of the radio units is provided in the specific handbooks listed in the current issue of the *Naval Aeronautics Publications Index* or in the *AF Index of Technical Publications and Confidential Orders.*

8-3. RADIO RACKS. Two radio racks are installed in the aircraft. The main radio rack is located aft of the main junction box on the left side of the aircraft. Additional radio equipment is installed above the radio operator's table, forward of station 97 on the right side of the aircraft *(see figure 8-5).* Two radio shelves are installed in the tail section of the aircraft for the radio altimeter receiver-transmitter and the IFF receiver-transmitter. These units are located on the right side of the tail section between stations 568 and 596 *(see figure 8-1).*

8-4. RADIO CONTROL PANELS.
(See figures 8-6, 8-7, and 8-8.)

8-5. PILOTS' RADIO CONTROL PANEL. A remote control panel for the radio equipment, for operation by the pilot or co-pilot, is installed above and aft of the pilot's and co-pilot's overhead electrical panels. The controls on the pilot's radio control panel are used to operate red or green radio compasses, VOR navigation radio, UHF command radio, and HF and VHF communication radio. The radio control panel is edge-lighted. Light from small lamps is directed through the edge of the transparent panel and across the face of the indicators, and also under the white lettering on the face of the panel.

8-6. RANGE RECEIVER AND VHF REMOTE CONTROL PANELS. The range receiver radio remote control panel (AN/ARC-5) and the VHF receiver (AN/ARC-1) are located above the pilot's side window, forward of station 47. The receiver remote control panels are edge-lighted, and contain a mechanical tuning control and volume control for the range receiver, a VHF communication channel selector knob, and a VHF transmitter-receiver selector knob. Refer to paragraph 8-111 for the location of the IFF control panel.

8-7. RADIO MAIN JUNCTION BOX. The radio junction box is installed on the forward side of the radio rack, between the bottom of the main junction box and the fuselage floor. There are eight horizontal rows of terminal strips. Each of the four top rows contain 52 numbered terminal posts, and each of the four bottom terminal strips contain 40 numbered terminal strips. Most of the wiring for the electronic equipment terminates on individual terminals in the radio junction box. The terminal posts are numbered as shown in the wiring diagrams in Section X, Wiring Data.

8-8. RADIO WIRING. The numbering system used on the radio wiring is similar to that used for the electrical systems. For example, wire number 1RN-793C20 is a typical wire number. The first number

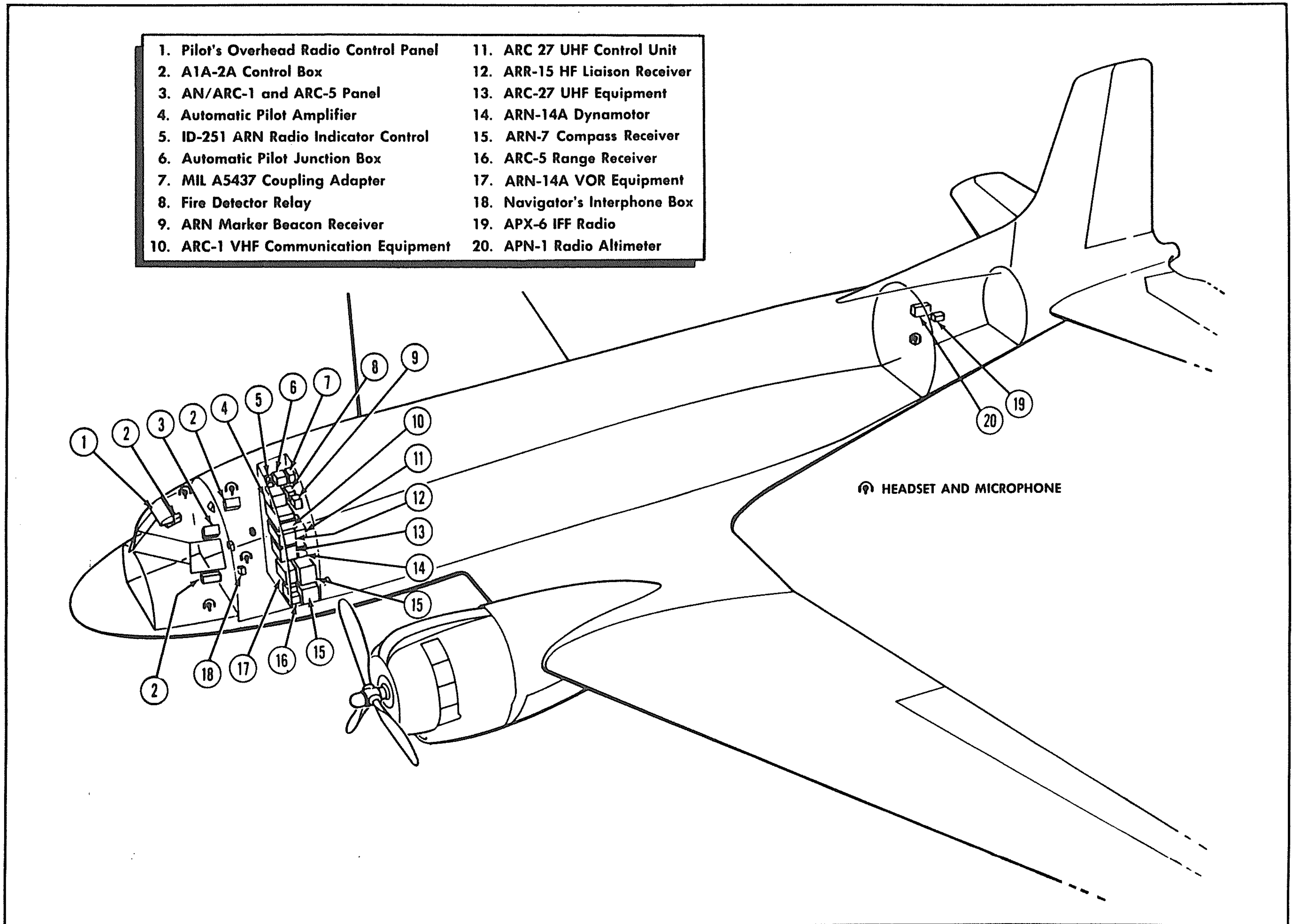


Figure 8-1. Electronics Equipment

	Primary Power Supply			*D-C Input*		*A-C Input*		*Max Power*		
Equipment Type Designation	*Battery and Generator System**	*Motor Alt Conv†*	*Dyna-motor*	*Voltage Range*	*Max Amp*	*Voltage Range*	*Amp*	*D-C Watts*	*A-C Volt Amps*	*PF*
ARR-15 Norm	X				3.1			87		
ARR-15 Chan	X				8.5			238		
BC-348	X				2.0			56		
ARC-27 Norm	X				17.0			475		
ARC-27 XMTR	X				3.0			84		
ART-13 Norm	X				11.0			308		
ART-13 XMTR	X				21.0			589		
AIA-2A	X				4.0			112		
ARC-5 (One Rec)	X				2.0			56		
ARN-7 Per Unit (Two)	X	X			0.6	105-125	0.87	17	132	0.96
ARN-8	X				2.0			56		
APN-1	X				2.8			78		
ARN-14	X	X	X		5.0	25-27	0.9	140	23	0.70
ARC-1 Norm	X				7.0			196		
ARC-1 XMTR	X				8.0			224		
APX-6	X	X			1.4	110-120	2.0	39	225	0.88
AM-40/AIC	X				2.0			56		

*28-Volt D-C Power Supply Consists of:
2 Type K8941850-G5 (300 Amp) Generators.
2 Exide 6FHM-13-1 (34 Amp Hr) Batteries.

†A-C Power Supply Consists of:
1 1500 va Inverter (E1737-1) 115 Volt A-C 400 Cycle.
1 250 va Inverter (E1617-1) 115 Volt A-C 400 Cycle.

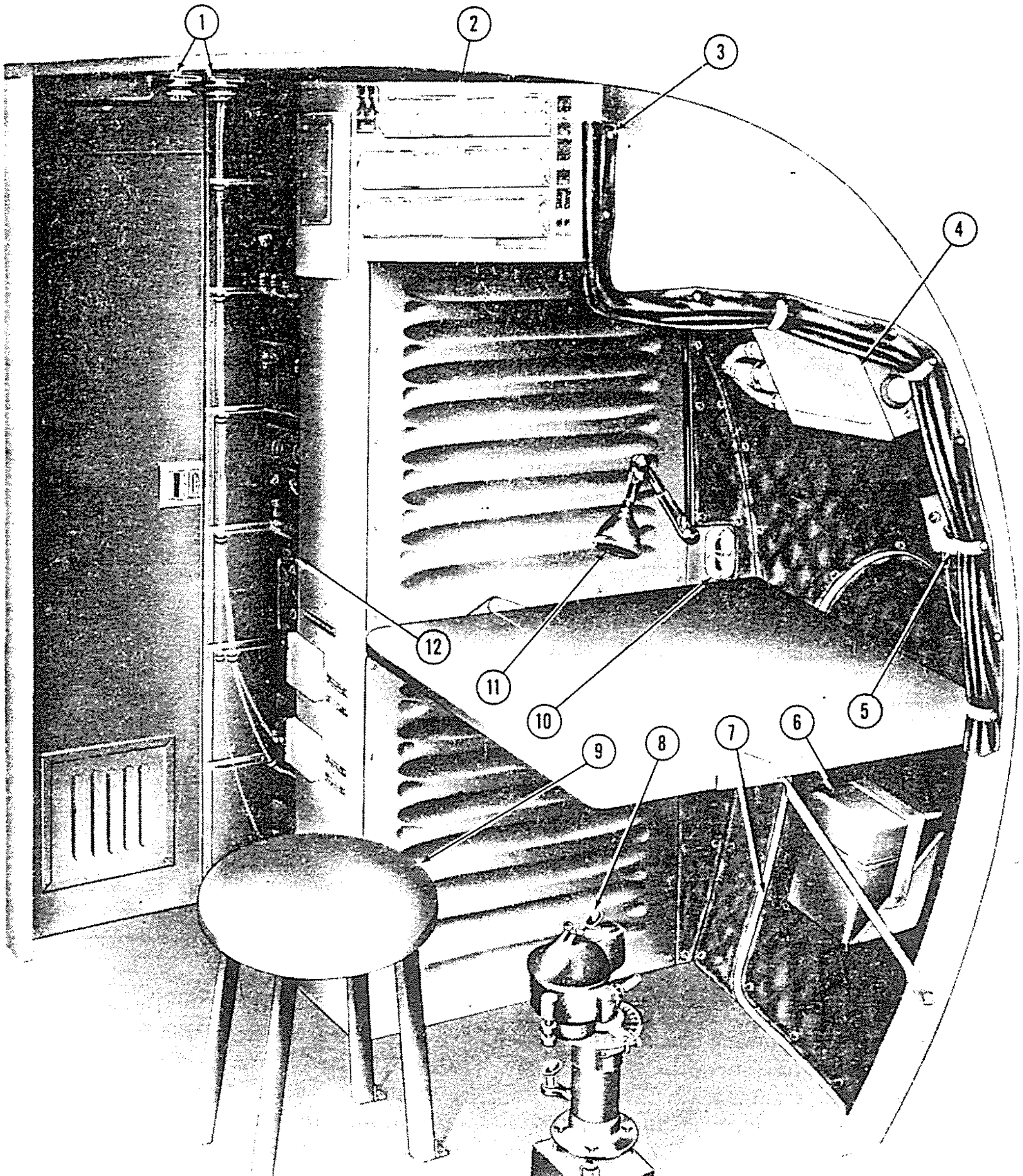
Figure 8-2. Electronic Equipment Power Input Chart

(1) is used when there are similar circuits which perform the identical functions, or serve the same purposes, but terminate in different places. The letters RN indicate the particular radio circuit. The number 793 indicates the series number of the wire. The letter C following the series number indicates the segment of the wire run. The last digits, 20, indicate the wire gage. The radio circuits are identified as follows:

RA	Glide Path Receiver
RC	Range Receiver
RG	(LORAN) Long Range Navigation
RL	HF Communication Equipment
RM	Marker Beacon Receiver
RN	VOR Navigation System
1RN	Red Radio Compass
2RN	Green Radio Compass
RU	UHF Transmitter-Receiver
RV	VHF Transmitter-Receiver
RZ	Interphone Radio Control and Public Address System
SA	Radio Altimeter
SX	IFF Equipment

8-9. POWER SUPPLY FOR ELECTRONICS EQUIPMENT. The electronics equipment using direct current is supplied with power from the aircraft batteries and generators. The equipment using alternating current is supplied from the two 115-volt, a-c inverters (see paragraph 7-54).

8-10. HF COMMUNICATION EQUIPMENT. A high-frequency communication transmitter (T-47/ART-13) is located on the second shelf above the radio operator's table to provide long-range, two-way, radio-telephone communication between the aircraft and ground stations, and also between the aircraft and other aircraft. In conjunction with the transmitter (T-47/ART-13), the high-frequency equipment consists of a radio receiver (BC-348), located on the shelf just below the high-frequency communication transmitter, a heterodyne frequency meter of the LM series, which is located on the shelf above the high-frequency communication transmitter, and a dynamotor unit



1. Periscopic Sextant Mounts
2. Main Circuit Breaker Panel
3. Anti-Glare Curtain
4. Junction Box
5. Interphone Control Box
6. Sextant Stowage
7. Emergency Escape Door
8. Driftmeter
9. Stool
10. Master Emergency Power Circuit Breaker Switch
11. Flexible Table Lamp
12. Radio Rack

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Figure 8-3. Navigator's Station

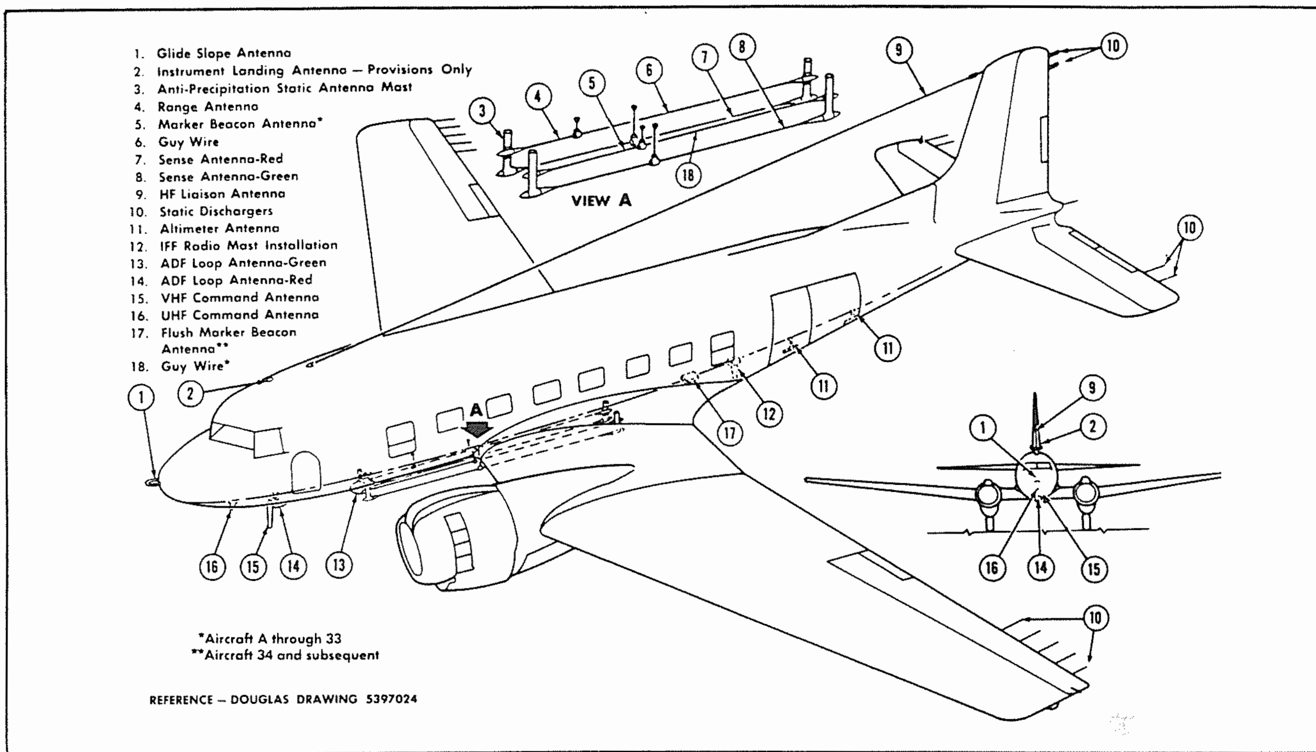


Figure 8-4. Radio Antennas

(DY-17/ART-13), located on the right side of the fuselage floor forward of station 97. In addition to the radio receiver (BC-348), located at the radio operator's station, another high-frequency communication liaison receiver (R-105A/ARR-15) is located on the radio rack forward of the UHF control panel. A disable relay (R20-4) is installed in the radio junction box for the purpose of making the BC-348 and the ARR-15 receivers inoperative when the ART-13 is keyed and the monitor switch is in the normal position. When the monitor switch is in the monitor position, the relay (R20-4) is disabled (the BC-348 and ARR-15 receivers are operative at all times), and the side tone from the ART-13 will be removed from the interphone system. This will enable the radio operator to check the RF output of the ART-13 with the receivers. The high-frequency communication equipment can be operated by the pilot or co-pilot by controls on the pilot's overhead radio control panel and by the radio operator. The high-frequency communications equipment operates in the high-frequency range between 2,000 and 18,100 kilocycles on multiple fixed-frequency channels. The high-frequency communications transmitter employs a master oscillator, which is rated at a power output of 30 to 90 watts at 100 per cent modulation. The high-frequency communication receiver (BC-348) furnishes "B" voltage for the heterodyne frequency meter. The antenna lead-ins from both receivers are connected to the antenna coupler (CU-92/APN), which is connected to the receiver terminal on the transmitter and is located outboard of the high-frequency communication transmitter. Electrical power for the HF communication equipment is supplied by the 28-volt, d-c circuit through a 10-ampere circuit breaker on the main circuit breaker panel for ARR-15 and through two circuit breakers, 35-ampere for ART-13, and 5-ampere for BC-348, located on the inboard side of the radio operator's junction box, which is located to the right of the companionway entrance on the forward side of station 97.

8-11. HF COMMUNICATION EQUIPMENT RADIO TRANSMITTER (T-47/ART-13).

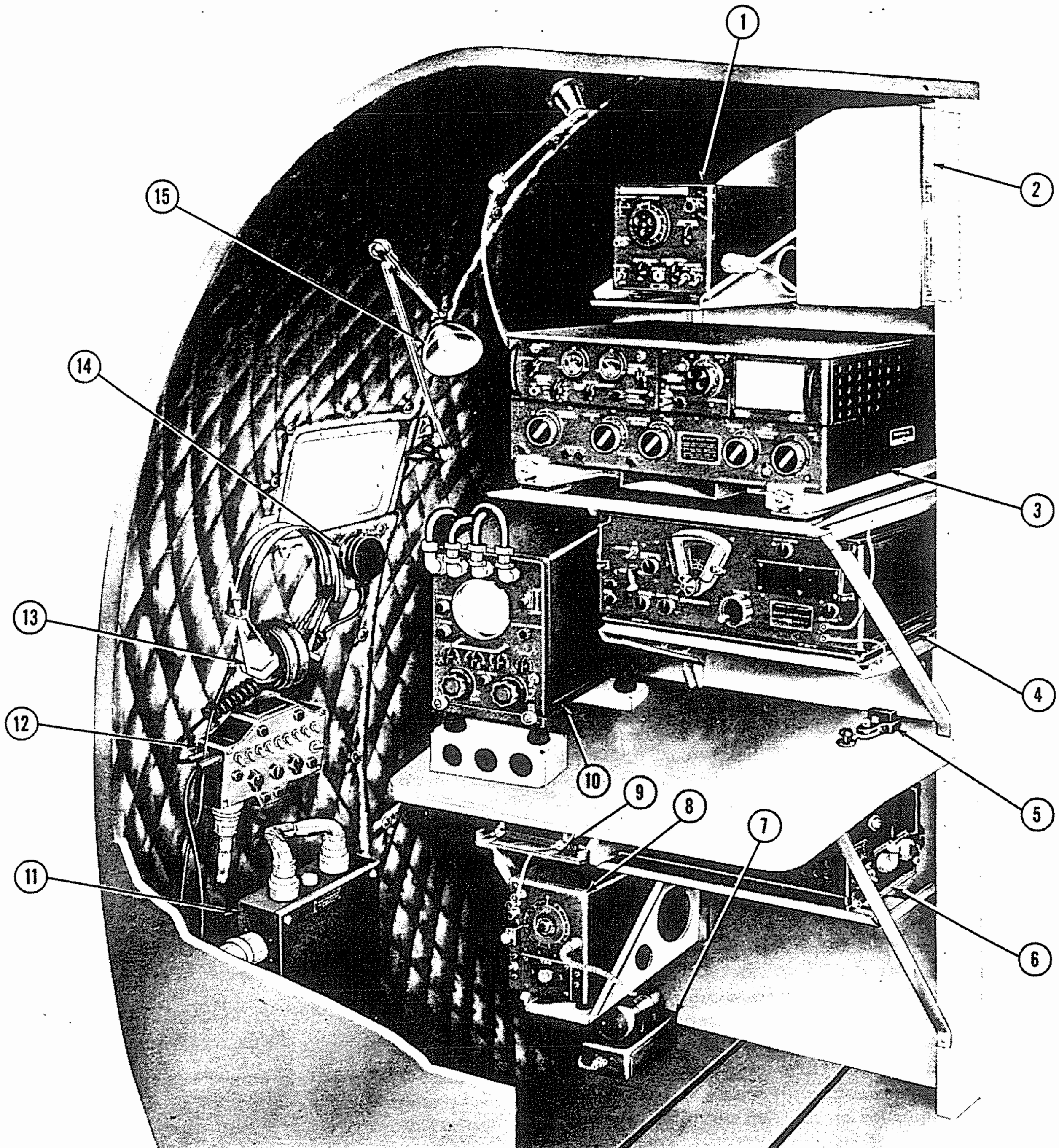
8-12. REMOVAL OF HF COMMUNICATION EQUIPMENT RADIO TRANSMITTER

a. Remove the two antenna lead-ins, ground wire, and two electrical disconnect plugs from the transmitter unit.

b. Loosen the two knobs at the forward base of the transmitter, and remove.

8-13. INSTALLATION OF HF COMMUNICATION EQUIPMENT RADIO TRANSMITTER. Reverse the removal procedure *(see figure 10-46).*

8-14. HF (LIAISON) COMMUNICATION ANTENNA. *(See figure 8-4.)* The fixed antenna for the HF liaison equipment extends from a lead-in mast (located on the top of the fuselage aft of station 78) to an insulator forward of the top leading edge of the vertical stabilizer.



1. LM Series Frequency Meter	9. AM-40/AIC Interphone Amplifier
2. Junction Box	10. AN/APN-4 Indicator
3. AN/ART-13 Transmitter	11. Junction Box
4. BC-348-1 Receiver	12. Audio Control Box
5. Key	13. Headset
6. AN/APN-4 Receiver	14. Microphone
7. AIA-2A Dynamotor	15. Flexible Table Lamp and Switch
8. R-23A/ARC-5 Receiver	

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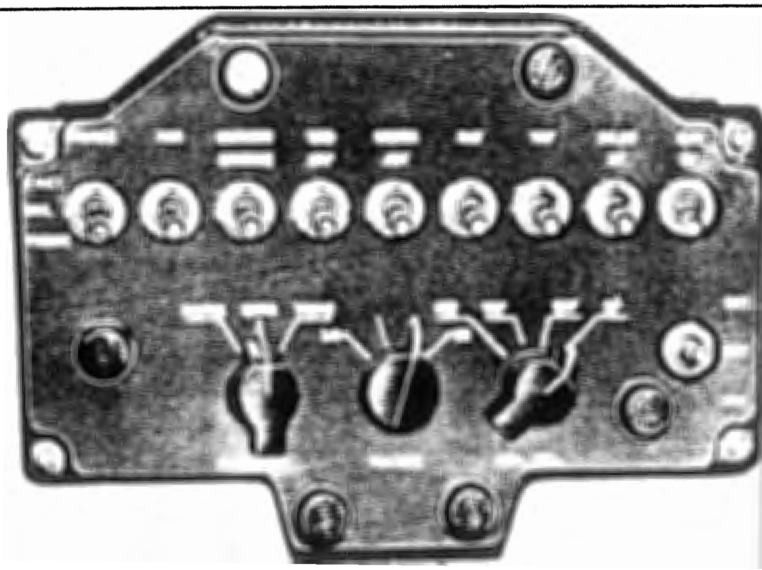
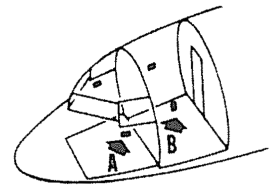
Figure 8-5. Radio Operator's Station

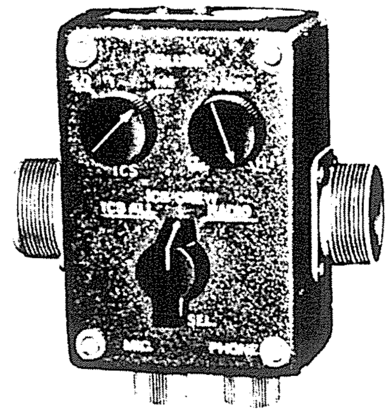


VIEW A
PILOT, CO-PILOT AND RADIO OPERATOR'S
INTERPHONE CONTROL BOX
(TYPICAL IN THREE PLACES)

Notes:

1. Range filter switch, controls only VOR and range receivers
2. Marker beacon and red ADF not affected by volume control





VIEW B
NAVIGATOR'S INTERPHONE
CONTROL BOX

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Figure 8-6. Audio Selector and Interphone Panels

8-15. REMOVAL OF HF (LIAISON) COMMUNICATION ANTENNA.

a. Remove the forward end of the antenna from the antenna mast chuck assembly by unscrewing the cap on the aft top end of the mast. Remove the chuck and cap from the antenna and reassemble to prevent the loss of parts.

b. Loosen the screw from the inboard end of the antenna mast to remove the lead-in cable.

c. Remove the eight antenna mast support screws from the base of the mast.

8-16. INSTALLATION OF HF (LIAISON) COMMUNICATION ANTENNA. Reverse the removal procedure, except when installing a new antenna. When installing a new antenna, proceed as follows:

a. Remove approximately ¾ inch of the insulation from the end of the antenna, and insert the stripped wire through the cap and into the chuck jaws.

b. Insert the chuck into the cap. The jaws will tighten the wire as the cap is screwed into place.

8-17. HF COMMUNICATION RADIO RECEIVER (BC-348).

8-18. REMOVAL OF HF COMMUNICATION RADIO RECEIVER.

a. Remove the antenna lead-in and ground wire.

b. Loosen the fasteners on the bottom forward base of the receiver to remove it.

8-19. INSTALLATION OF HF COMMUNICATION RADIO RECEIVER (BC-348). Reverse the removal procedure *(see figure 10-46).*

8-20. HF COMMUNICATION HETERODYNE FREQUENCY METER (LM SERIES).

8-21. REMOVAL OF HF COMMUNICATION HETERODYNE FREQUENCY METER.

a. Remove the electrical disconnect plug.

b. Loosen the fasteners at the base of the unit and remove it.

8-22. INSTALLATION OF HF COMMUNICATION HETERODYNE FREQUENCY METER. Reverse the removal procedure *(see figure 10-46).*

8-23. HF COMMUNICATION DYNAMOTOR UNIT (DY-17/ART-13).

8-24. REMOVAL OF HF COMMUNICATION DYNAMOTOR UNIT.

a. Remove the two electrical disconnect plugs from the dynamotor unit.

b. Loosen the fasteners at the base of the dynamotor, and remove the unit.

8-25. INSTALLATION OF HF COMMUNICATION DYNAMOTOR UNIT. Reverse the removal procedure *(see figure 10-46).*

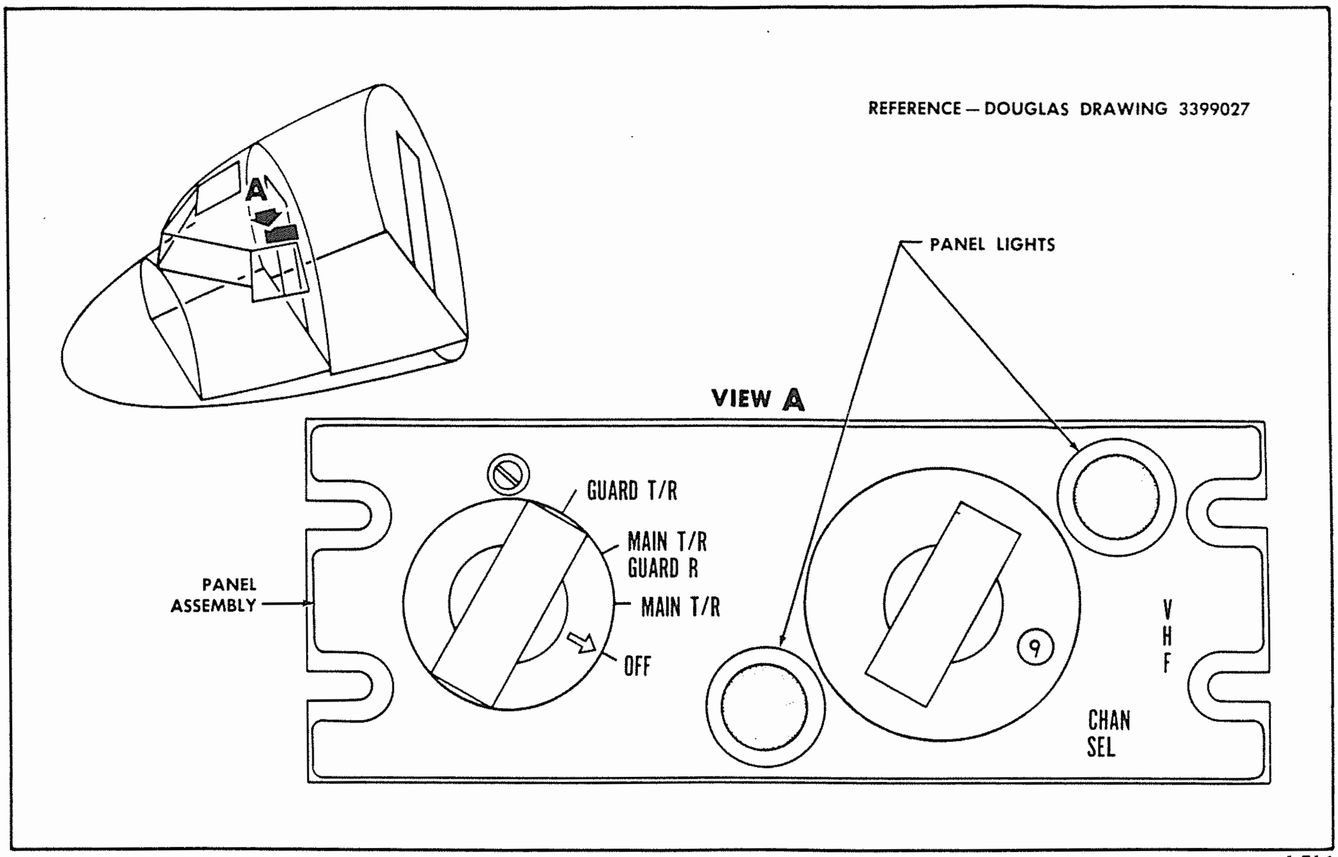

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Figure 8-7. VHF Transmitter-Receiver Control Panel

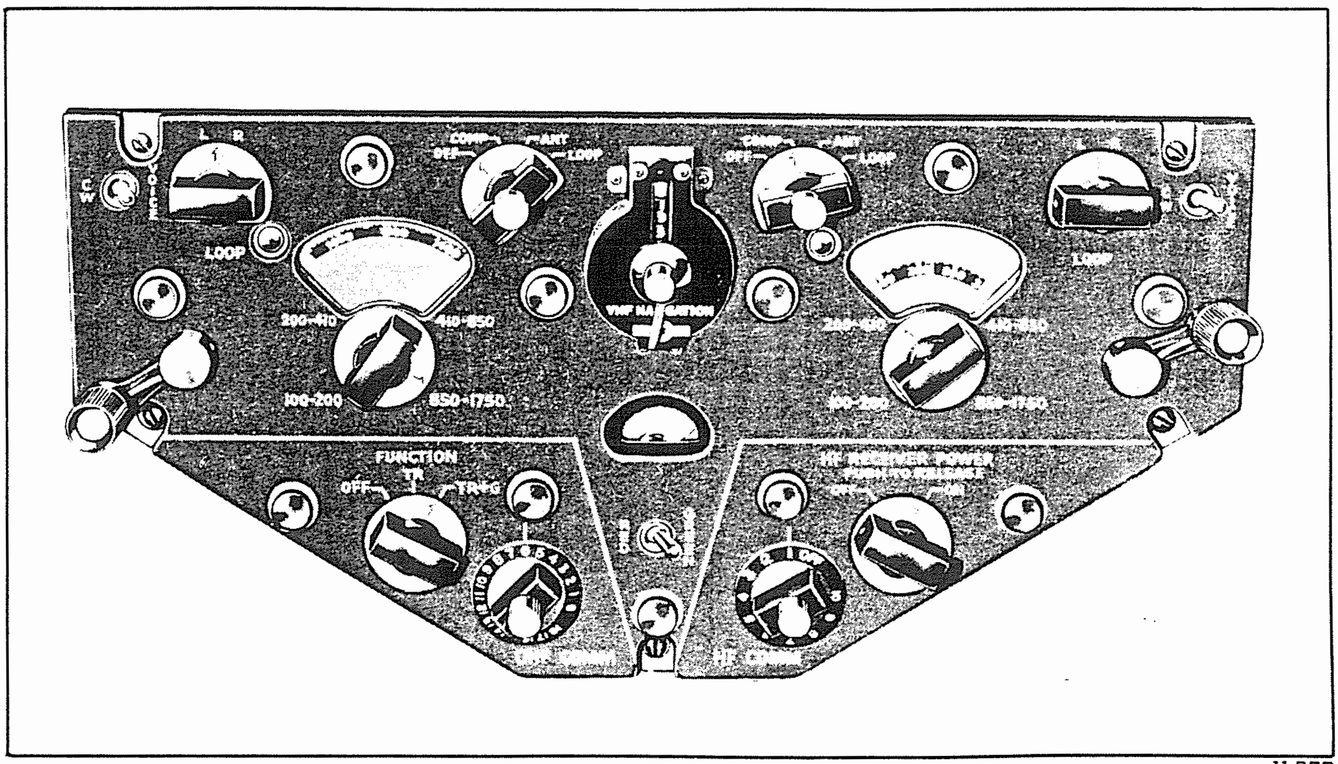

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Figure 8-8. Pilot's Radio Remote Control Panel

8-26. HF COMMUNICATION RADIO RECEIVER (R-105A/ARR-15).

8-27. REMOVAL OF HF COMMUNICATION RADIO RECEIVER.

a. Remove the antenna lead-in from the receiver.

b. Loosen the two wing nuts at the inboard base of the receiver and remove it.

8-28. INSTALLATION OF HF COMMUNICATION RADIO RECEIVER. Reverse the removal procedure *(see figure 10-46).*

8-29. HF COMMUNICATION DISABLE RELAY (8504-2).

8-30. REMOVAL OF HF COMMUNICATION DISABLE RELAY. *(See figure 10-46.)*

a. Remove the radio operator's junction box cover.

b. Remove the six wires from the relay.

c. Remove the attach screws in the base of the relay.

8-31. INSTALLATION OF HF COMMUNICATION DISABLE RELAY. Reverse the removal procedure.

8-32. RADIO OPERATOR'S HF COMMUNICATION TRANSMITTING KEY. A code transmitting key is installed on the radio operator's table for CW operation.

8-33. REMOVAL OF RADIO OPERATOR'S HF COMMUNICATION TRANSMITTING KEY.

a. Remove the two wires from the terminals on the transmitting key base.

b. Remove the four screws, washers, and nuts from the base of the transmitting key.

8-34. INSTALLATION OF RADIO OPERATOR'S HF COMMUNICATION TRANSMITTING KEY. Reverse the removal procedure *(see figure 10-46).*

8-35. VHF COMMUNICATION TRANSMITTER-RECEIVER. The VHF communication equipment is a radio-telephone transmitter-receiver (RT-18/ARC-1), operating in VHF range of 108 to 155.88 megacycles on multiple fixed-frequency, crystal-controlled channels and one guard channel. The transmitter section has a nominal R-F carrier output of eight watts and is capable of approximately 100 per cent modulation with voice signals. The distance over which communication can be conducted with this equipment varies considerably. Under normal conditions, reliable communication may be expected over line-of-sight distances. The receiver section features background noise-squelch operating from a pre-determined noise level and modified or delayed AVC, allowing reception on the selected main-channel frequency, plus monitoring of a separate guard channel frequency. While the equipment is in service, a cover on the front panel protects the crystal units, the control knobs, the radio receiver sensitivity adjustment, the squelch-disabling switch, the test jacks, and a secondary power control switch. The transmitter-receiver is located on the fifth shelf of the main radio rack, and includes all of the equipment necessary for transmission and reception of voice signals in the VHF range. The transmitter-receiver is controlled from the pilot's overhead VHF control unit located above the pilot's side window, forward of station 47, and operates through the audio selector panels. Electrical power for the VHF communication equipment is supplied by the 28-volt d-c circuit through a 15-ampere circuit breaker on the circuit breaker panel. The output is controlled through the ARC-1 volume control resistor (JLU5011) located in the radio main junction box.

8-36. REMOVAL OF VHF TRANSMITTER-RECEIVER.

a. Remove the antenna from the unit.

b. Loosen the two wing nuts at the inboard base of the transmitter-receiver to slide the unit inboard and remove the unit from the radio rack.

8-37. INSTALLATION OF VHF TRANSMITTER-RECEIVER. Reverse the removal procedure *(see figure 10-54).*

8-38. VHF TRANSMITTER-RECEIVER ANTENNA. *(See figure 8-4.)* The VHF transmitter-receiver antenna is installed on the bottom of the fuselage, aft of station 24, between left longerons 23 and 24.

8-39. REMOVAL OF VHF TRANSMITTER-RECEIVER ANTENNA.

a. Remove the lead-in connection from the top of the antenna assembly, under the pilot's floor.

b. Loosen the two bonding clamp screws along the forward side of the antenna mast.

c. Loosen the four bolts in the antenna clamp assembly.

d. Loosen the clamp assembly to remove the antenna from the cutout in the fuselage.

8-40. INSTALLATION OF VHF TRANSMITTER-RECEIVER ANTENNA. Reverse the removal procedure.

Note

When installing the antenna, determine that there is $\frac{3}{16}$-inch clearance between the metallic coating on the mast and the fuselage skin.

8-41. UHF TRANSMITTER-RECEIVER. The UHF communication equipment (AN/ARC-27) is designed to provide AM radio-telephone communication in the frequency range of 225.0 to 399.9 megacycles between aircraft and ships, aircraft and land, or between aircraft. The transmitter-receiver (RT-178/ARC-27) may be tone modulated at 1020 cycles per second for

emergency-finding purposes. The equipment provides 1750 frequency channels. The transmitter section has a nominal R-F carrier output of nine watts minimum and is capable of approximately 95 per cent modulation of voice signals. The UHF transmitter-receiver is operated from the overhead radio panel in the cockpit and through the audio selector control panels for the pilot, co-pilot, and radio operator. A control panel (C-626/ARC-27), for the transmitter-receiver is installed on the fourth shelf of the radio rack. Any of 18 preset frequencies may be selected or any guard channel frequency may be used on the control panel. Transmission and reception are on the same frequency and both utilize the same antenna. The equipment operates on 28-volt d-c through a 25-ampere circuit breaker, located on the circuit breaker control panel.

8-42. REMOVAL OF UHF TRANSMITTER-RECEIVER.

a. Remove the antenna lead-in and two electrical disconnect plugs from the unit.

b. Loosen the two wing nuts at the inboard base of the transmitter-receiver and remove the transmitter-receiver.

8-43. INSTALLATION OF UHF TRANSMITTER-RECEIVER. Reverse the removal procedure *(see figure 10-53).*

8-44. UHF CONTROL PANEL (C-626/ARC-27).

8-45. REMOVAL OF UHF CONTROL PANEL.

a. Remove the three electrical disconnect plugs from the base of the control panel.

b. Loosen the fasteners at the inboard base of the control panel and remove the panel.

8-46. INSTALLATION OF UHF CONTROL PANEL. Reverse the removal procedure *(see figure 10-53).*

8-47. UHF TRANSMITTER-RECEIVER ANTENNA. *(See figure 8-4.)* The UHF transmitter-receiver antenna is installed on the bottom of the fuselage, at station 1, to the right of the center line of the aircraft.

8-48. REMOVAL OF UHF TRANSMITTER-RECEIVER ANTENNA.

a. Remove the four attaching bolts, washers, and nuts from the antenna support, and the one forward screw in the top of the antenna. Then lower the mast.

b. Remove the lead-in connection from the top of the antenna.

8-49. INSTALLATION OF UHF TRANSMITTER-RECEIVER ANTENNA. Reverse the removal procedure.

8-50. UHF RECEIVER SQUELCH CONTROLS. The UHF receiver squelch controls (RT-178/ARC-27) are located on the front panel of the receiver-transmitter unit. The squelch controls should be set *during flight* under normal or average conditions regarding the presence of radio noise in the UHF spectrum.

8-51. ADJUSTMENT OF UHF RECEIVER SQUELCH CONTROLS.

a. Channel the unit to a frequency on which *no signals are being received.*

b. Operate the unit from the pilot's control box by placing the LOCAL-REMOTE switch on the radio operator's control box in the REMOTE position.

c. Place the function switch on the pilot's control box in T/R position. (In this position, the main receiver *only* is heard in the headset.)

d. Plug in a headset at the phone jack on the front panel of the receiver-transmitter unit.

e. Turn both squelch controls fully clockwise.

f. If no background hiss is heard in the headset, the receiver may be unusually quiet regarding internal thermal noise (a desirable condition), or it may be inoperative. To check this push the SQ OFF button located above the MAIN REC squelch control. If the hiss is heard, then the receiver is operating normally and the squelch requires no further adjustment. If no hiss is heard, the receiver is probably inoperative.

Note

If background hiss is heard *before* pushing the squelch SQ OFF button, turn the MAIN REC squelch control *slowly* counterclockwise to a point just slightly beyond the point at which the noise in the headset is no longer heard.

g. Place the function switch on the pilot's control box in T/R and G REC position. (In this position both the main and guard receivers are heard in the headset. If the main receiver has been silenced as in step f, any noise heard will be from the guard receiver.)

h. Repeat the operation, as in step f, using the GUARD REC squelch and SQ OFF button.

Note

This completes the setting of receiver squelch controls. It is emphasized that these adjustments should not be made in an area where the presence of radio noise in the UHF spectrum is known to be excessive. To do so will render the receivers insenitive to any but the very strongest transmission signals. When properly set, the squelch may be expected to "crack" occasionally during take-off, or under unusual conditions causing noise of temporary duration.

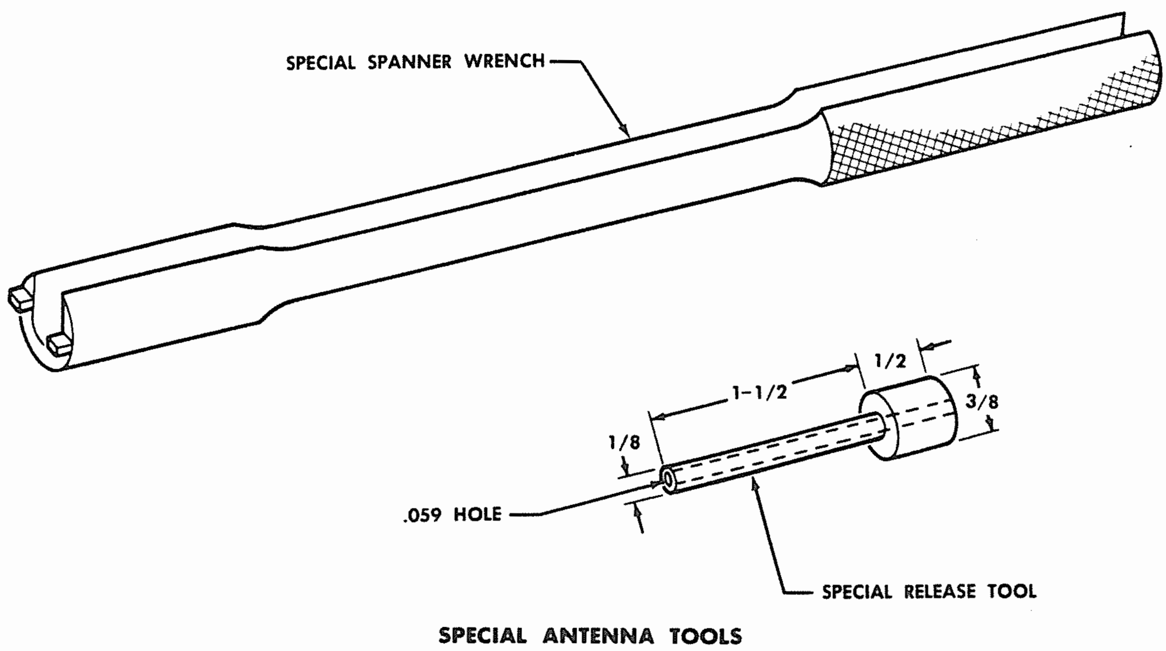


Figure 8-9. Special Antenna Tools

8-52. MARKER BEACON RECEIVER. A marker beacon receiver (R-8/ARN-8) is installed in the main radio rack, next to the top shelf outboard of the fire detector relay. The marker beacon receiving equipment is used as a navigational aid, and provides either a visual light indication or a signal on the interphone system when the aircraft passes over a marker beacon transmitter on the ground. The equipment operates on 28-volt d-c power through a 5-ampere circuit breaker located on the circuit breaker panel. The marker beacon receiver operates the marker indicator lights located in the pilot's course indicator, and the co-pilot's deviation indicator located on the pilot's and co-pilot's instrument panels. The marker beacon indicator lights are connected to the automatic dimming relay located in the main junction box.

8-53. REMOVAL OF MARKER BEACON RECEIVER.

a. Remove the engine fire detector relay box, located inboard of the marker beacon receiver, by removing the electrical disconnect plugs and the shock mount fasteners at the base of the relay.

b. Remove the antenna lead-in and electrical disconnect plugs from the marker beacon receiver.

c. Loosen the inboard slide fasteners at the base of the receiver to slide the receiver inboard, and lift the receiver over the fire detector relay box shock mounting bracket.

8-54. INSTALLATION OF MARKER BEACON RECEIVER. Reverse the removal procedure *(see figure 10-49).*

8-55. MARKER BEACON ANTENNA (AIRCRAFT A, B, AND 1 THROUGH 33). The marker beacon antenna is installed on the bottom left side of the fuselage (between station 102 and station 188), and is the upper forward wire on the same masts used for the GREEN ADF sense antenna.

8-56. REMOVAL OF MARKER BEACON ANTENNA (AIRCRAFT A, B, AND 1 THROUGH 33).

a. Remove the antenna support sleeve from the mast body by unscrewing the sleeve end from the threaded portion of the mast.

b. Unscrew the (methylmethacrylate) screw from the internal threads of the mast body, using a narrow screw driver or a special spanner wrench *(see figure 8-9).*

c. Remove the cap from the dead end body assembly.

Note

When it is necessary to replace the antenna wire, cut the antenna wire approximately four inches from the dead end body and, using a pair of pliers, mash the remaining insulation on the wire to remove the insulation from the wire. Slide the special release tool *(see figure 8-9)* over the conductor, hold the dead end body in one hand, and press the special release tool against the jaws in the dead end body to release and remove the conductor from the assembly. (When replacing an antenna that has been installed for a long period of time, it will be found that the jaws in the dead end body will not release the conductor when the release tool is used normally. In such cases it will be necessary to use a nonmetallic tool to tap lightly against the release tool to separate the jaws from the conductor.)

8-57. INSTALLATION OF MARKER BEACON ANTENNA (AIRCRAFT A, B, AND 1 THROUGH 33). Reverse the removal procedure, except when installing a new antenna. When installing a new antenna, proceed as follows:

a. Strip off one inch of the insulation from the end of the antenna to insert into the dead end body chuck jaws.

b. When replacing a dead end body seal, dip the seal in Silicone oil before inserting the seal into the end of the body. Use Dow-Corning Silicone oil; viscosity at 21.6° C (71° F) equals 56 centipoises.

CAUTION

Always keep the dead end body clean and free from foreign particles. Do not soak the entire body in any oils.

Note

When assembling the antenna on the dead end mast, coat the outside insulation and threads of sleeve with DC-4 (AN-C-128).

8-58. FLUSH-TYPE MARKER BEACON ANTENNA (AIRCRAFT C, D, AND 34 THROUGH 96). The flush-type marker beacon antenna (AT-134-A/ARN) is installed at station 372 to the left side of the bottom center line.

8-59. REMOVAL OF FLUSH-TYPE MARKER BEACON ANTENNA (AIRCRAFT C, D, AND 34 THROUGH 96).

a. Remove the screws holding the antenna to the skin of the aircraft.

b. Drop the antenna, and remove the coaxial antenna feeder.

8-60. INSTALLATION OF FLUSH-TYPE MARKER BEACON ANTENNA (AIRCRAFT C, D, AND 34 THROUGH 96). Reverse the removal procedure.

8-61. TUNING OF FLUSH-TYPE MARKER BEACON ANTENNA (AIRCRAFT C, D, AND 34 THROUGH 96).

a. Raise the tail of the aircraft to a level or flight attitude.

b. Using a 75-megacycle, modulated test oscillator, peak the antenna trimmer for maximum signal.

c. Check the marker indicator light for operation during modulation.

Note

The tail of the aircraft is raised to reduce ground effect.

8-62. RANGE RECEIVER. Low-frequency range signals are received by the range receiver equipment. The range receiver and dynamotor unit (R-23/ARC-5) is located on the bottom shelf of the radio rack on aircraft A, B, and 1 through 8. On aircraft C, D, 9, and subsequent, this range receiver and dynamotor is installed on the floor at the radio operator's position and is controlled by the pilot from a control panel located above the pilot's side window. The range receiver radio control panel contains a mechanical tuning control for the range receiver, and a sensitivity control. An audio selector switch is installed on the pilot's, copilot's, and radio operator's interphone control boxes. Power is supplied to the range receiver equipment from the aircraft's 28-volt d-c circuit through a 5-ampere circuit breaker.

8-63. REMOVAL OF RANGE RECEIVER.

a. Remove the mechanical tuning shaft end from the receiver coupler.

b. Disconnect the antenna from the receiver.

c. Loosen the two receiver support clamps on the bottom inboard side of the receiver to slide the receiver inboard, and remove the receiver from the radio rack.

8-64. INSTALLATION OF RANGE RECEIVER. Reverse the removal procedure. After installation, synchronize the control head with the dial on the receiver. Re-tighten the knurled nuts on the control head and the receiver dials.

8-65. RANGE RECEIVER ANTENNA. *(See figure 8-4.)* The range receiver antenna (E-1795) is the anti-precipitation static type and is installed on the bottom right side of the fuselage (between station 102 and station 318). It is the upper wire on the same masts used for the RED ADF compass sense antenna.

8-66. REMOVAL OF RANGE RECEIVER ANTENNA.

a. Remove the antenna support sleeve from the mast body by unscrewing the sleeve end from the threaded portion of the mast.

b. Unscrew the (methlymethacrylate) screw from the internal threads of the mast body, using a narrow screw driver or special spanner wrench *(see figure 8-9)*.

c. Remove the cap from the dead end body assembly.

Note

When it is necessary to replace the antenna wire, cut the antenna wire approximately four inches from the dead end body and, using a pair of pliers, mash the remaining insulation on the wire to remove the insulation from the wire. Slide the special release tool *(see figure 8-9)* over the conductor, hold the dead end body in one hand, and press the special release tool against the jaws in the dead end body to release and remove the conductor from the assembly. (When replacing an antenna that has been installed for a long period of time, it will be found that the jaws in the dead end body will not release the conductor when the release tool is used normally. In such cases it will be necessary to use a nonmetallic tool to tap lightly against the release tool to separate the jaws from the conductor.)

8-67. INSTALLATION OF RANGE RECEIVER ANTENNA. Reverse the removal procedure, except when a new antenna is being installed. If a new antenna is being installed, proceed as follows:

a. Strip off one inch of the insulation from the end of the antenna to insert into the dead end body chuck jaws.

b. When replacing a dead end body. seal, dip the seal in Silicone oil before inserting the seal into the end of the body. Use Dow-Corning Silicone oil; viscosity at 21.6° C (71° F) equals 56 centipoises.

CAUTION

Always keep the dead end body clean and free from foreign particles. Do not soak the entire body in any oils.

Note

When assembling the antenna on the dead end mast, coat the outside insulation and threads of sleeve with DC-4 (AN-C-128).

8-68. AUTOMATIC RADIO COMPASSES — RED AND GREEN. Two automatic radio compasses (AN/ARN-7) are installed in the aircraft. One radio compass is designated as the RED radio compass and the other is designated as the GREEN radio compass. The frequency range of each radio compass unit is 100 to 1750 kilocycles covered in four bands. The RED compass receiver is located on the bottom shelf of the main radio rack. The GREEN compass receiver is located on the shelf above the RED compass receiver. Two separate controls are installed on the pilot's overhead radio control panel for mechanical tuning of the two compasses. Both loop and sense antennas are used in the operation of the two receivers. Power is supplied to the equipment through power relays in the radio main junction box from the aircraft's 28-volt d-c system and from the 1500 va inverter, 115-volt a-c system.

8-69. COMPASS RECEIVERS — RED OR GREEN (R-5ARN-7).

8-70. REMOVAL OF COMPASS RECEIVERS — RED OR GREEN.

a. Disconnect the sense antenna and ground connections, and remove the two electrical disconnect plugs from the receiver receptacles.

b. Remove the mechanical tuning shaft end from the receiver.

c. Loosen the two fasteners in the base of the receiver in order to slide the receiver inboard and to remove the receiver from the radio rack.

8-71. INSTALLATION OF COMPASS RECEIVERS — RED OR GREEN. Reverse the removal procedure.

8-72. ADF POWER RELAYS — RED OR GREEN.

8-73. REMOVAL OF ADF POWER RELAYS — RED OR GREEN.

a. Remove the six wires from the relay terminals located inboard of the bottom terminal strip in the radio junction box *(see figures 10-42 and 10-44)*.

Note

The RED ADF power relay is located next to the terminal strip. The GREEN ADF power relay is located inboard of the RED ADF power relay.

b. Remove the attaching screws in the base of the relay.

8-74. INSTALLATION OF ADF POWER RELAYS — RED OR GREEN. Reverse the removal procedure.

8-75. AUTOMATIC RADIO COMPASS SENSE ANTENNAS. *(See figure 8-4.)* The two automatic radio compass sense antennas (E-1795) are the anti-precipitation static type and are installed on both left and right lower sides of the fuselage, extending from station 102 to station 318. The lower antenna on each side is the sense antenna; the lower right antenna is provided for the RED ADF radio compass, and the lower left antenna for the GREEN ADF radio compass. The upper antenna on the left mast is the marker beacon antenna, on aircraft A, B, and 1 through 33. On aircraft C, D, 34 and subsequent, a flush-type antenna is installed at station 372 to the left side of the bottom center line. The upper antenna on the right mast is the range antenna.

8-76. REMOVAL OF AUTOMATIC RADIO COMPASS SENSE ANTENNAS.

a. Remove the antenna support sleeve from the mast body by unscrewing the sleeve end from the threaded portion of the mast.

b. Unscrew the (methylmethacrylate) screw from the internal threads of the mast body, using a narrow screw driver or special spanner wrench *(see figure 8-9)*.

c. Remove the cap from the dead end body assembly.

Note

When it is necessary to replace the antenna wire cut the antenna wire approximately four inches from the dead end body and, using a pair of pliers, mash the remaining insulation on the wire to remove the insulation from the wire. Slide the special release tool *(see figure 8-9)* over the conductor, hold the dead end body in one hand, and press the special release tool against the jaws in the dead end body to release and remove the conductor from the assembly. (When replacing an antenna that has been installed for a long period of time, it will be found that the jaws in the dead end body will not release the conductor when the release tool is used normally. In such cases it will be necessary to use a non-metallic tool to tap lightly against the release tool to separate the jaws from the conductor.)

8-77. INSTALLATION OF AUTOMATIC RADIO COMPASS SENSE ANTENNAS. Reverse the removal procedure, except when installing a new antenna. If a new antenna is being installed, proceed as follows:

a. Strip off one inch of the insulation from the end of the antenna to insert into the dead end body chuck jaws.

b. When replacing a dead end body seal, dip the seal in Silicone oil before inserting the seal into the end of the body. Use Dow-Corning Silicone oil; viscosity at 21.6° C (71° F) equals 56 centipoises.

CAUTION

Always keep the dead end body clean and free from foreign particles. Do not soak the entire body in any oils.

Note

When assembling the antenna to the dead end mast, coat the outside insulation and threads of the sleeve with DC-4 (AN-C-128) compound.

c. Weather-seal all masts at the openings in the wing surface by applying duPont Fairprene 5112. Work the material well into cracks with a spatula; fair-in all material to present an even contour.

8-78. AUTOMATIC RADIO COMPASS LOOP ANTENNAS. *(See figure 8-4.)* Automatic radio compass loop antennas for both the RED ADF and the GREEN ADF radio compasses are installed on the bottom of the fuselage. The RED ADF automatic radio compass loop antenna is located between stations 24 and 47; the GREEN ADF automatic radio compass loop antenna is located between stations 97 and 117. Both of the radio compass loop antennas have a rotating range of 360 degrees. Dual (Azimuth) loop antenna indicators are provided on the left and right sides of the main instrument panel to indicate the position of either or both RED and GREEN ADF loop antennas. On aircraft C, D, 69 and subsequent, the ID-250-type azimuth indicator is installed on the main instrument panel, one for the pilot and one for the co-pilot. On the pilot's side, the wide hand is connected to the VOR receiver and the narrow hand is connected to the RED ADF receiver. On the co-pilot's side, the wide hand is connected to the VOR receiver but the narrow hand is actuated by the GREEN ADF receiver. A switch is installed adjacent to each (ID-250) indicator which enables the pilot or co-pilot to use one ADF receiver and one VOR receiver, or both ADF receivers.

8-79. REMOVAL OF AUTOMATIC RADIO COMPASS LOOP ANTENNAS.

a. Remove the attaching screws from around the antenna base.

b. Remove the two plug connections.

c. Remove the dehydrator piping from the loop.

8-80. INSTALLATION OF AUTOMATIC RADIO COMPASS LOOP ANTENNAS. Reverse the removal procedure.

8-81. AUTOMATIC RADIO COMPASS LOOP DEHYDRATORS. Both the RED and GREEN ADF automatic radio compass loop antennas are equipped with a dehydrator unit to prevent the accumulation of moisture condensation inside the units. The dehydrator is a transparent container filled with activated silica gel crystals impregnated with cobalt-chloride. The containers are installed vertically, aft of the radio operator's window on the right side of the fuselage, and are connected to the loop units by flexible tubing and suitable fittings so that the loop units breathe through the moisture-absorbing material. Because the loops are sealed, all air entering or leaving the loop housings must pass through the column of silica gel, where water vapor is removed from the air. The small amount of cobalt-chloride in the silica gel is a color indicator of moisture content, being dark blue when dry, and light blue or pink when enough moisture has been absorbed to require reactivation.

8-82. SWINGING AND COMPENSATING AUTOMATIC RADIO COMPASS — RED AND GREEN (AN/ARN-7). All loops installed should be calibrated according to the chart in figure 8-10. Swinging and compensating the automatic radio compass will not be necessary unless a major change is made in the external configuration of the aircraft. Distortion of the radio frequency field pattern in the vicinity of the aircraft is caused by wings, engines, propellers, antennas, and other parts of the aircraft. This distortion makes it necessary to check the direction of radio bearings at intervals of 15 degrees or less with respect to the forward-and-aft axis of the aircraft. Errors determined by a calibration check may be compensated for, provided the errors do not exceed ±25 degrees. The accuracy of a ground calibration of an aircraft having the loop mounted on the lower side of the fuselage may be affected by the proximity of the loop and sense antennas to the ground. Such a calibration should be considered only approximately until checked in the air. Aircraft error increases with the frequency so that the greatest errors will occur at the highest frequencies used for radio compass receiver operation. Consequently, calibration should be made on at least one station in each band and on frequencies most generally used, where greatest accuracy is required. The errors to be corrected in the loop compensator should be those occurring at a frequency where it is expected that the greatest bearing accuracy will be required. Accurate bearings may generally be obtained on stations in the frequency range of 200 to 1000 kilocycles. The error caused by a change in frequency from 200 to 1000 kilocycles does not usually exceed three degrees. When calibration data obtained at the midpoint of this range is used for compensation of the loop, bearings generally read directly from the bearing indicator at any other frequency between 200 and 1000 kilocycles, and should not be in error by more than two degrees. The equipment is calibrated in flight by one of two methods. Method 1 can be used when a suitable ground reference line can be found. Method 2 can be used from two opposite 360-degree turns. Method 2 is performed in considerably less time than Method 1, but calls for close cooperation between the pilot and the observer.

8-83. Perform Method 1 as follows:

a. Select a series of landmarks that will provide a direct line toward a suitable radio station 25 to 100 miles distant. A road, railroad, power line, or section line makes a good reference line. Distortion of the radio field caused by structure on the reference line (a power line, for example), should be checked by noting the action of the bearing indicators as the aircraft is flown on a steady heading across the reference line. If irregularities occur in the radio bearing indications as the line is approached or crossed, distortion of the radio wave is present and should be eliminated by flying at a higher altitude or using a different location.

b. Head the aircraft toward the transmitter while flying directly over and along the reference line. Fly at an altitude low enough to avoid sighting error, and keep the aircraft in level flight. Record the reading of the directional gyro and use it for future reference as the zero heading. Note and record readings of the bearing indicators for this zero heading of the aircraft. The indicator readings should be 0 (±2) degrees, if there is no drift and if all adjustments have been made properly.

c. A diagram of the flight procedure to be used is shown in figure 8-10. After the reading of the zero heading is obtained, turn the aircraft to the left and fly far enough from the reference line so that the aircraft may be turned to the right and a gyro heading displaced 15 degrees from the zero heading established before the reference line is crossed. With the aircraft held in level flight on a steady gyro heading, 15 degrees displaced from the zero heading, observe the bearing indicator readings when the aircraft is exactly over the reference line. Record the readings on a form similar to the one shown in figure 8-11.

d. Fly far enough to the right of the reference line so that a left turn may be made, and establish a gyro heading displaced (345 degrees) from zero heading before the reference line is recrossed. Note and record the bearing indicator readings when the aircraft is exactly over the reference line in level flight and on a steady gyro heading.

e. Repeat steps c and d for gyro headings displaced 30 and 330 degrees from the zero heading. After the bearing indicator readings are obtained for the head-

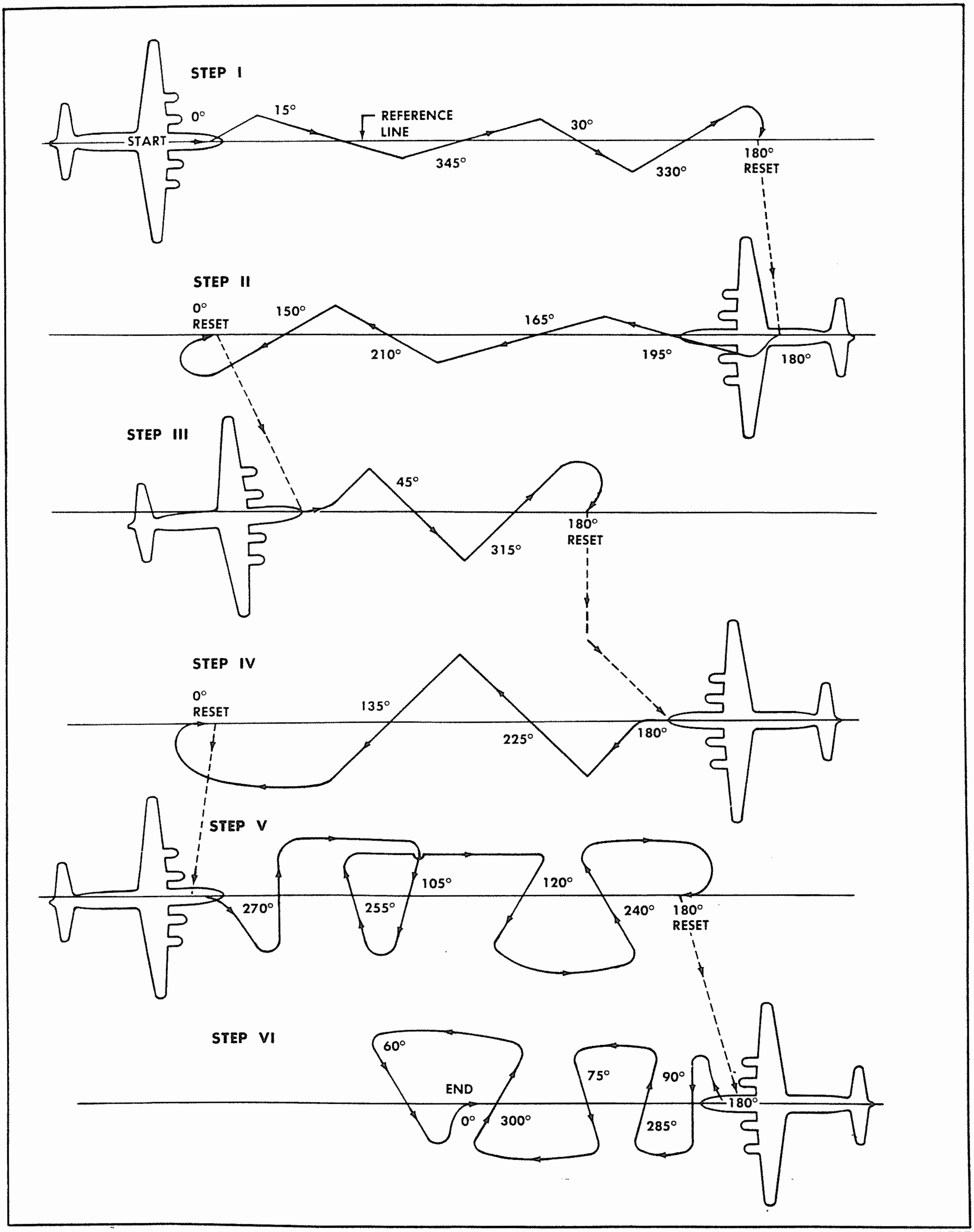


Figure 8-10. Radio Compass Deviation Calibration

Note:
Head toward station over predetermined point on reference line, steady, determine and record zero bearing.

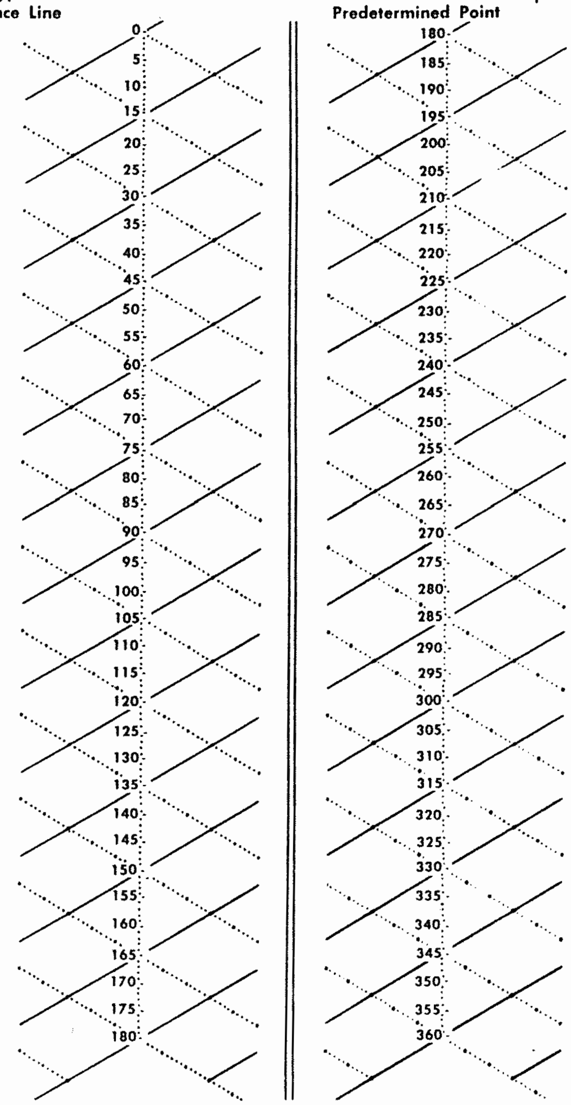
Station Used ....
Frequency ....
Plane No. ....
Pilot ....
Recorder ....

FLIGHT TEST DATA FOR CURVE

COLUMN 1	COLUMN 2	COLUMN 3	COLUMN 4	COLUMN 5
GYRO BEARING	AIRPLANE TO RADIO STATION BEARING	INDICATED BEARING	COMPENSATOR INNER SCALE ZERO BEARING	COMPENSATOR POINTER BEARING
0	0	0	0	0
15	345		15	
345	15		345	
30	330		30	
330	30		330	
180	180		45	
195	165		315	
165	195		60	
210	150		300	
150	210		75	
45	315		285	
315	45		90	
225	135		270	
135	225		105	
270	90		255	
105	255		120	
255	105		240	
120	240		135	
240	120		225	
90	270		150	
285	75		210	
75	285		165	
300	60		195	
60	300		180	

RADIO COMPASS DEVIATION CALIBRATION CURVE

Aircraft No. Pilot Date
Station Frequency kc. Receiver Type
Loop Type Receiver No. Loop No.
Reference Line Predetermined Point



RADIO COMPASS DEVIATION CALIBRATION CURVE

Figure 8-11. Reference Line Flight Procedure for Radio Compass Calibration

ing displaced 330 degrees, make a right turn, so that the aircraft is heading away from the transmitter and is flying directly over the reference line.

f. While flying away from the transmitter, check the directional gyro reading when the heading of the aircraft coincides with the reference line. The gyro reading should read within one degree of a displacement of 180 degrees, if all turns have been carefully made. If considerable precession of the directional gyro is noted when the 180-degree reference line course is checked, it is recommended that the calibration procedure be repeated, or the directional gyro checked if necessary.

g. Record the bearing indicator readings for the setting displaced 180 degrees from the zero heading, with the aircraft in level flight and on a steady course over the reference line.

h. Following a procedure similar to the one in steps a through g, obtain bearing indicator readings for gyro headings for displacements of 195, 165, 210, and 150 degrees from the zero heading. Turn the aircraft to head toward the transmitter along the reference line, establish a zero heading with respect to the reference line, and check the gyro reading in relation to the previously established zero heading.

i. Continue to fly to and from the transmitter over the reference line, alternating the heading of the aircraft from left to right until the bearing indicator readings have been obtained for every 15-degree change in heading of the aircraft. Record the data obtained on a form similar to the one shown in figure 8-11, for use in determining the corrections to be applied at the compensation mechanism of the loop.

j. If greater accuracy is required, use 5- or 10-degree increments instead of 15-degree increments for changes in heading. Do not make more than four alternate bearing checks between any two 0- and 180-degree gyro check points, unless it can be proved that more than four checks can be made without introducing excessive errors. If the reference line is not long enough, or if difficulty is experienced in checking the 0- and 180-degree gyro headings of the aircraft against the reference line after a series of turns, reduce the number of alternate right and left headings of the aircraft from four to two or even to one.

k. The procedure in steps a through j may be reduced to operation over a single point, such as a road intersection, provided that the bearing of the ground check point to the radio transmitter is accurately known. Fly a series of figure-eights, always crossing the ground point on a 15-degree change in gyro heading from the previous course, and read the bearing indicator when the aircraft is exactly over the ground point.

8-84. Perform Method 2 as follows:

a. Coincidence of the zero-degree radio bearing with a reference zero heading of the aircraft must be checked before performing Method 2. The coincidence can be checked on the ground or in the air by the procedures given in Method 1, or by the following:

b. Head the aircraft toward a transmitter having a vertical radiator which is clearly visible from all angles of approach. Fly directly into or with the existing winds to eliminate drift, if a location and period free from winds is not available.

c. Tune the radio compass receiver to the transmitter, and align the forward-and-aft axis of the aircraft with a radial line from the transmitting antenna as accurately as possible. Do not rely on visual heading checks from the flight compartment, since considerable error due to parallax may result. The use of a telescopic cross-hair sight, mounted in the flight compartment and accurately aligned with the axis of the aircraft, is recommended. If such equipment is not available, the heading may be aligned by sighting along the aircraft center line through the center of the windshield. Direct the pilot to change the heading until the forward-and-aft axis of the aircraft coincides with a line from the transmitter. Alignment observation should be made along the center line of the aircraft as far aft as possible from the center of the windshield in order to increase the accuracy of alignment. A two-inch deviation to the right or left from the center of the aircraft, as a distance of 10 feet from the center vertical line of the windshield, corresponds to an angle of one degree.

d. Note the bearing indicator pointer readings while the aircraft is held in level flight and headed directly toward the transmitting antenna. The bearing indicators should read zero (± 2) degrees, if all adjustments have been correctly made.

e. Approaching the transmitter from the opposite direction, repeat steps b through d preceding, at least once, to check the first reading.

f. Record the bearing indicator readings obtained for the zero heading of the aircraft.

g. The 180-degree radio bearing can be checked by holding the above-established course and flying over and past the transmitting antenna far enough to obtain steady indications.

h. After the zero bearing has been checked, a calibration flight procedure similar to that shown in figure 8-12 should be followed. Briefly, the procedure is to obtain bearing indicator readings for every 15-degree change in the gyro heading of the aircraft during the two 360-degree turns of equal diameter. The turns are made in opposite directions, and started and finished over a given point on the ground. The lengths of all 15-degree courses should be equal so that a circle is approximated. Two bearing indicator readings will be obtained for a given gyro reading after the two circles have been completed (one reading for each circle). The average of these two indicator readings is used in determining the compensator corrections.

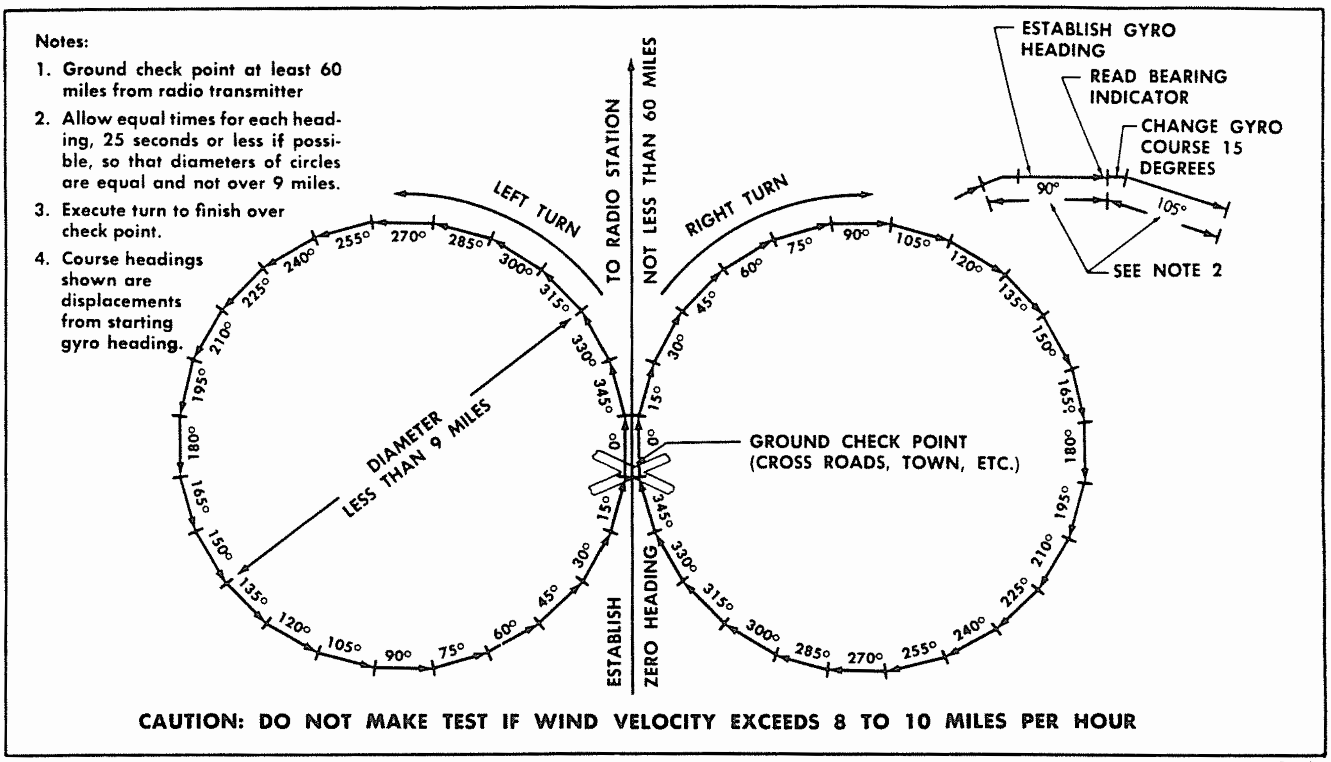


Figure 8-12. Figure-Eight Flight Procedure for Radio Compass Calibration

i. Locate a suitable ground check point, such as a road intersection or small town not less than 60 miles from the radio station. The station should provide good bearings with little or no fluctuation of the bearing indicators in the vicinity of the ground check point. Approach the ground check point from the side opposite the transmitter so that the aircraft heads directly toward the transmitter. Use the radio bearing indicators to establish this heading by orienting the aircraft until a radio bearing of zero degrees is obtained. If the results of the zero-heading check in steps a through g indicate that the zero heading of the aircraft does not coincide with the zero radio bearing within ±2 degrees, orient the aircraft so that the zero heading radio bearing obtained in steps a through g is indicated.

j. With the aircraft in steady, level flight, and on a heading of zero degrees with respect to the transmitter, record the reading of the directional gyro for use as a zero heading. The radio bearing indicator readings must be zero, or the same reading obtained in steps a through g, when the aircraft is directly over the ground check point.

k. Change the heading of the aircraft by a smooth and even left turn to establish a gyro heading displaced 345 degrees from the zero heading. After a steady heading has been attained with the aircraft in level flight, note and record the radio bearing indicator readings on a form similar to the one shown in figure 8-11. Not more than 20 to 25 seconds should be required for each heading after several trial headings have been made.

l. Decrease the gyro heading 15 degrees by a smooth turn. Note and record the radio bearing indicator readings after a steady and level heading has been established, and just before the turn is started for the next gyro heading.

m. Continue in this manner to obtain bearing indicator readings for each 15-degree change in gyro heading of the aircraft, always establishing steady course and level flight conditions before the bearing indicators are read. Record the readings for each course on a form similar to the one shown in figure 8-11.

n. The completion of the 15-degree gyro course should bring the aircraft directly over the ground check point, if the complete left turn has been made properly. After the radio bearing has been obtained for a 15-degree gyro heading, turn the aircraft for a zero-degree radio bearing. The aircraft should now be flying on the original starting line over the ground check point and headed directly toward the transmitter. Note and record the gyro reading for this heading of the aircraft. The gyro reading should check its original setting within one degree, if all turns have been made carefully.

o. If the gyro reading checks with its original zero heading, when the aircraft is over the ground check point and headed directly toward the transmitter, as indicated by the radio bearing indicators, make a right turn to establish a gyro heading displaced 15 degrees to begin the second half of the calibration procedure.

If the gyro reading does not check its original setting, establish a zero reference course.

p. Obtain radio bearing indicator readings for every 15-degree change in gyro heading of the aircraft by using a procedure similar to that in steps k through o, making a series of right turns instead of left turns. Make the complete 360-degree right turn so that the diameters of the two circles, approximated during left and right turns, are as nearly equal as possible. Record data obtained on a form, as shown in figure 8-11.

q. Take the average of each pair of radio bearing indicator readings for a given gyro heading. These averages are to be used for obtaining the corrections to be applied on the loop compensator unit.

r. The accuracy of the calibration procedure depends on the diameter of the two circles and the distance from the ground check point to the radio station, if other errors due to flight conditions and observational errors are neglected. The ground check point should be as remote as possible from the radio station, consistent with good radio bearing indications, and the diameter of the two circles should be equal and as small as possible. For example, a maximum error of 0.5 degree is introduced on some courses if the ground check point is 90 miles from the radio station, and if two circles are 8.5 miles in diameter. This error would be increased to approximately 2.0 degrees for a distance of 30 miles and circle diameters of 8.5 miles. Therefore, it is recommended that the ground check point be at least 60 miles from the radio station and the diameters of the circles less than 9 miles.

8-85. DETERMINATION OF COMPENSATION DATA. *(See figure 8-11.)* After the radio compass receiver deviation has been determined, plot the data and interpolate from the resulting curve, as follows:

a. Lay a straightedge parallel to the dotted line *(see figure 8-11)*, and through the chosen points of Column 3 draw a fine line. The point at which this line intersects the solid 15-degree line (Column 2) is the plot point. For a true bearing of 15 degrees (Column 2), for example, the indicated bearing is 7 degrees (Column 3).

b. Repeat step a for each of the 24 15-degree positions.

c. Draw a smooth curve through the plotted points to form the deviation curve *(see figures 8-13 and 8-14)*.

d. Determine the values for Column 5 from the deviation curve by drawing fine lines *(see figures 8-13 and 8-14)*, parallel to the solid lines between the intersections *(see figures 8-13 and 8-14)* of the plotted deviation curve and the dotted lines to the vertical graduations.

e. In Column 5, record values for the points of intersection *(see figures 8-13 and 8-14)* as read on the vertical graduations beside the 15-degree dotted line values in Column 4. For example, to determine the corrected pointer bearings for the loop position of 60 degrees (Column 4), lay the straightedge parallel to the solid line and draw a fine line through the intersection of the dotted 60-degree line and the deviation curve *(see figures 8-13 and 8-14)*. This line passes through the graduations of 73 degrees. The bearing value is recorded in Column 5. Similarly, a bearing of 105 degrees from Column 4 gives a bearing of 100 degrees for Column 5.

8-86. ADJUSTMENT OF COMPENSATOR SCREWS.
(See figures 8-13 and 8-14.)

a. Remove the eight screws from the loop base and remove the loop assembly.

b. Remove the bottom cover plate.

c. Take out the four screws to disconnect the four connector lugs from the terminal board on the compensator assembly.

d. Remove the three fillister-head mounting screws, and lift the compensator assembly from the loop base casting.

e. Check that no correction is set up on the compensator; however, if a correction is set up, set the adjusting screws to zero compensator and calibrate again.

f. Lay the compensator assembly with the Azimuth scale up. Check to see that the correct Azimuth scale is in place (use red scale for this installation).

g. If a temporary clamp on the compensator holds the pointer at zero, remove and discard the clamp.

h. Before adjusting the screws, remove the wrench from its helical spring mounting on the side of the compensator assembly.

i. Set the zero correction mark, on the inner dial opposite the degree marking on the Azimuth scale, to correspond with the heading in Column 4, figures 8-13 and 8-14. With the wrench from the compensator assembly, adjust the compensator screw, opposite the degree marking on the scale, until the pointer reads the correct value shown in Column 5.

j. It is usually necessary to make the complete compensator adjustment in a series of from three to five cycles. To avoid permanently damaging the cam strip in the compensator, do not let the adjustment of individual screws exceed the adjustment of adjacent screws by more than five degrees. When all screws have been adjusted to this extent, repeat the process until the total compensation has been set up. Some compensation curves may require a variation of more than five degrees of the adjustment screws. The larger variations should be set up last.

k. Replace the setscrew wrench.

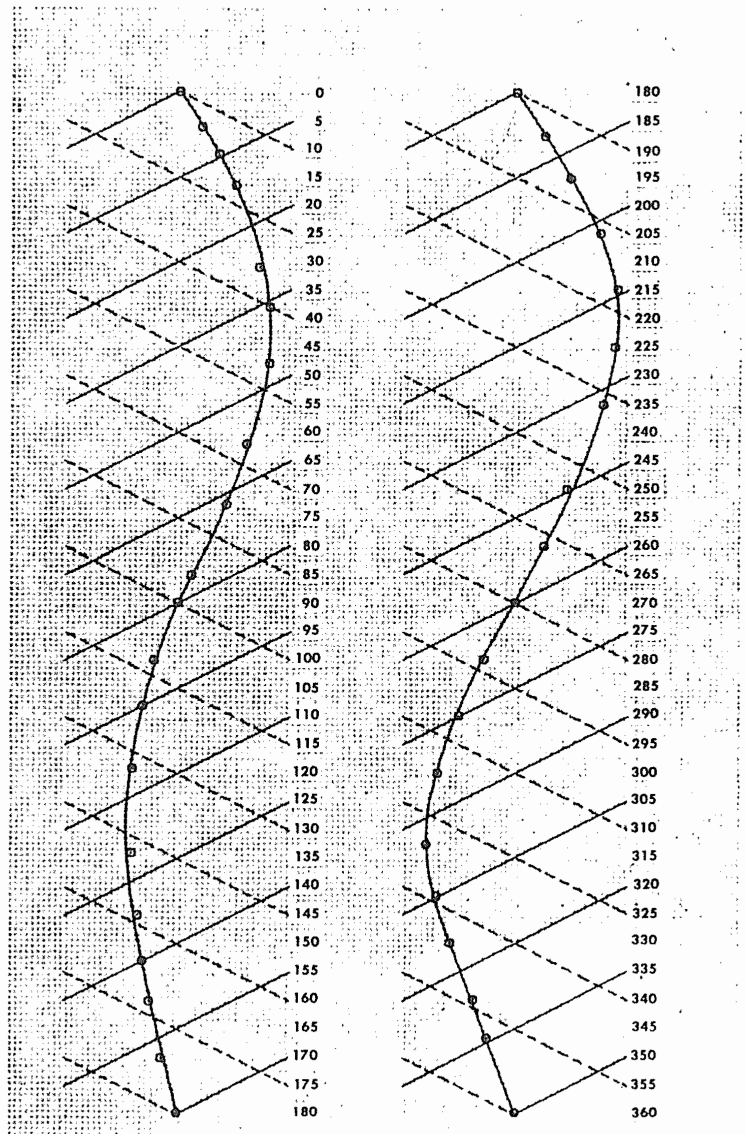
l. Make certain that the coupling pin on the compensator enters the coupling slot in the adjustment plate on the loop gear; reassemble the compensator assembly into the loop assembly.

m. Install three mounting screws.

FLIGHT TEST DATA FOR CURVE

COLUMN 1	COLUMN 2	COLUMN 3	COLUMN 4	COLUMN 5
GYRO BEARING	AIRCRAFT TO RADIO STATION BEARING	INDICATED BEARING	COMPENSATOR INNER SCALE ZERO BEARING	COMPENSATOR POINTER BEARING
0	0	0	0	0
15	345	349.5	15	27
30	330	339.5	30	46
45	315	329	45	60.5
60	300	315.5	60	71.5
75	285	295	75	81
90	270	270	90	90
105	255	245	105	100
120	240	223	120	112.5
135	225	208	135	126
150	210	196	150	142
165	195	187	165	161
180	180	180	180	180
195	165	168	195	213
210	150	156	210	228
225	135	143.5	225	241
240	120	128.5	240	252
255	105	111	255	261
270	90	90	270	270
285	75	65	285	278.5
300	60	45	300	288
315	45	29	315	300
330	30	17	330	316
345	15	7.5	345	338

RADIO COMPASS DEVIATION CALIBRATION CURVE



RADIO COMPASS DEVIATION CALIBRATION CURVE

Figure 8-13. Radio Compass Calibration Correction and Curve — Pilot's Red Compass

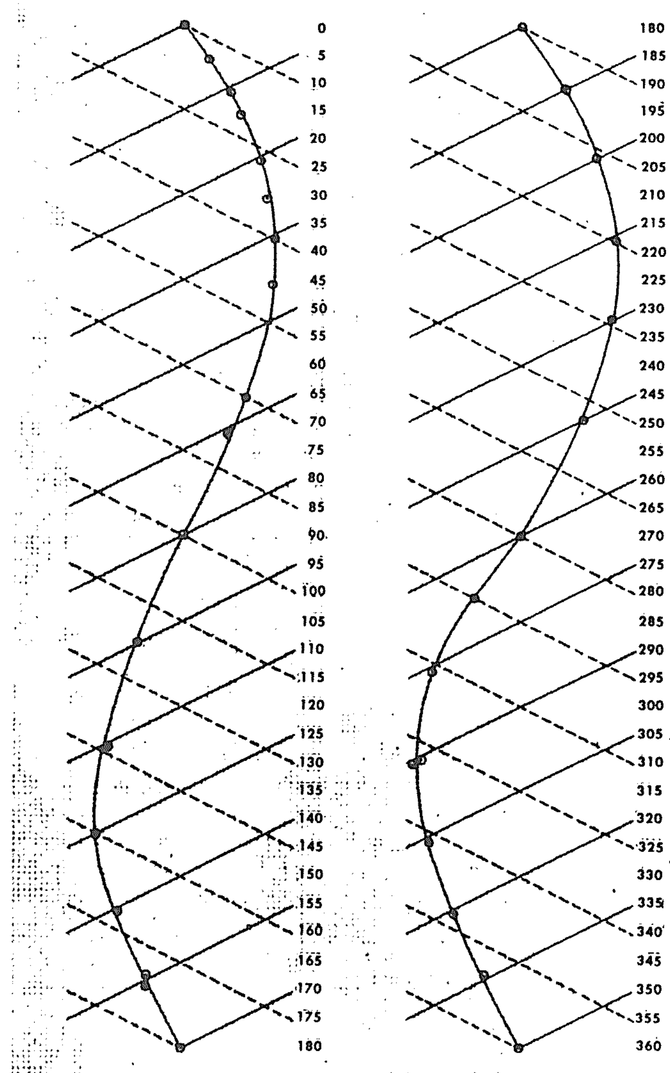
FLIGHT TEST DATA FOR CURVE

COLUMN 1	COLUMN 2	COLUMN 3	COLUMN 4	COLUMN 5
GYRO BEARING	AIRCRAFT TO RADIO STATION BEARING	INDICATED BEARING	COMPENSATOR INNER SCALE ZERO BEARING	COMPENSATOR POINTER BEARING
0	0	0	0	0
15	345	351.5	15	33
30	330	342	30	46
45	315	331	45	60
60	300	317.5	60	71
75	285	300.5	75	80.5
90	270	270	90	90
105	255	244	105	100
120	240	224	120	110
135	225	209	135	121.5
150	210	196.5	150	135
165	195	187.5	165	156
180	180	180	180	180
195	165	171	195	208
210	150	161	210	226.5
225	135	150	225	241
240	120	133	240	252
255	105	112	255	261.5
270	90	90	270	270
285	75	65.5	285	278
300	60	45	300	284.5
315	45	29	315	297
330	30	16.5	330	314
345	15	7	345	334.5

RADIO COMPASS DEVIATION CALIBRATION CURVE

Note:

This is the actual correction used on all R4D-8 aircraft for the co-pilot's (Green) compass, taken from test flight data.



RADIO COMPASS DEVIATION CALIBRATION CURVE

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Figure 8-14. Radio Compass Calibration Correction and Curve — Co-Pilot's Green Compass

n. Connect four terminals.

o. After checking that the Neoprene gaskets are in place, install the bottom cover plate. Apply Glyptal cement to the screws holding the cover plate, and secure the cover plate to the base casting.

p. Mount the loop by first attaching the two transmission cables and then securing the loop assembly with the eight mounting screws.

8-87. LONG RANGE NAVIGATION (LORAN) EQUIPMENT. (Partial provisions only.) The long range navigation set (AN/APN-4) is used to determine the long range geographic position of the aircraft during flight. The equipment consists of the receiver-indicator (ID-6B/APN-4), located on a tilting rack above the radio operator's table; an antenna coupler (CU-92A/APN), located on the forward side of bulkhead station 97 above the receiver indicator; and a radio receiver (R-9B/APN-4), located on the shelf below the radio operator's table. The LORAN, the HF communication radio-receiver (BC-348), and the HF communication radio-receiver (R-105A/ARR-15) are all connected to the HF communication antenna by means of the antenna coupler (CU-92A/-APN). Power is supplied to the LORAN equipment from the aircraft's 115-volt a-c power system through a 5-ampere fuse on the inboard side of the main junction box.

8-88. REMOVAL OF LONG RANGE NAVIGATION (LORAN) EQUIPMENT.

a. Disconnect the power and antenna plugs.

b. Loosen the two knurled thumb nuts on the front of the shock mount frame.

c. Slide the receiver or indicator forward and lift free of the mount.

8-89. INSTALLATION OF LONG RANGE NAVIGATION (LORAN) EQUIPMENT. Reverse the removal procedure *(see figure 10-48).*

8-90. VOR NAVIGATION EQUIPMENT. (Complete provisions.) The VOR navigation equipment (AN/ARN-14A) employs a radio receiver (R-252C/ARN-14) and a dynamotor unit (DY-84/ARN-14A). The radio receiver and dynamotor unit are located in the main radio rack and are used for VOR and VAR navigation and localizer position indications. One course indicator (ID-249A/ARN) and one course indicator (ID-250/ARN) are installed on the pilot's side of the instrument panel; one course indicator (ID-251/ARN) is installed on the top shelf on the main radio rack for the navigator. The course indicators are used in conjunction with the G-2 compass to select the degree of track or course desired during flight. In addition to the course indicators, a deviation indicator (ID-48/ARN) is installed on the co-pilot's side of the instrument panel. The vertical pointer of the deviation indicator is used in conjunction with the VOR navigation equipment. The VOR navigation equipment is controlled by knobs located on the pilot's overhead radio remote control panel. A range filter for the VOR navigation equipment is installed at each of the pilot's, co-pilot's, and radio operator's audio control boxes. Electrical power for the navigation equipment is supplied by both the 115-volt a-c and 28-volt d-c power. The a-c power is supplied through a 1-ampere fuse located on the inboard post of the main radio rack, and is transformed to 26-volt a-c power for use with the VOR navigation equipment. The 28-volt d-c electrical power is supplied through a 10-ampere circuit breaker located on the main circuit breaker panel. Normally, 115-volt a-c power is supplied for the VOR navigation equipment by the 1500 va inverter. If the 1500 va inverter should fail to operate, power is then supplied by the 250 va inverter, when the pilot's instrument inverter switch is transferred to the auxiliary position.

8-91. VOR NAVIGATION RADIO RECEIVER (R-252/ARN-14).

8-92. REMOVAL OF VOR NAVIGATION RADIO RECEIVER.

a. Remove the antenna coaxial lead-in from the radio receiver (R-252C/ARN-14) located on the second shelf of the main radio rack.

b. Loosen the clamps at the base of the receiver, and slide the receiver inboard to remove the receiver from its mount.

8-93. INSTALLATION OF VOR NAVIGATION RADIO RECEIVER. Reverse the removal procedure.

8-94. VOR NAVIGATION DYNAMOTOR UNIT (DY-84/ARN-14).

8-95. REMOVAL OF VOR NAVIGATION DYNAMOTOR UNIT.

a. Remove the electrical disconnect plug from the dynamotor unit located on the third shelf of the radio rack.

b. Loosen the clamps at the inboard base of the dynamotor unit to slide the unit inboard, and remove the unit from the radio rack.

8-96. INSTALLATION OF VOR NAVIGATION DYNAMOTOR UNIT. Reverse the removal procedure *(see figure 10-57).*

8-97. INSTRUMENT LANDING EQUIPMENT RECEIVING ANTENNA. *(See figure 8-4.)* (Mounting provisions only.) One antenna is provided for the operation of both the VOR and localizer receivers. The antenna is installed on top of the fuselage, aft of station 47.

8-98. REMOVAL OF INSTRUMENT LANDING EQUIPMENT RECEIVING ANTENNA. (When installed.)

a. Disconnect the antenna coaxial plug.

b. Remove the six attach screws from the bottom of the antenna mast and the top of the fuselage plating.

c. Remove the antenna assembly and gasket from the top of the fuselage.

8-99. INSTALLATION OF INSTRUMENT LANDING EQUIPMENT RECEIVING ANTENNA. Reverse the removal procedure.

8-100. GLIDE PATH ANTENNA. *(See figure 8-4.)* A glide path antenna is installed on the upper nose cap of the aircraft at station —39; a coaxial cable connects it to the radio rack where the end of the cable is stowed for possible future use.

8-101. REMOVAL OF GLIDE PATH ANTENNA.

a. Remove the coaxial feeder from the glide path antenna.

b. Remove the six screws, washers, and nuts in the base of the glide path antenna and nose cap.

8-102. INSTALLATION OF GLIDE PATH ANTENNA. Reverse the removal procedure.

8-103. RADIO ALTIMETER. The radio altimeter equipment (AN/APN-1) provides direct measurement of absolute altitude during flight from zero to 4000 feet. Operation of the equipment is based on the electric measurement of the time required for a radio signal transmitted from the aircraft to travel to earth from the transmitting antenna and return to the aircraft through the receiving antenna, the time being proportional to the altitude of the aircraft above the earth's surface. To accomplish this, a frequency-modulated signal is delivered to both the transmitting dipole antenna and the balanced detector in the receiver section of the transmitter-receiver unit. The transmitted signal is radiated earthward and, upon being reflected back, is received by a second dipole receiving antenna. The balanced detector in the receiving section combines the reflected signal with the direct transmitted signal to produce a low-frequency signal, the frequency of which is proportional to the altitude of the aircraft above the reflecting surface. This signal is suitably transformed to actuate the altitude indicator, located on the left side of the pilot's instrument panel. The altitude indicator (ID-14/APN-1) is calibrated in feet, and operates on a low or high scale selected by the range switch located on the altitude indicator. The low setting of the range switch selects a scale of 0 to 400 feet on the indicator. The high setting of the range switch operates the indicator on a scale of 0 to 4000 feet. An SA-1/ARN-1 altitude limit switch is located on the pilot's instrument panel below the radio altimeter indicator. A predetermined minimum altitude setting may be selected by the switch (SA-1/ARN-1). During flight, the red indicator light located on the pilot's instrument panel will be ON when the terrain is below the minimum altitude setting on the switch. The low limit indicator light is connected to the dimming relay assembly in the main electrical junction box. Electrical power for the radio altimeter is supplied from the 24-volt d-c circuit through a 10-ampere circuit breaker located on the circuit breaker panel.

8-104. RADIO ALTIMETER TRANSMITTER-RECEIVER (RT-7 APN-1).

8-105. REMOVAL OF RADIO ALTIMETER TRANSMITTER-RECEIVER.

a. Remove the two antenna coaxial feeders and electrical disconnect plugs from the unit, located in the tail section of the aircraft on the right side.

b. Loosen the two slide fasteners at the base of the unit.

8-106. INSTALLATION OF RADIO ALTIMETER TRANSMITTER-RECEIVER. Reverse the removal procedure *(see figure 10-39).*

8-107. RADIO ALTIMETER TRANSMITTER AND RECEIVER ANTENNAS. *(See figure 8-4.)* Two radio altimeter antennas are installed on the bottom center line of the fuselage. The forward antenna, located forward of station 492, is the transmitter antenna. The aft antenna, located forward of station 583, is the receiver antenna.

8-108. REMOVAL OF RADIO ALTIMETER TRANSMITTER AND RECEIVER ANTENNAS.

a. Remove the four attaching screws from the antenna and the bottom of the fuselage plating.

b. Remove the coaxial feeder from the top of the antenna mast.

8-109. INSTALLATION OF RADIO ALTIMETER TRANSMITTER AND RECEIVER ANTENNAS. Make certain that the skin of the aircraft and the antenna surfaces are clean before installation, and then reverse the removal procedure.

8-110. IFF EQUIPMENT. The IFF equipment (AN/APX-6) is used to identify the aircraft, in which it is installed, as friendly when correctly challenged by an interrogator-responsor. It also permits surface tracking and control of the aircraft in which it is installed. Functionally, the equipment receives challenges which are initiated by an interrogator-responsor, and transmits replies back to the interrogator-responsor where the replies are displayed on the indicators along with the associated targets. When a target is accompanied by a proper IFF reply, the target is considered friendly. Provisions are also incorporated for destroying the IFF equipment to prevent intact capture by the enemy. The IFF receiver-transmitter (RT-82/APX-6) is located on the right radio equipment shelf in the tail section of the aircraft. It is controlled by the pilot from the control panel located on the forward side of the partition, station 47, aft of the pilot. The IFF equipment is energized both by the 28-volt d-c circuit through a 5-ampere circuit breaker on the circuit breaker panel, and by the 115-volt a-c circuit through a 5-ampere fuse located on the inboard post of the main junction box. A 10-ampere destructor circuit breaker is located on the outboard side of the radio junction box.

8-111. IFF CONTROL. The IFF control unit (C-554/APX-6) is located aft of the pilot on the forward side

of bulkhead station 47. The control unit incorporates four switches operated as follows:

a. To turn the equipment on, rotate the MASTER selector NORM.

b. To explode the destructors within the equipment, raise the switch guard labeled DESTRUCT and raise the switch handle to the ON position.

WARNING

Do not fire destructors unless the AN/APX-6 is in danger of falling into enemy hands. When in doubt about the security of the area where a forced landing takes place, fire the destructors. Destructors will be fired if the destruct switch is turned on or if the impact switch is tripped, regardless of the setting of the master selector. There is no impact switch installation on aircraft C, D, 89 and subsequent.

8-112. REMOVAL OF IFF CONTROL. When preparing to remove the control unit, make sure that all electrical power is OFF.

a. Unfasten the four Dzus mounting attachments.

b. Pull the set control forward from the partition, station 47, and remove the electrical disconnect plug from the unit.

8-113. INSTALLATION OF IFF CONTROL. Reverse the removal procedure *(see figure 10-47)*.

8-114. INSTALLATION OF DETONATOR FOR RECEIVER-TRANSMITTER.

CAUTION

The detonator in the receiver-transmitter is to be installed only by qualified ordnance personnel using BuOrd procedures.

8-115. REMOVAL OF RECEIVER-TRANSMITTER. When preparing to remove the receiver-transmitter, make sure that all electrical power is OFF.

a. Disconnect the antenna coaxial feeder and the three electrical disconnect plugs from the unit.

b. Unscrew the two clamping collars at the base of the unit, until they can be disengaged from the hold-down lugs, and allow the collar to drop clear.

c. Pull the receiver-transmitter slightly aft to disengage the tapered plungers in the rear of the unit; then lift the unit clear of the mounting base.

8-116. INSTALLATION OF RECEIVER-TRANSMITTER. Reverse the removal procedure *(see figure 10-47)*.

8-117. IFF RADIO ANTENNA. *(See figure 8-4.)* The IFF (Identification Friend or Foe) radio antenna is installed on the bottom of the fuselage, forward of station 450. The lead-in coaxial cable from the antenna terminates at the IFF transmitter-receiver which is located on the radio equipment shelf on the right side of the tail section between station 568 and station 583.

8-118. REMOVAL OF IFF RADIO ANTENNA.

a. Remove the lead-in coupler from the top of the antenna by gaining access through the fuselage floor.

b. Remove the eight attaching screws in the base of the antenna mast and the bottom of the fuselage.

8-119. INSTALLATION OF IFF RADIO ANTENNA. Make certain that the skin of the aircraft and the antenna surfaces are clean before installation, and then reverse the removal procedure.

8-120. INTERPHONE AND RADIO CONTROL EQUIPMENT. The interphone and radio control or audio selector panels (AN/AIA-2A) are located under both the pilot's and co-pilot's side windows and on the right side of the fuselage at the radio operator's station. In addition to the three audio selector panels for the pilot, co-pilot, and radio operator, there is an additional interphone control box for the navigator *(see figure 8-3)*, located on the left side of the fuselage aft of station 47, and a cabin station located on the bulkhead aft of the main cargo door. Each of the three audio selector panels for the pilot, co-pilot, and radio operator are edge-lighted. The navigator's interphone and cabin interphone control boxes are not edge-lighted, as the only control necessary for the navigator and cabin stations is the volume and selector control for the interphone communication "crew" and interphone communication "all" stations. Each control box contains jacks for a microphone and headset; also, provisions are made for emergency operation of only one receiver at a time. Nine toggle switches permit any combination of interphone or receiver listening desired on the pilot's, co-pilot's, and radio operator's audio selector panels. A range filter control is provided on each audio control box to permit the reception of either voice or range signals, or both, on the range and VOR receivers only *(see figure 8-6)*. The range filters are located in the interphone control junction box, located on the right side of the fuselage below the radio operator's audio control panel. The navigator's and cabin area audio control boxes are identical, and contain a volume control and a control to select either the crew or all ICS. Microphone and headset jacks are located on the bottom of each of the two audio control boxes. The pilot's, co-pilot's, and radio operator's audio control boxes are connected directly to the interphone-radio control junction box. The volume control on these boxes is inoperative on the marker beacon and the RED ADF receivers *(see figure 8-6)*. Both the navigator's and the cabin area audio control boxes are connected to the interphone-radio control junction box through the main radio junction box. Connection is

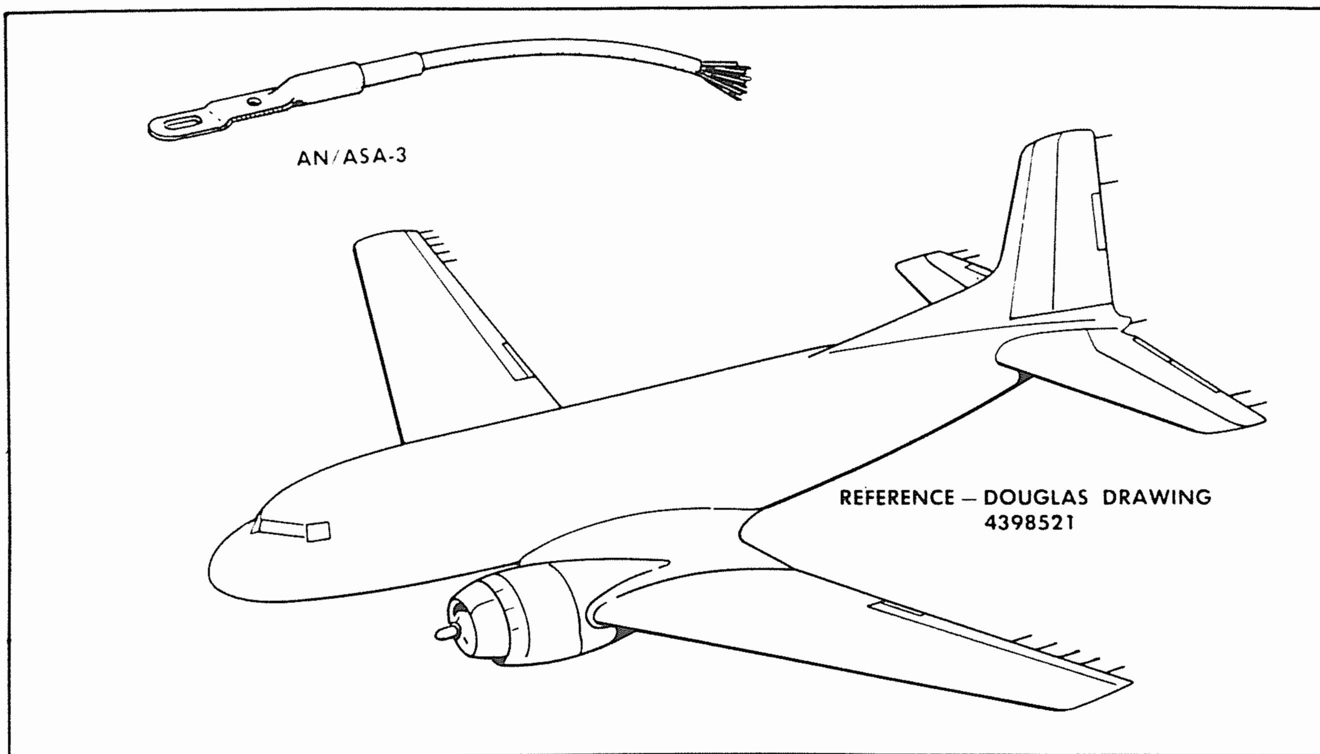

Figure 8-15. Static Dischargers

also made from the interphone-radio junction box to a dynamotor (DY-13A/AIA-2A), located below the radio operator's table and to the ICS amplifier (AM-40/AIC) located on the shelf below the radio operator's table. The dynamotor supplies 250 volts d-c for the interphone and radio control panel. The amplifier (AM-40/AIC) provides the interphone communication between the main cabin and the crew stations.

8-121. CABIN INTERPHONE AND PUBLIC ADDRESS SYSTEM. On aircraft A and B, additional electronic equipment is installed, consisting of an entertainment radio, six cabin speaker assemblies, one steward's speaker assembly, a handset for passenger use and one for the steward, and an additional audio amplifier, all of which are electrically connected to an additional radio junction box located in the main cabin on the forward right side of fuselage, station 465. In addition to the foregoing cabin electronics, one additional handset and public address control switch is located in the pilot's compartment. The entertainment radio can be utilized for air traffic audio from the pilot's interphone system *(see figure 10-40).*

8-122. STATIC DISCHARGERS. *(See figure 8-15.)* Seventeen static dischargers (AN/ASA-3) are installed in the following positions: five dischargers on the outer trailing edge of both left and right ailerons, two on the outer trailing edge of each elevator, two on the upper trailing edge of the rudder, and one on the aft end of the tail cone assembly above the red and white navigation lights. Each static discharger consists of a wick, mounted in a plastic tube, which is connected to the surface of the aircraft with screws. The static dischargers tend to reduce the static interference to a minimum by providing a high-resistance discharge path for accumulated static electricity, thereby improving the operation of the radio, navigation, and electronic equipment.

8-123. SERVICING STATIC DISCHARGERS. In order to maintain the dischargers at their highest efficiency, the dischargers should be trimmed at each No. 2 service (100-hour inspection). This is necessary because rain washes out the silver conducting particles from the wick end. Trim the static dischargers as follows:

a. Cut the wick off even with the trailing end of the plastic tube.

b. Cut and remove the plastic tube to expose a minimum of one inch of the wick. Use caution in cutting to prevent damage to the wick.

c. Fan out the newly exposed wick to provide the proper discharge effect.

d. Replace the assembly with a new static discharger when the total length of the wick assembly (tube and wick) is less than six inches long, or if the wick becomes soaked with oil.

8-124. REMOVAL OF STATIC DISCHARGERS. Remove two screws from the discharger mounting bracket; remove the discharger.

8-125. INSTALLATION OF STATIC DISCHARGERS. Reverse the removal procedure.

AN 01-40NK-2

Handbook

Maintenance Instructions

NAVY MODELS

R4D-8, R4D-8Z AIRCRAFT

SECTION IX ARMAMENT AND PHOTOGRAPHIC EQUIPMENT

THIS SECTION SUPERSEDES SECTION IX OF AN 01-40NK-2
DATED 15 MAY 1952

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AND THE CHIEF OF THE BUREAU OF AERONAUTICS

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SECTION IX

ARMAMENT AND PHOTOGRAPHIC EQUIPMENT

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NAVY MODELS

R4D-8, R4D-8Z
AIRCRAFT

SECTION X
WIRING DATA

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SECTION X

WIRING DATA

TABLE OF CONTENTS

SECTION X

WIRING DATA

10-1. WIRING DATA.

10-2. DESCRIPTION. The purpose of the wiring data is to furnish aircraft maintenance personnel with information on the wiring of electrical and electronic equipment. The section contains information on wiring and wire specifications, a chart of wiring symbols, an electrical equipment list, an alphabetical index of the wiring diagrams, and the wiring diagrams. To locate a specific wiring diagram, turn to the wiring diagram index and locate the diagram alphabetically. The wiring diagrams should be referred to, in conjunction with the text on the specific system on which maintenance work is being performed, during trouble shooting, maintenance, repair, and check-out procedures of all electrical and electronic systems. Wires are identified on the diagrams by code letters and numbers which are the same as those used on the aircraft.

10-3. WIRING. Wire sizes in the electrical system range from No. 00 to No. 20. The size of an electrical wire decreases as the gage number increases; No. 00 is the largest wire and No. 20 is the smallest.

10-4. WIRE IDENTIFICATION. The number and letter system for electrical wire designation in the wiring diagram book in each aircraft has been designed to expedite electrical servicing. A typical wire number is 1P33A18. The prefix 1 indicates engine No. 1 or left unit of operation (the number 2 indicates the right engine or unit). The letter P indicates the circuit classification (see table following). The number 33 between the letters P and A indicate the basic wire number, as shown on the schematic and circuit diagrams. The letter A indicates the segment of wire run from point to point, as shown on the wiring diagrams. The number 18 following the letter A indicates the wire size. Certain wire installations that are neither left nor right (such as heater circuit wires in the fuselage) will not have a number prefix. Wires that are installed forward of the firewall will not have the number prefix, as the engine installations are interchangeable. The following table identifies the circuits and their letter symbols:

Automatic Pilot	C
De-Icing	D
D-C Power	P
Electronics	R
Engine Controls	K
Engine Instruments	E
Flight Instruments	F
Fuel Pumps	Q
Ground Wires	N
Heating	H
Ignition and Starter	J
Inverter Control	V
Lighting	L
Metering	M
Warning Devices	W

Single-letter prefixes are used throughout the system, except when a wire within a circuit is a ground wire, or in certain radio circuits. For example, if a wire is installed in a warning circuit, the letter prefix W is used. However, if the wire within this warning circuit is a ground wire, an N is added to the wire designation ahead of the basic letter prefix so that the wire number may become NW1A20. Wires distributing 70 volts or over are marked H.V.

10-5. WIRE SPECIFICATIONS. Shielded wire is used in the electrical system to reduce electrostatic radio disturbances. High-temperature wire is installed in the engine section and in emergency circuits where wire must withstand higher temperatures. High-voltage wire is installed in all high-tension circuits to prevent voltage breakdowns.

Unshielded Wire:	
Single-Conductor Copper	DES-SM2006
High-Temperature	Rockbestos (Fire Zone)
High-Temperature	G. E. 57351
Thermocouple	Specification No. AN-W-8, Type II, Class "A" and Lewis LW1-9
Shielded Wire:	
Single-Conductor	DES-SM2006-1S
Multiple-Conductor	DES-SM2006-2S or DES-SM2006-3S
High-Voltage Wire	Specification No. AN-J-C-56

10-6. TERMINAL CONNECTORS. Wire junctions that do not terminate in pin-and-socket type connector

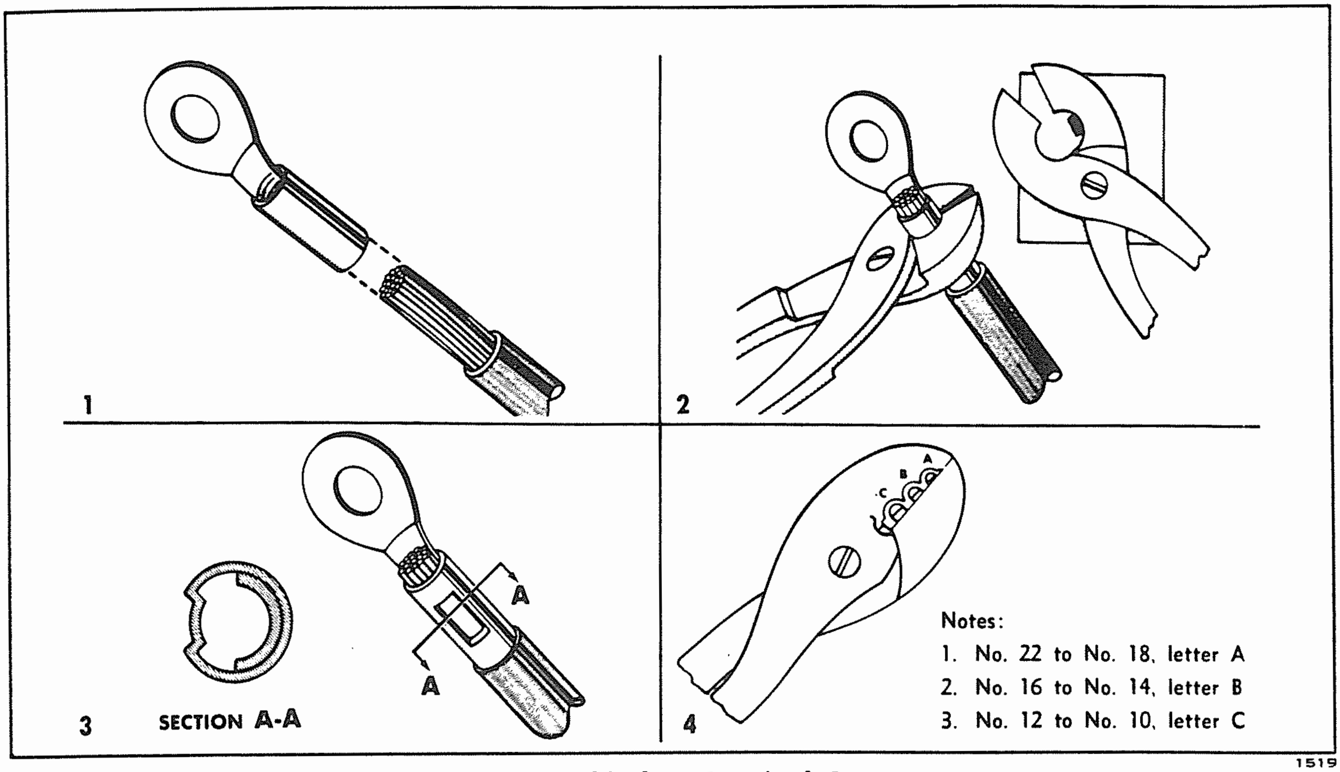

Figure 10-1. Solderless Terminal Connectors

plugs are lugged for fastening to the terminal strips with solderless connectors. Solderless connectors for all sizes of aircraft wire are available *(see figure 10-1).* The terminals are pre-tinned and are slipped on over the ends of the aircraft wire after the insulation has been removed, and then staked on by means of pressure applied by a special plier-type tool *(see figure 10-2).* This plier makes a small rectangular indenture in the side of the lug and clamps it firmly to the wire. This type of connection facilitates rapid repair and creates a low-resistance, mechanically secure connection.

10-7. ELECTRICAL SYMBOLS.

10-8. For a complete explanation of all graphic symbols used in the wiring diagrams, see figure 10-3.

10-9. ELECTRICAL AND ELECTRONIC EQUIPMENT LIST.

10-10. The electrical and electronic equipment list is arranged alphabetically, and lists the part name, the AN or vendor number, and the manufacturer of the part, where this information is necessary for proper identification of the part.

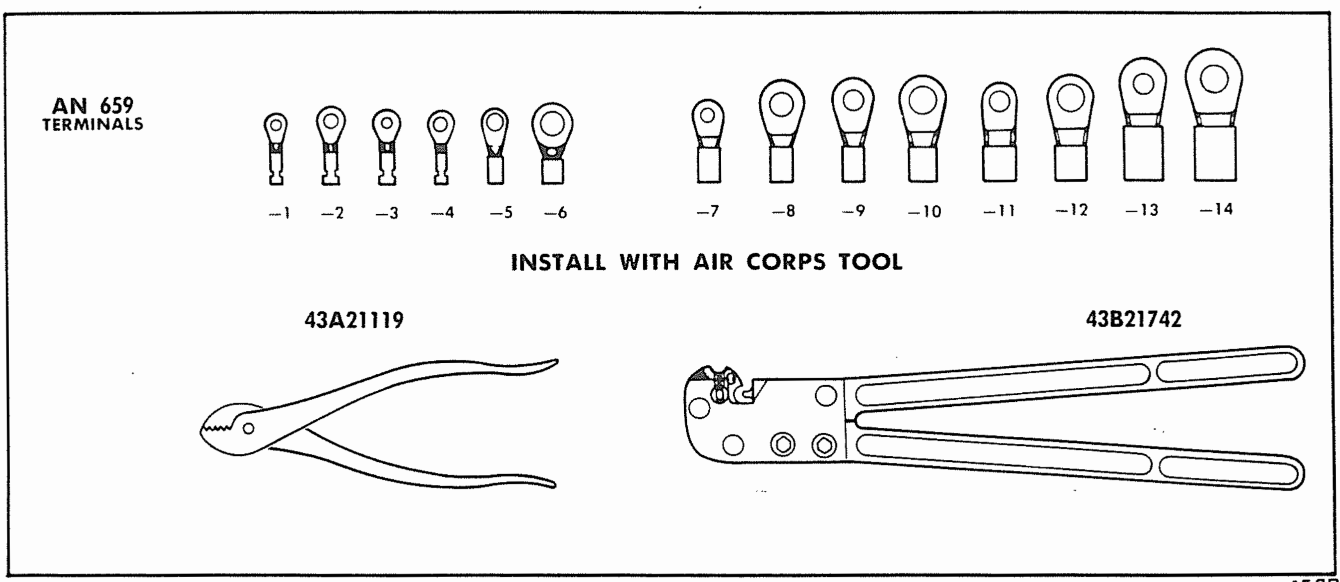

Figure 10-2. Terminal Connectors and Special Connector Tools

ELECTRIC AND ELECTRONIC EQUIPMENT LIST

Part Name	*AN or Vendor No.*	*Manufacturer*
Adapter—Automatic Pilot Amplifier	DR-19	Pioneer
Adapter—G2 Compass	MIL A5437	
	NAF Dwg 1300	BuAer
Ammeter—Main Generators	K8941850-G5	General Electric
Ammeter—Airscoop and Pitot Heater	K8941850-G2	General Electric
Amplifier—Automatic Pilot Channel	12317-1-D	Pioneer
Amplifier—G2 Compass	8KE1AC2	General Electric
Antenna—Altimeter	AT-4/APN-1	
Antenna—Glide Path	37P2	Collins
Antenna—HF Communication	5397012	Douglas
Antenna—IFF Communication	AS-133/APX	
Antenna—Loop (GREEN) ADF	LP-21	
Antenna—Loop (RED) ADF	LP-21	
Antenna—Marker Beacon	5397012	Douglas
Antenna—Marker Beacon	4460276	Douglas
Antenna—Range Receiver	5397012	Douglas
Antenna—Sense (GREEN) ADF	5397012	Douglas
Antenna—Sense (RED) ADF	5397012	Douglas
Antenna—UHF Communication	AT-141/ARC	
Antenna—VHF Communication	AN-104AX	
Antenna—VOR Receiver	AT-172/ARN-14	
Battery	6FHM-13-1	Exide
Bulb—Carburetor Air Temperature	AN5525-1	
Bulb—Free Air Temperature	AN5525-1	
Bulb—Oil Temperature	AN5525-1	
Box—Heater Fuel Control	94A42	Surface Combustion
Box—Power Junction (Automatic Pilot)	DR-21-A1	Pioneer
Cable—ADF Loop (GREEN)	CG-42/ARN-7	
Cable—ADF Loop (RED)	CG-42/ARN-7	
Cable—Altimeter Receiver Antenna	S-4391470-A	Douglas
Cable—Altimeter Transmitter Antenna	S-4391470-A	Douglas
Cable—ARR-15 Antenna	S-4391470-M	Douglas
Cable—ART-13 to CU92/APN	3357013-570	Douglas
Cable—BC-348 Antenna	3357013-572	Douglas
Cable—CU92/APN to Terminal Strip	3357013-574	Douglas
Cable—Glide Path Antenna	S-4391470-F	Douglas
Cable—IFF Antenna	S-4391470-K	Douglas
Cable—LORAN Antenna	S-4391470-J	Douglas
Cable—LORAN Gain	S-4391470-H	Douglas
Cable—LORAN Ampere Balance	S-4391470-H	Douglas
Cable—LORAN Interconnection	CO-239-D	
Cable—LORAN Video	S-4391470-H	Douglas
Cable—Marker Beacon Antenna	S-4391470-L	Douglas
Cable—Range Receiver Antenna	S-4391470-M148	Douglas
Cable—(GREEN) Sense Antenna	S-4391470-M184	Douglas
Cable—(RED) Sense Antenna	S-4391470-M184	Douglas
Cable—UHF Antenna	S-4391470-F	Douglas
Cable—VHF Antenna	S-4391470-A	Douglas
Cable—VOR Antenna	S-4391470	Douglas
Circuit Breaker—A-C Output 1500 va Inverter	AN3160-15	
Circuit Breaker—Alcohol Quantity	AN3160-5	
Circuit Breaker—APN-1 Altimeter	AN3160-10	
Circuit Breaker—ARC-27 UHF	AN3160-25	
Circuit Breaker—ARC-1 VHF	AN3160-15	
Circuit Breaker—ARR-15A Receiver	AN3160-10	
Circuit Breaker—ART-13 Transmitter	AN3160-35	
Circuit Breaker—BC-348 Receiver	AN3160-5	
Circuit Breaker—Cabin Dome Lights	AN3160-10	
Circuit Breaker—Co-Pilot's Turn-and-Bank	AN3160-5	

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Circuit Breaker—Co-Pilot's Upper Switch Panel and Compass Light	AN3160-5	
Circuit Breaker—Cowl Flaps	AN3160-15	
Circuit Breaker—De-Icer Equipment	AN3160-10	
Circuit Breaker—Door Warning	AN3160-5	
Circuit Breaker—Engine Fire Warning	AN3160-5	
Circuit Breaker—Engine Start and Ignition Boost	AN3160-10	
Circuit Breaker—Fuel Booster Pump	AN3160-15	
Circuit Breaker—Fuel and Oil Pressure Warning	AN3160-5	
Circuit Breaker—Fuel Quantity	AN3160-5	
Circuit Breaker—G2 Compass Adapter	AN3160-5	
Circuit Breaker—Generator Field	AM1614-15-28-2	Heineman
Circuit Breaker—Generator Warning Lights	AN3160-5	
Circuit Breaker—Glide Path Receiver	AN3160-5	
Circuit Breaker—(GREEN) ADF	AN3160-5	
Circuit Breaker—Ground Blower	9301-70-B	Square D
Circuit Breaker—Ground Power Control	AN3160-5	
Circuit Breaker—Heater Airscoop	9301-70-8	Square D
Circuit Breaker—Heater Control	AN3160-10	
Circuit Breaker—Heater Fire Warning	AN3160-5	
Circuit Breaker—Heater Ground Control	AN3160-10	
Circuit Breaker—ICS Amplifier	AN3160-5	
Circuit Breaker—IFF Destructor	AN3160-10	
Circuit Breaker—IFF Power	AN3160-5	
Circuit Breaker—Interphone Junction Box	AN3160-10	
Circuit Breaker—Inverter Warning Lights	AN3160-5	
Circuit Breaker—Alighting Gear Position Indicator	AN3160-5	
Circuit Breaker—Alighting Gear Warning	AN3160-10	
Circuit Breaker—Landing Lamp Position Control and Lamp Relay	AN3160-10	
Circuit Breaker—Landing Lights	AN3160-35	
Circuit Breaker—Leading Edge Lights	AN3160-10	
Circuit Breaker—Marker Beacon Receiver	AN3160-5	
Circuit Breaker—Miscellaneous Cockpit Lights and Automatic Warning Light Dimming Relay	AN3160-10	
Circuit Breaker—MJB Floodlight	AN3160-5	
Circuit Breaker—Navigation Lights	AN3160-10	
Circuit Breaker—Oil Cooler Doors	AN3160-15	
Circuit Breaker—P-1 Automatic Pilot	AN3160-10	
Circuit Breaker—Pilot and Center Instrument Panel Lights	AN3160-5	
Circuit Breaker—Pitot Heaters	AN3160-10	
Circuit Breaker—Propeller Feathering Control	AN3160-10	
Circuit Breaker—Radio Panel Lights	AN3160-5	
Circuit Breaker—Range Receiver	AN3160-5	
Circuit Breaker—(RED) ADF	AN3160-5	
Circuit Breaker—Tail Access and Lavatory Dome Lights	AN3160-5	
Circuit Breaker—Temperature Instrument	AN3160-5	
Circuit Breaker—1500 va Inverter Control	AN3160-10	
Circuit Breaker—250 va Inverter Feed	AN3160-25	
Circuit Breaker—1500 va Inverter Feed	9300-125-B	Square D
Circuit Breaker—VOR Receiver	AN3160-10	
Coils—Cabin Heater Ignition	61A78	Surface Combustion
Condensers—Automatic Warning Light Dimming Arc Suppression	430M .25X.25	Cornell Dubilier
Condensers—Cycle Switch Arc Suppression	DYR 6050	Aerovox
Condensers—Glide Path Pointer Damping	FA 1210	Cornell Dubilier
Condensers—Localizer Damping	3750 MFD (S17378)	Sprague
Condensers—Localizer Pointer Damping	FA1210	Cornell Dubilier
Condensers—VOR Damping	FA 1210	Cornell Dubilier
Condensers—VOR Damping	1250 MFD (S17377)	Sprague
Controller—Turn-and-Bank (Automatic Pilot)	15704-1-A	Pioneer
Detectors—Cabin Heater Compartment Area	17243-61-250F	Fenwall
Detectors—Cabin Heater Duct Fire	2363847	Douglas Fenwall
Detectors—Engine Section Fire	35534-4	Edison

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Filters—Altimeter Power	CP-53B-1EB-405K	
Filters—BC-348 Screen	3397710	Douglas
Filters—Ignition Noise	B105	Potter
Filters—Master Radio Feed Noise	Lewis RF8-100	Potter
Flashers—Navigation Light	FA-122	Wallace and Tiernan
Fuses—A-C Output 250 va Inverter	414002	Littlefuse
Fuses—Airscoop Meter Isolation	414003	Littlefuse
Fuses—Automatic Pilot	414003	Littlefuse
Fuses—Co-Pilot's Directional Gyro	414001	Littlefuse
Fuses—Co-Pilot's Horizon	414001	Littlefuse
Fuses—Drift Meter	414001	Littlefuse
Fuses—G2 Compass	414002	Littlefuse
Fuses—(GREEN) ADF	414003	Littlefuse
Fuses—Heater Drop-Out	414003	Littlefuse
Fuses—IFF	414005	Littlefuse
Fuses—LORAN	414005	Littlefuse
Fuses—Master Radio Power (Limiter)	FLLC 100A	Burndy
Fuses—Pilot's Horizon and Turn-and-Bank	414001	Littlefuse
Fuses—Radio Feed Master Limiter	FLLD 130A	Burndy
Fuses—Rear Bus Limiter	FLLE 150A	Burndy
Fuses—(RED) ADF	414003	Littlefuse
Fuses—Secondary Bus	FLLG 225A	Burndy
Fuses—Starter Limiter	FLLE 150A	Burndy
Fuses—VOR Instrumentation	414001	Littlefuse
Generators—Engine Tachometer	AN5531-2	
Generators—Generator Main	AN3633-1	
Heaters—Airscoop	5369622	Douglas
Heaters—Pitot	Type G1 AN5813-1	
Horns—Alighting Gear Warning	318	Edwards
Indicator—Alcohol Quantity	EA-100AN-93	Liquidometer
Indicator—Altitude Indicator	ID-14/APN-1	
Indicator—Automatic Pilot Master Indicator	8KD1AB2	General Electric
Indicator—Automatic Pilot (Horizon) Vertical Gyro	12402-1F-B1	Pioneer
Indicator—Automatic Pilot (Turn-and-Bank) Rate Gyro	15300-1D-1-C1	Pioneer
Indicator—Cabin Heater Temperature (2 Ohm, Iron-Constantan)	49B10	Lewis
Indicator—Carburetor Air Temperature	AN5795-6	
Indicator—Co-Pilot's ADF Dual Azimuth	MN-58D	Bendix
Indicator—Co-Pilot's Horizon	R881-1325	
Indicator—Co-Pilot's Turn-and-Bank	R88-I-3221	
Indicator—Course Indicator	ID-250/ARN	
Indicator—Course Indicator	ID-249/ARN	
Indicator—Deviation Indicator	ID-48/ARN	
Indicator—Directional Gyro	R88I-1006-20	
Indicator—Engine Cylinder Temperature	94-27971B Dual Type B11	
Indicator—Engine Tachometer	AN5530-2 Dual	
Indicator—Free Air Temperature	AN5790-6	
Indicator—Fuel Quantity Main	EA-48-5-24	
Indicator—Fuel Quantity Outer Wing	EA-711-C-5	Liquidometer
Indicator—Alighting Gear Position	R88-I-1888	Liquidometer
Indicator—Oil Temperature	AN5795-6	
Indicator—Pilot's ADF Dual Azimuth	MN-58D	Bendix
Indicator—Pitch and Trim	15100-1C-A1	Pioneer
Indicator—Radio Indicator Control	ID-251/ARN	
Inverters—1500 va Inverter	E1737-1	
Inverters—250 va Inverter	E1617-1	
Keys—R/O	NAF 213265-1	
Lamp Assemblies—Cabin Dome	HG-1001	Gem City
Lamp Assemblies—Co-Pilot's Flood	A4255A	Grimes

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Lamp Assemblies—Door Open Warning	AN3157-6	
Lamp Assemblies—Fire Warning	204BP (RED)	Korry
Lamp Assemblies—Fuel and Oil Pressure Warning	(RED) AN3157-6	
Lamp Assemblies—Gear Unsafe	AN3157-6	
Lamp Assemblies—General Cutout Warning	AN3157-6	
Lamp Assemblies—Glide Path Warning	AN3157-6	
Lamp Assemblies—Heater Inoperative Warning	AN3157-6	
Lamp Assemblies—Inverter Warning	AN3157-6	
Lamp Assemblies—Left Wing Tip Navigation	A1285-R-1524	Grimes
Lamp Assemblies—Localizer Warning	AN3157-6	
Lamp Assemblies—Main Junction Box	1252	Wemac
Lamp Assemblies—Pedestal Flood	A-4255A-2	Grimes
Lamp Assemblies—Pedestal Panel	1617C	Wemac
Lamp Assemblies—Pilot and Co-Pilot Cockpit	C4-A	
Lamp Assemblies—Pilot's Flood	A-4255A	Grimes
Lamp Assemblies—Radio Operator's and Navigator's Goose Neck	AN3047-3A	
Lamp Assemblies—Red Tail Light	A1943-4R-310	Grimes
Lamp Assemblies—Right Wing Tip Navigation	A1285-G-1524	Grimes
Lamp Assemblies—Tail Access and Lavatory Dome	HG-1001	Gem City
Lamp Assemblies—Top and Bottom Fuselage Navigation	B4580-24 (clear)	Grimes
Lamp Assemblies—Upper Switch Panel	1625C	Wemac
Lamp Assemblies—White Tail Light	A1943-4-310 (white)	Grimes
Lamps—Altimeter Low Limit	AN3121-313	
Lamps—Cabin Dome	307	Mazda
Lamps—Center Panel	327	Mazda
Lamps—Compass	323	Mazda
Lamps—Co-Pilot's Marker Beacon	AN3121-313	
Lamps—Co-Pilot's Panel	327	Mazda
Lamps—Co-Pilot's Panel Flood	303	Mazda
Lamps—Door Open Warning	313	Mazda
Lamps—Fire Warning	313	Mazda
Lamps—Fuel and Oil Pressure Warning	313	Mazda
Lamps—Gear Unsafe	313	Mazda
Lamps—General Cutout Warning	313	Mazda
Lamps—Heater Inoperative Warning	313	Mazda
Lamps—IFF Control Box	AN3140-327	
Lamps—Interphone-Radio Control Box	AN3140-327	
Lamps—Inverter Warning	313	Mazda
Lamps—Landing Light and Assembly	G3800A-5	Grimes
Lamps—Localizer Warning	313	Mazda
Lamps—MJB	1385	General Electric
Lamps—Navigation Wing Tip	1524	Grimes
Lamps—Overhead Radio Control	AN3140-327	
Lamps—Pedestal Flood	303	Mazda
Lamps—Pedestal Panel	327	Mazda
Lamps—Pilot's and Co-Pilot's Cockpit Area Lights	313	Mazda
Lamps—Pilot's Marker Beacon	AN3121-313	
Lamps—Pilot's Panel	327	Mazda
Lamps—Pilot's Panel Flood	303	Mazda
Lamps—Radio Operator's and Navigator's Goose Neck Light	307	Mazda
Lamps—Radio Panel	327	Mazda
Lamps—Range Receiver Control Box	AN3140-327	
Lamps—Tail Access and Lavatory Dome	307	Mazda
Lamps—Tail Light	310	Mazda
Lamps—Top and Bottom Fuselage Navigation	309	Mazda
Lamps—Upper Switch Panel	327	Mazda
Lamps—UHF Control Panel	AN3140-327	
Lamps—Wing Edge	4502	Mazda
Light Assemblies—Altimeter Low Limit	AN3157-6	
Motors—Alcohol Pump	20007-3	Adel
Motors—Boot De-Icer Valve	572-4	Eclipse
Motors—Cowl Flap (Power Unit Assembly)	29696	AiResearch

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Motors—Engine Starter	AN4116R4	
Motors—Fuel Booster Pump	608 PG	Pesco
Motors—Heater Ground Blower	7390196	Douglas
Motors—Oil Cooler Doors	30416	AiResearch
Motors—Propeller Feather Pump	IE-VR-280BHC	
Plugs—Adapter to Amplifier	AN3106-22-14S	
Plugs—Adapter to G2 Amplifier	AN3106-16S-1S	Cannon
Plugs—Adapter to Servos	AN3106-18-1S	
Plugs—Alcohol Pump Motor	AN3108-10S-2S	
Plugs—Alcohol Quantity Indicator	AN3106-14S-2S	
Plugs—Alcohol Quantity Liquidometer	AN3108-14S-1S	
Plugs—ALT (Battery Input)	AN3108B-18-5S	
Plugs—ALT (Indicator)	AN3108B-18-30P	
Plugs—ALT (IND)	AN3106B-18-30S	
Plugs—ALT (Limit Indicator)	AN3108B-18-20P	
Plugs—ALT (Limit Switch)	AN3108B-18-31P	
Plugs—ALT Limit Switch	AN3106B-18-20S	
Plugs—Amplifier to Amplifier Adapter	AN3106-20-16S	
Plugs—Amplifier to Compartment Transmitter	AN3106-14S-2P	Cannon
Plugs—Amplifier to Controller	AN3108B-20-11S	Cannon
Plugs—Amplifier to G2 Adapter	AN3106-14S-6S	Cannon
Plugs—Amplifier to Master Direction Indicator	AN3106-22-14P	Cannon
Plugs—Amplifier to Power Junction Box	AN3106-20-16S	Cannon
Plugs—APX-6 (CONTROL)	AN3106B-18-1P	
Plugs—APX-6 (DESTRUCTOR)	AN3108B-8S-1P	
Plugs—APX-6 (POWER)	AN3108B-14S-5S	
Plugs—ARR-15	AN3106B-12S-3S	
Plugs—ARR-15	Part of Rack	
Plugs—ART-13	AN3106B-18-12S	
Plugs—ART-13 Control	U-8/U	
Plugs—ART-13 Power	U-7/U	
Plugs—Barometric Altimeter Control	AN3108B-18-12S	
Plugs—BC-348	PL-Q-103	
Plugs—BC-348	AN3106B-12S-3S	
Plugs—Cabin Heater Fuel Control	AN3106-14S-2S	
Plugs—Cabin Station Box	AN3106B-22-19P	
Plugs—Carburetor Air Temperature Bulb	AF06-12S-3S	
Plugs—Carburetor Air Temperature Indicator	AN3106-14S-5S	
Plugs—Center Instrument Panel	AN3106-24-28S	Cannon
Plugs—Compass Light	AN3116	
Plugs—Controller	AN3108B-22-14S	
Plugs—Co-Pilot's Dual Azimuth Indicator	GK-9-23C½	Cannon
Plugs—Co-Pilot's ICS Control	AN3106B-28-15S	
Plugs—Co-Pilot's ICS Control	AN3106B-28-15P	
Plugs—Co-Pilot's Jack Box	ARC-9821	
Plugs—Co-Pilot's Jack Box	AN3106B-18-17P	
Plugs—Co-Pilot's Turn-and-Bank	AN3106-10SL-3S	
Plugs—Cowl Flap	AF06-14S-2S	
Plugs—Directional Gyro	AN3106-10SL-3S	
Plugs—Driftmeter	AN3106-12S-3S	
Plugs—Dynamotor	AN3106B-14S-7S	
Plugs—DY-17/ART-13 Power In	U-10/U	
Plugs—DY-17/ART-13 Power Out	U-9/U	
Plugs—Engaging Control	AN3108B-14S-5S	
Plugs—Engine Fire Detector Engine Disconnect	06AF-12S-4S	Cannon
Plugs—Engine Fire Detector Relay Box (Detector Input)	AN3108-14S-5S	
Plugs—Engine Fire Detector Relay Box (Power Input)	AN3108-14S-5SX	
Plugs—Engine Primer	AF06-12S-45	
Plugs—Engine Tachometer Generator	06AF-14S-1S	
Plugs—Engine Tachometer Indicator	AN3106-14S-1S	
Plugs—Firewall Fire Detector	AF06-12S-4S	
Plugs—Free Air Temperature Bulb	AN3106-12S-3S	
Plugs—Free Air Temperature Indicator	AN3106-14S-2S	
Plugs—Fuel Booster	AN3108-18-4S	

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Plugs—Fuel Pressure Warning Switch	08AF-14S-9S	
Plugs—G2 Adapter to Automatic Pilot	AN3108B-14S-6S	Cannon
Plugs—G2 Adapter Power	AN3108B-14S-2S	Cannon
Plugs—Generator Field Firewall Disconnect	FW06-16-11S	
Plugs—General Output — Firewall Disconnect	60832-5 65084-1	AiResearch
Plugs—General Output — Firewall Disconnect	60832-5 65084-1	AiResearch
Plugs—General Disconnect Firewall	FW08-24-28P	
Plugs—Glide Path	Part of Rack	
Plugs—(GREEN) ADF Receiver	AN3106B-12S-3S	
Plugs—(GREEN) ADF Receiver	PL-122	
Plugs—(GREEN) Loop Control	PL-112	
Plugs—Gyro Horizon to Control	AN3106-14S-2S	Cannon
Plugs—Gyro Horizon to 115V 3 ϕ	AN3106-14S-1S	Cannon
Plugs—Gyron Horizon to Miscellaneous Connections	AN3106-14S-6S	Cannon
Plugs—Horizon Gyro (Co-Pilot's)	AN3106-14S-1S	
Plugs—ICS Amplifier	AN3106B-16S-1S	
Plugs—ICS Amplifier	AN3106B-16S-8S	
Plugs—ICS Power Input	AN3106B-12S-3P	
Plugs—ID-250/ARN	AN3106B-22-14S	
Plugs—ID-249/ARN	AN3106B-28-12S	
Plugs—ID-48/ARN	AN3106B-18-1S	
Plugs—ID-251/ARN	AN3106B-20-29S	
Plugs—IFF Control Box	AN3106B-20-27S	
Plugs—Ignition Firewall	FW06-18-22S	
Plugs—Interphone Dynamotor	AN3106B-14S-7S	
Plugs—Alighting Gear Position Indicator	AN3106-18-1S	
Plugs—LM Frequency Meter	15A7009-2	
Plugs—Main Fuel Quantity Indicator	AN3106-16S-1S	
Plugs—Main Tank Unit	AN3106-14S-1S	
Plugs—Main Fuel Valve (Heater)	2123-3	
Plugs—Marker Beacon Receiver	AN3106B-16S-1S	
Plugs—Marker Beacon Receiver Indicator	AN3106B-16S-1S	
Plugs—Marker Beacon Receiver Power	AN3106B-16S-4S	
Plugs—Marker Beacon Switch	AN3106B-14S-7S	
Plugs—Marker Light	AN3106B-18-9S	
Plugs—Master Direction Indicator	AN3106-14S-2S	Cannon
Plugs—Navigation Lights Flasher	AN3106-20-4S	
Plugs—Navigation Station Box	AN3106B-22-19P	
Plugs—Oil Cooler Door	AF06-14S-7S	
Plugs—Oil Cooler Door Thermo Control	FW06-16-10S	
Plugs—Oil Dilution	2123-3	
Plugs—Oil Dilution Firewall	3672-28	Cannon
Plugs—Oil Pressure Warning Switch	08AF-14S-9S	
Plugs—Oil Temperature Bulb	AN3106-12S-3S	
Plugs—Oil Temperature Indicator	AN3106-14S-5S	
Plugs—Outer Wing Fuel Quantity Indicator	AN3106-14S-2S	
Plugs—Outer Wing Fuel Quantity Tank Unit	AN3106-14S-2S	
Plugs—Overhead Control Panel	AN3106B-28-21PX	
Plugs—Overhead Control Panel	AN3106B-28-21PW	
Plugs—Overhead Control Panel	AN3106B-28-21P	
Plugs—Pedestal to FPC	AN3108-B20-29S	
Plugs—P1-G2 Interconnector Box	AN3108B-14S-5P	
Plugs—P1-G2 Interconnector Box	AN3108B-14S-5P	
Plugs—Pilot's Dual Azimuth Indicator	GK-9-23C½	Cannon
Plugs—Pilot's ICS Control	AN3106B-28-15S	
Plugs—Pilot's ICS Control	AN3106B-28-15P	
Plugs—Pilot's Jack Box	ARC-9821	
Plugs—Pilot's Jack Box	AN3106B-18-9P	
Plugs—Pitot Heater	AN3115-1	
Plugs—Power Input Power Junction Box	AN3108B-20-4S	Cannon
Plugs—Power Junction Box to Channel Amplifier	AN3108B-20-16S	Cannon
Plugs—Propeller Feather Cutout Switch	AF06-12S-4S	Cannon
Plugs—Range Receiver	AN3106B-12S-3S	
Plugs—Range Receiver Control Box	AN3106B-16S-8P	
Plugs—Range Receiver Power	ARC9125	
Plugs—(RED) ADF Receiver	PL-122	

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Plugs—(RED) ADF Receiver	AN3106B-12S-3S	
Plugs—(RED) Loop Control	PL-112	
Plugs—Remote Compass Transmitter	AN3106-14S-2S	Cannon
Plugs—R/O ICS Control	AN3106B-28-15S	
Plugs—R/O ICS Control	AN3106B-28-15P	
Plugs—R/O Jack Box	AN3106B-20-7P	
Plugs—R/O Jack Box	ARC-9821	
Plugs—Servo to Adapter	AN3106-18-15	
Plugs—Starter Firewall	60829-5 65083-1	AiResearch
Plugs—Station Box	AN3106B-22-19P	
Plugs—Throttle Servo	AN3106-18-1S	
Plugs—Trim Tab Servo	AN3106-18-1S	
Plugs—Trim Tab Servo Adapter to Amplifier Adapter	AN3106B-18-1S	
Plugs—Trim Tab Servo Adapter to Elevator Servo	AN3106B-18-1S	
Plugs—Trim Tab Servo to Trim Tab Servo	AN3106B-18-1S	
Plugs—Turn-and-Bank to Miscellaneous Wiring	AN3106-14S-1S	
Plugs—Turn-and-Bank to 115V 3 ϕ	AN3106-14S-5S	
Plugs—UHF to ICS Junction Box	AN3106B-28-21PX	
Plugs—UHF Master Control	AN3108B-24-5P	
Plugs—UHF Master Control	AN3108B-24-20P	
Plugs—UHF Master Control	AN3108B-24-7S	
Plugs—UHF Transmitter Receiver	AN3108B-24-5S	
Plugs—UHF Transmitter Receiver	AN3108B-24-7S	
Plugs—250 va Inverter	AN3106-22-4S	
Plugs—VHF Control Units	AN3106B-20-29P	
Plugs—VHF Frequency Control	AN3106B-20-1P	
Plugs—VHF Power	AN3106B-20-16P	
Plugs—VHF Transmitter	AN3106B-18-9S	
Plugs—VOR Dynamotor	Part of Rack	
Plugs—VOR Receiver	AN3106B-12S-3S	
Plugs—VOR Receiver	Part of Rack	
Receptacles—Center Instrument Panel	AN3102-24-28P	
Receptacles—Engine Fire Detector Engine Disconnect	FW00-12S-4P	Cannon
Receptacles—Engine Fire Detector Firewall Disconnect	FW02-12S-4S	
Receptacles—Firewall Ignition	FW02-18-22P	Cannon
Receptacles—Flight Path Compartment	Part of Base Assembly	
Receptacles—General Disconnect Firewall	FW02-24-28S	Cannon
Receptacles—General Field Firewall Disconnect	FW02-16-11P	Cannon
Receptacles—Generator Output — Firewall Disconnect	60816-5	AiResearch
Receptacles—Generator Output — Firewall Disconnect	60816-5	AiResearch
Receptacles—Ground Power Disconnect	AN2552-A3	
Receptacles—Oil Dilution Firewall	FW02-10S-2P	Cannon
Receptacles—Periscopic Sextant	38A 1549	Air Force
Receptacles—Periscopic Sextant Mount	AN3106-10SL-35	
Receptacles—Starter Firewall Disconnect	60816-5	AiResearch
Receptacles—Throttle Servo Amplifier	Part of Base Assembly	
Receptacles—To Overhead Control Panel	AN3100A-28-21S	
Receptacles—To Overhead Control Panel	AN3100A-28-21SW	
Receptacles—To Overhead Control Panel	AN3100A-28-21SX	
Receptacles—To Range Control Box	AN3100A-16S-8S	
Receptacles—To VHF Control Box	AN3100A-20-29S	
Regulators—Generator Voltage	E1597-1	BuAer Drawing
Rheostats—Aft Cabin Dome	4371381-508	Douglas
Rheostats—Air Control Adjustment	JLU1021	Allen Bradley
Rheostats—Center Instrument Panel Lights	4371381-506	Douglas
Rheostats—Co-Pilot's Instrument Panel Lights	4371381-502	Douglas
Rheostats—Co-Pilot's Panel Floodlights	4371381-507	Douglas
Rheostats—Gooseneck Lights	4371381-504	Douglas
Rheostats—Pilot's and Co-Pilot's Upper Switch Panel Lights	4371381-502	Douglas
Rheostats—Pilot's Instrument Panel Lights	4371381-506	Douglas
Rheostats—Pilot's Panel Floodlights	4371381-507	Douglas

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Relays—A-C Interlock	8504-2	Electro Switch
Relays—A-C Interlock Low-Voltage Transfer	8504-2	Electro Switch
Relays—A-C Warning	2365673	Douglas
Relays—Automatic Pilot Engaging Control	8504-2	Electro Switch
Relays—Auto Warning Light Dimming Rotary	2397436	Douglas
Relays—Battery	B4A	
Relays—Co-Pilot's Directional Gyro Power Transfer	8504-2	Electro Switch
Relays—Co-Pilot's Horizon Power Transfer	8504-2	Electro Switch
Relays—Engine Fire Detector	35884 Mod. 117-52A	 Edison
Relays—FPC Power	8507-2	Electro Switch
Relays—G2 Compass Transfer	8504-2	Electro Switch
Relays—General Cutout	8504-2	Electro Switch
Relays—Reverse Current	A-750D	Hartman
Relays—Generator Auxiliary	8504-2	Electro Switch
Relays—(GREEN) ADF Power	8504-2	Electro Switch
Relays—Ground Blower Control	AN3350-2	
Relays—Ground Power	AN3025-1	
Relays—Ground Power Control	8504-2	Electro Switch
Relays—Ground Power Transfer	8504-2	Electro Switch
Relays—Heater Slave	8504-2	Electro Switch
Relays—HF Receiver Disable	8504-2	Electro Switch
Relays—Horn Isolation	8504-2	Electro Switch
Relays—Horn Silence Engine No. 1 Throttle	8504-2	Electro Switch
Relays—Horn Silence Engine No. 2 Throttle	8504-2	Electro Switch
Relays—IFF Mode Change (UHF)	8504-2	Electro Switch
Relays—IFF Mode Change (VHF)	8504-2	Electro Switch
Relays—Landing Lights	AN3350-2	
Relays—Overvoltage Control	1623-6	Eclipse
Relays—Pilot's Horizon and Turn-and-Bank Power Transfer	8504-2	Electro Switch
Relays—Propeller Feather	B4A	
Relays—(RED) ADF Power	8504-2	Electro Switch
Relays—Scoop Heater	7064-758	Leach
Relays—Starter	B8A	
Relays—Throttle Servo Disconnect	8504-2	Electro Switch
Relays—Throttle Servo Power	8507-21	Electro Switch
Relays—250 va Inverter Slave	AN3350-2	
Relays—VOR Damping	8504-2	Electro Switch
Relays—VOR Damping	8504-2	Electro Switch
Resistors—Automatic Pilot Torque Limiting	0200E	Ohmite
Resistors—ARC-1 Volume	JLU5011	Allen-Bradley
Resistors—ARR-15 Sensitivity	JLU5011	Allen-Bradley
Resistors—Automatic Warning Light Dimming	1013	Ohmite
Resistors—Compass Light Voltage Drop	0371	Dividohm
Resistors—Engine Fire Detection Sensitivity	4.7 ohm ½ watt BW½	 IRC
Resistors—Heater Cycle Arc Suppressor	5ohm 10 watt Brolo	 Ohmite
Resistors—Horizon Pointer Loading	BTS1000	IRC
Resistors—Periscopic Sextant Voltage Drop	1013	Ohmite
Resistors—Vertical Pointer Loading	BTS1000	IRC
Servos—Automatic Pilot	15604-1-3-A2	Pioneer
Servos—Throttle	15621-1-A	Pioneer
Servos—Trim Tab	15622-1-A	Pioneer
Shunts—Airscoop Heater Ammeter	4142341G49	General Electric
Shunts—Main Generator Ammeter	AN3200-450	
Shunts—Pitot Heater Ammeter	4142341G47	General Electric
Solenoids—Engine Primer	392273	Furnished with carburetor
Solenoids—Heater Main Fuel Valve	G104-D2	Minneapolis-Honeywell
Solenoids—Alighting Gear Safety	4673	Cannon
Solenoids—Oil Dilution	G104-D2	Minneapolis-Honeywell

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Switches—A-C Interlock Bypass Switch	AN3021-2	
Switches—Airvalve Heater Control	BZR31	Micro
Switches—Alcohol Pump Motor	AN3022-2	
Switches—Altitude Control	CQ-21-A1	Pioneer
Switches—Altitude Limit	SA-1/ARN-1	
Switches—Blue Right	AN3021-2	
Switches—Cabin Heater Fire Detector Test	A-406-M4	Heatherington
Switches—Cabin Heater Master	AN3022-2	
Switches—Compass Light Dimming	AN3021-1	
Switches—Cowl Flap Control	8905K671	C-H
Switches—Disengage (Pilot and Co-Pilot) Automatic Pilot	A402Y4Y MC992-4	Heatherington
Switches—Engine Fire Detector Test	A-406-M4	Heatherington
Switches—Engine Primer	8905K55	C-H
Switches—Engine Start Select	AN3027-1	
Switches—Forward Door Open Warning	BZ-3YT	Micro
Switches FPC Selector	239917	Douglas
Switches—Free Directional Gyro (Automatic Pilot)	AN3023-2	
Switches—Fuel Booster Pump Control	7662-K4	
Switches—Fuel Pressure Warning	417-15B-37	Meletron
Switches—Generator Cutout	AN3027-2	
Switches—Heater Cycle	2359360-512	
Switches—Heater Drop-Out	2359360-508	Douglas
Switches—Horn Silence	AN32131	Heatherington
Switches—Ignition	A8006	
Switches—Impact	SA-3/A	
Switches—Alighting Gear Lever	BZ-R31	Micro
Switches—Landing Light Control	AN3022-2	
Switches—Landing Light Extend-Retract	8207K3	
Switches—Left and Right Gear Down	BZ-3YT	Micro
Switches—Left and Right Oleo	BZ-3YT	Micro
Switches—Navigation Lights Control	AN3023-1	
Switches—Oil Cooler Doors	8905K608	C-H
Switches—Oil Dilution	AN3021-7	
Switches—Oil Pressure Warning	4171-21B-11	Meletron
Switches—Pitot Heater	AN3023-2	
Switches—Pitot and Scoop Heater Metering Select	7262-K6	C-H
Switches—Power Selector	AN3021-1	
Switches—Propeller Feathering	2CC1B4	General Electric
Switches—Propeller Pressure Cutout	69040	Hamilton Standard
Switches—Right and Left Gear Up	BZ-R31	Micro
Switches—R/O Monitor	AN3023-3	
Switches—Scoop Heater	AN3022-2	
Switches—Start and Safety	8905K755	C-H
Switches—Tail Access and Lavatory Dome Lights	AN3022-2	
Switches—Tail Wheel Down	V3-1	Micro
Switches—Tail Wheel Up	BZ-R31	Micro
Switches—Throttle Lock	AN3234-1	
Switches—1500 va Inverter Control	7662-K4	C-H
Switches—250 va Inverter Control	AN3021-3	
Switches—Wing Edge Lights Control	AN3022-2	
Terminal Strip—Center Instrument Panel	NAS 191	
Terminal Strip—Circuit Breaker Power Feed	NAS 45-416-1	
Terminal Strip—Co-Pilot's Instrument Panel	NAS 191	
Terminal Strip—De-Icer Boot Motor Disconnect	NAS 191	
Terminal Strip—Engine Section Junction Box	NAS 191	
Terminal Strip—Entrance Way Dome Light Junction Box	NAS 191	
Terminal Strip—Generator	4398750	Douglas
Terminal Strip—Generator Tunnel Terminal	4118916	Douglas
Terminal Strip—Alighting Gear Unlock Solenoid	NAS 191	
Terminal Strip—Left and Right Nacelle Wing Junction Boxes	NAS 191	
Terminal Strip—MJB	NAS 191	
Terminal Strip—Nacelle Junction Box Right	NAS 191	
Terminal Strip—Nacelle Junction Box Left	NAS 191	
Terminal Strip—Nose	NAS 191	

ELECTRICAL AND ELECTRONIC EQUIPMENT LIST *(Continued)*

Part Name	*AN or Vendor No.*	*Manufacturer*
Terminal Strip—Nose Terminal	NAS 45-6-60	
Terminal Strip—Pilot's Instrument Panel	NAS 171	
Terminal Strip—Radio Junction Box	NAS 45-6 (368 Terminals)	
Terminal Strip—Receiver Antenna Junction	NAS 191-1	
Terminal Strip—R/O Junction Box	NAS-45-6 (20 Terminals)	
Terminal Strip—R/O Junction Box Power Feed	NAS 45-416-1	
Terminal Strip—Side Panel Lights	NAS 191	
Terminal Strip—Tail Cone Disconnect	NAS 191	
Terminal Strip—Tail Alighting Gear	NAS 191	
Terminal Strip—Tunnel	NAS 191	
Thermocouples—Cabin Heater Temperature	8T31B81	Lewis
Thermocouples—Engine Cylinder Head Temperature	5541-1	
Transformers—Inverter 115v to 26v 3 ϕ	DW-57-0	Eclipse-Pioneer
Transformers—VOR Instrumentation	70G176	General Electric
Transmitters—Alcohol Quantity	EA584-786	Liquidometer
Transmitters—G2 Compass	8KJ2AA1	General Electric
Transmitters—Main Tank Fuel Quantity	EA-15	Liquidometer
Transmitters—Outer Wing Fuel Quantity Inboard	EA-828-799	Liquidometer
Transmitters—Outer Wing Fuel Quantity Outboard	EA-821-798	Liquidometer
Units—Barometric Altimeter Control	15505-1-A1	Pioneer
Units—Cabin Station Box	C-70A/AIA-2A	
Units—Co-Pilot's ICS Control Box	C-69A/AIA-2A 5398248	Douglas Modified
Units—Co-Pilot's Jack Box	J-22/ARC-5	
Units—Engaging Control Assembly	CM-8-A1	Pioneer
Units—Frequency Meter	LM-(2)	
Units—Flight Path Compartment	E16004	Pioneer
Units—Glide Path Receiver	R-322/ARN-18	
Units—(GREEN) ADF Receiver	R-5A/ARN-7	
Units—HF Receiver	BC-348	
Units—HF Receiver	R-1 05/ARR-15A	
Units—HF Transmitter	T47/ART-13	
Units—HF Transmitter Dynamotor	DY-17/ART-13	
Units—ICS Amplifier	AM-40/AIC	
Units—ICS Dynamotor	DY-13A/AIA-2A	
Units—ICS Junction Box	J-36A/AIA-2A	
Units—IFF Control	C-544/APX-6	
Units—IFF Transmission Receiver	RT-82/APX-6	
Units—LORAN Antenna Coupler	CU-92A/APN	
Units—Marker Beacon Receiver	R-8/ARN-8	
Units—Navigator's Station Box	C-70Z/AIA-2A	
Units—Pilot's ICS Control Box	C-69A/AIA-2A 5398248	
Units—Pilot's Jack Box	J-22/ARC-5	Douglas Modified
Units—Radio Altimeter	RT-7/APN-1	
Units—Range Receiver	R-23A/ARC-5	
Units—Range Receiver Control Box	C-125/ARC-5	
Units—(RED) ADF Receiver	R5A/ARN-7	Douglas Modified
Units—R/O ICS Control Box	C-69A/AIA-2A 5398248	
Units—R/O Jack Box	J-22/ARC-5	Douglas Modified
Units—UHF Frequency Control Panel	C-626/ARC-27	
Units—UHF Receiver Transmitter	RT-178/ARC-27	
Units—VHF Control Unit	3399027	
Units—VHF Transmitter Receiver	RT-18/ARC-1	
Units—VOR Dynamotor	DY-66/ARN-14	
Units—VOR Receiver	R-252/ARN-14	
Vibrators—Engine Induction	VJR24C5	Bosch

10-11. INDEX OF WIRING DIAGRAMS.

10-12. In the following index all wiring diagrams are listed alphabetically by titles. All wiring diagrams apply to aircraft A through D, and 1 through 96, unless otherwise noted.

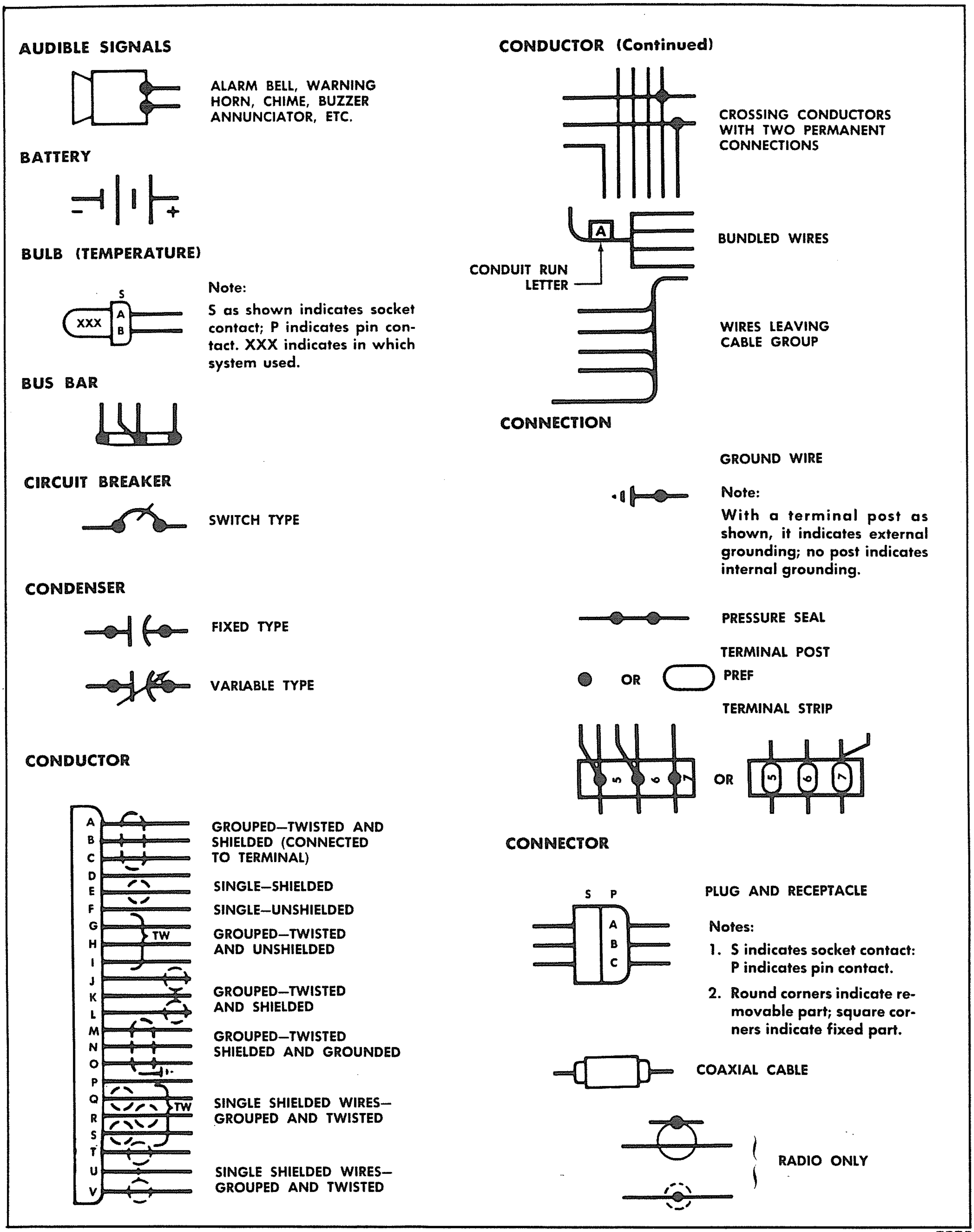

Figure 10-3 (Sheet 1 of 4 Sheets). Electrical Symbols

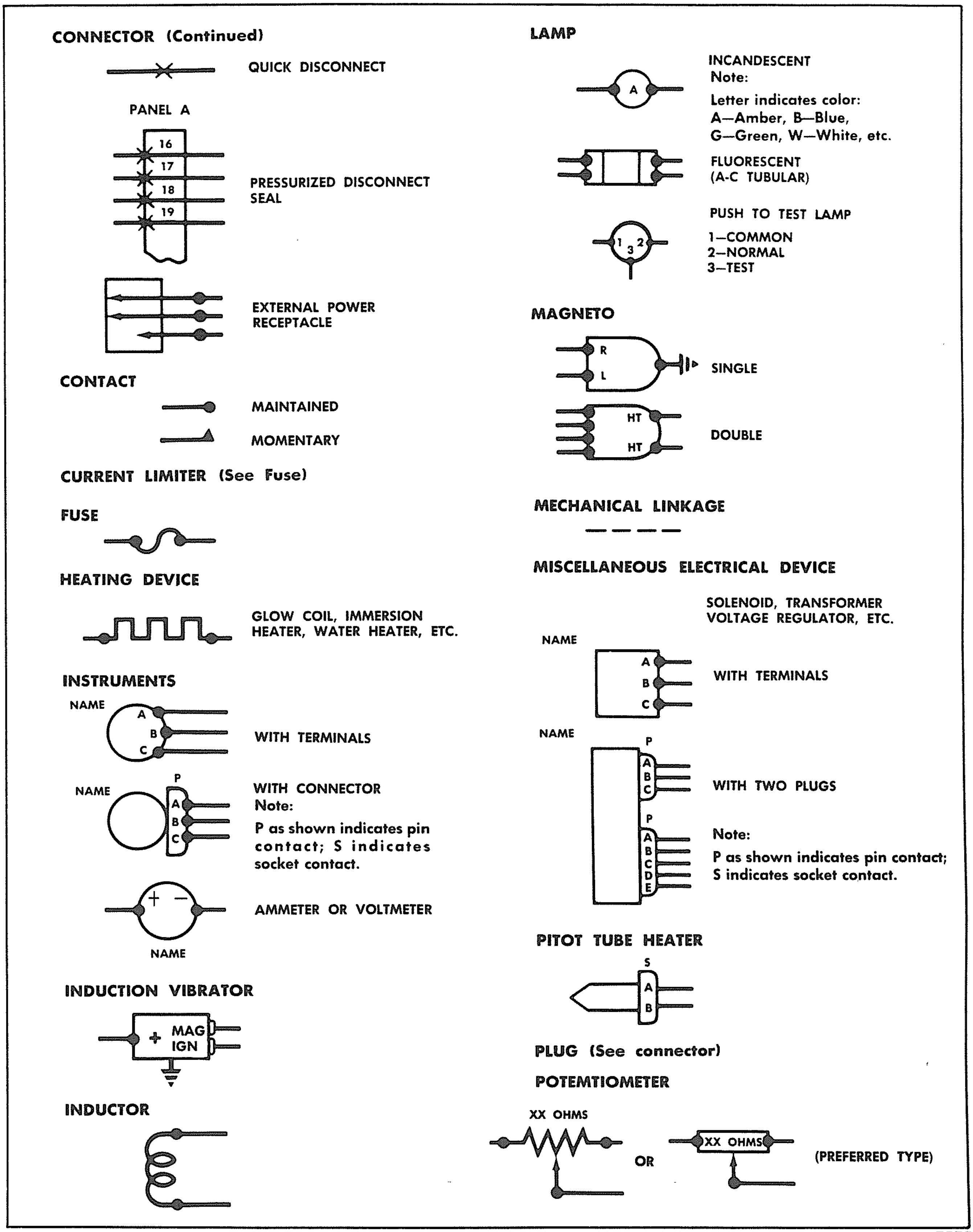

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Figure 10-3 (Sheet 2 of 4 Sheets). Electrical Symbols

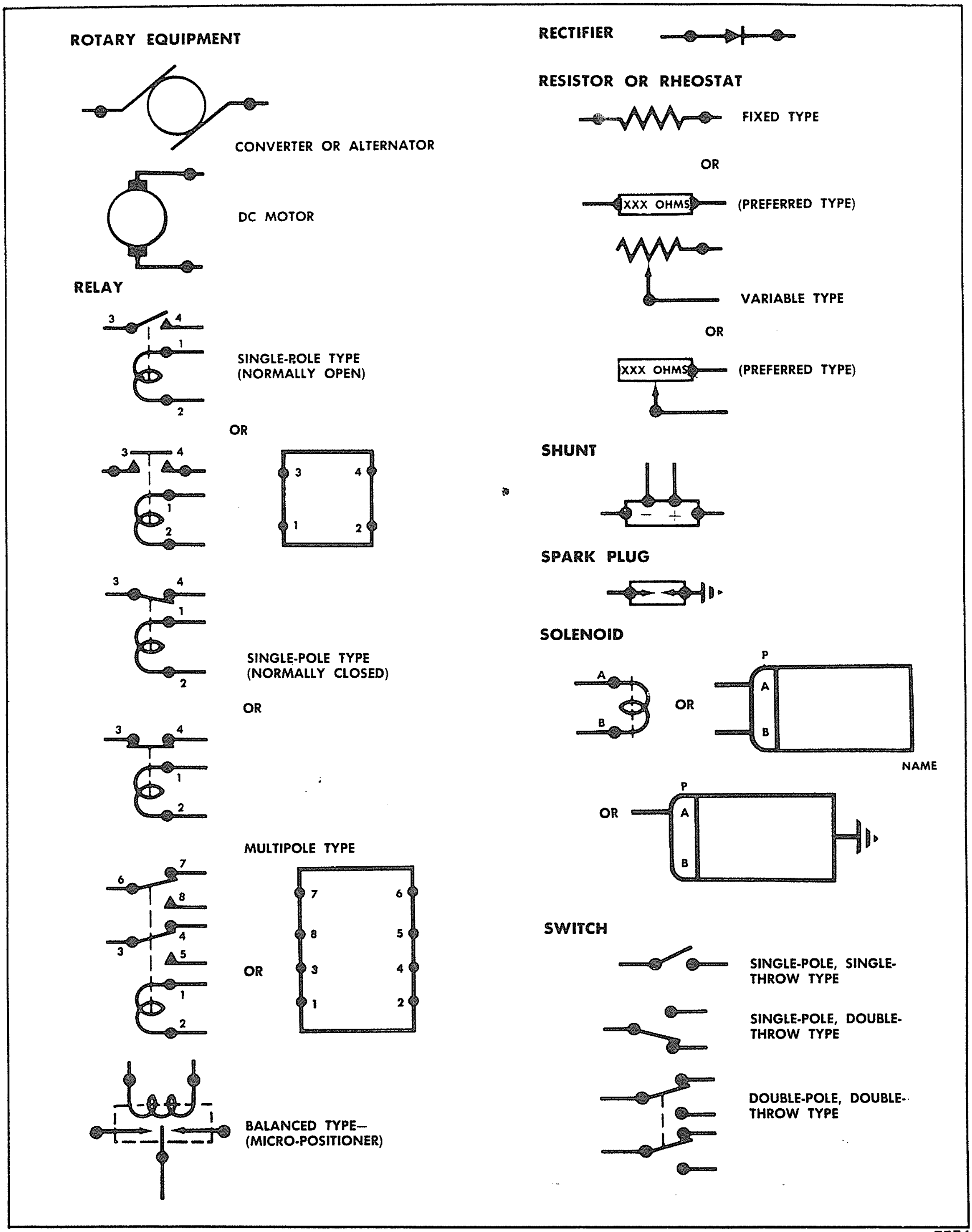


Figure 10-3 (Sheet 3 of 4 Sheets). Electrical Symbols

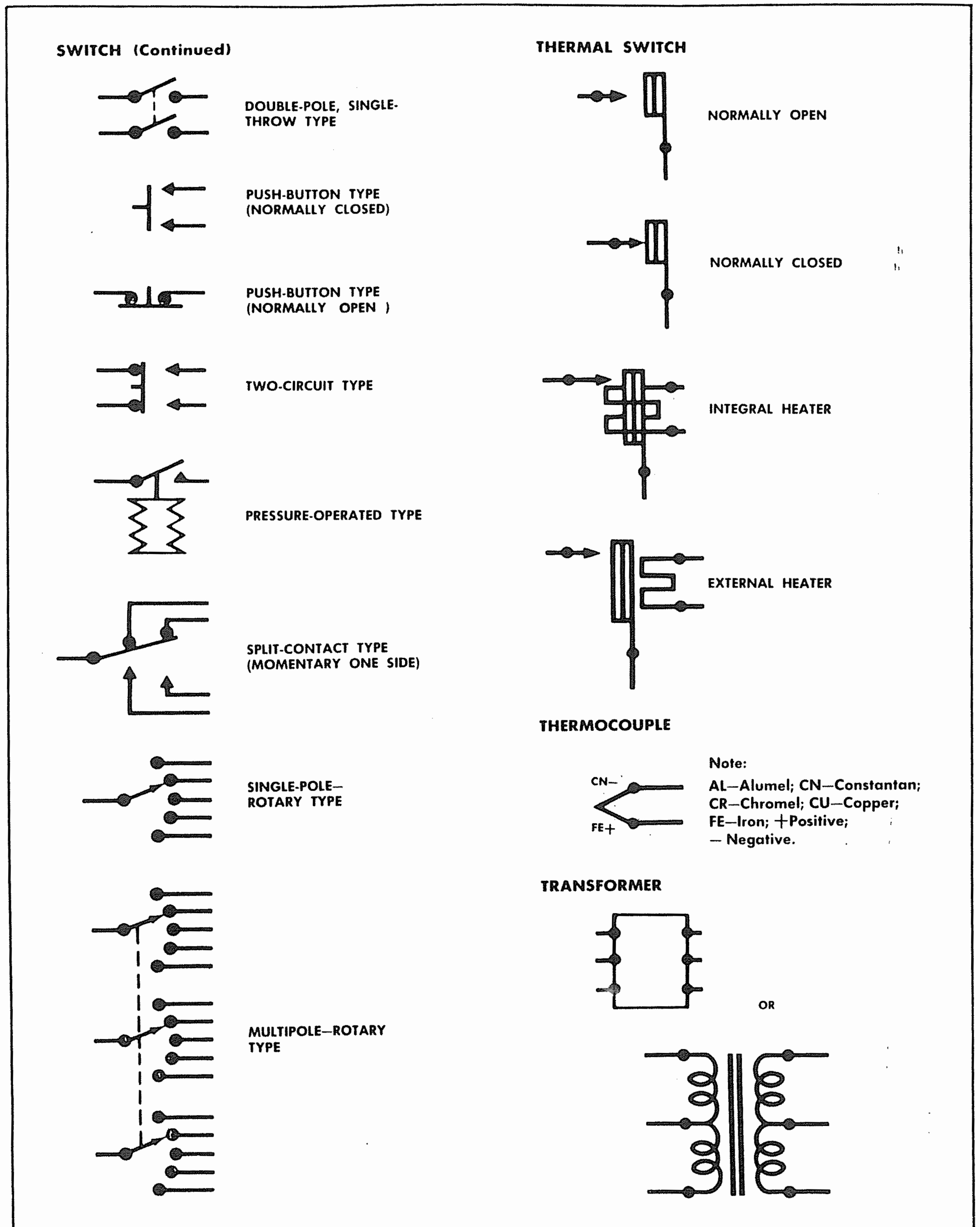
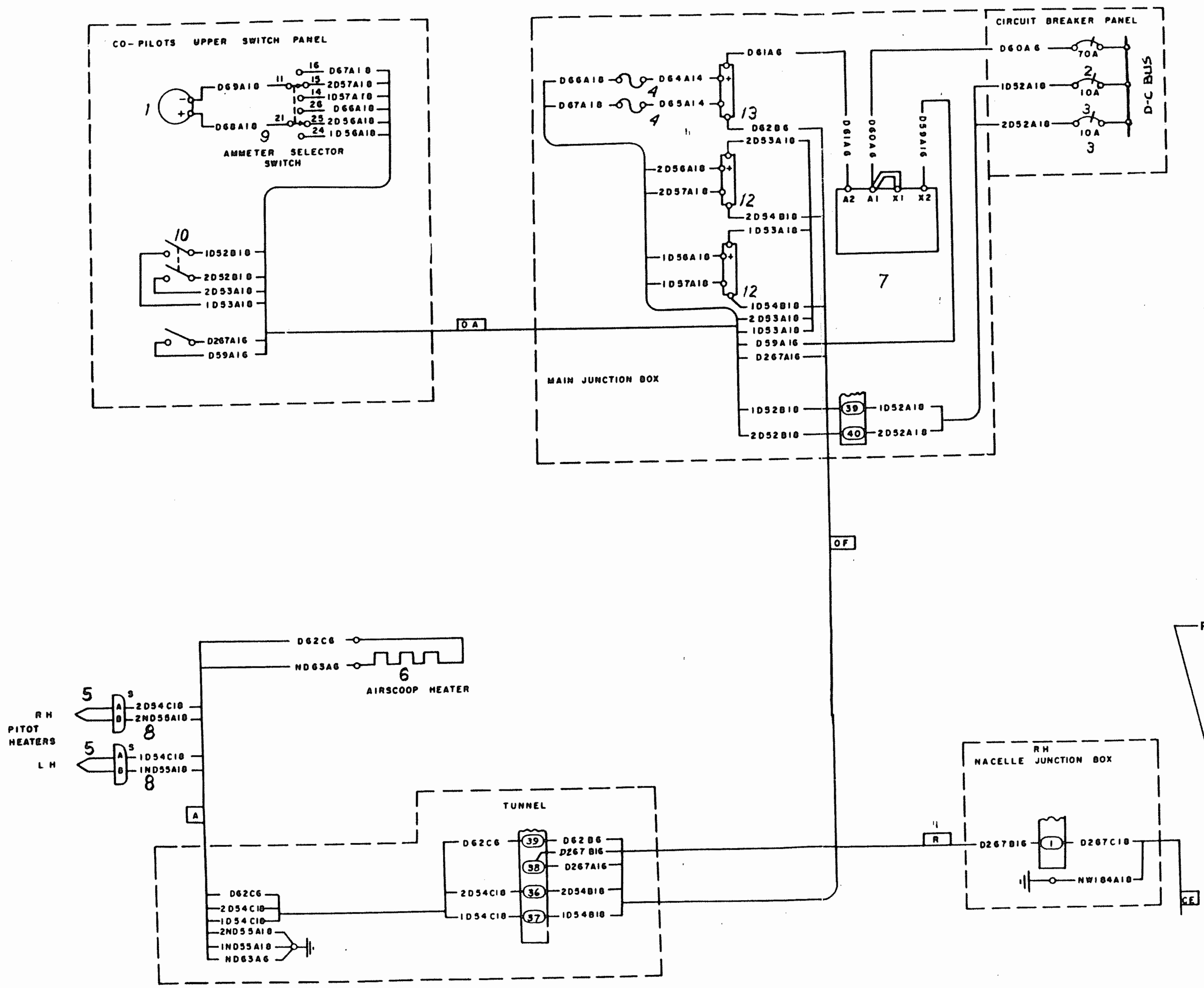


Figure 10-3 (Sheet 4 of 4 Sheets). Electrical Symbols



ITEM	PART AND DESCRIPTION
1	HEATER AMMETER
2	CIRCUIT BREAKER (70A)
3	CIRCUIT BREAKER (10A)
4	FUSE
5	PITOT HEATER
6	AIRSCOOP HEATER
7	SCOOP HEATER RELAY
8	PITOT HEATER PLUG
9	AMMETER SELECTOR SWITCH
10	PITOT HEATER SWITCH
11	SCOOP HEATER SWITCH
12	PITOT HEATER SHUNT
13	AIRSCOOP HEATER SHUNT

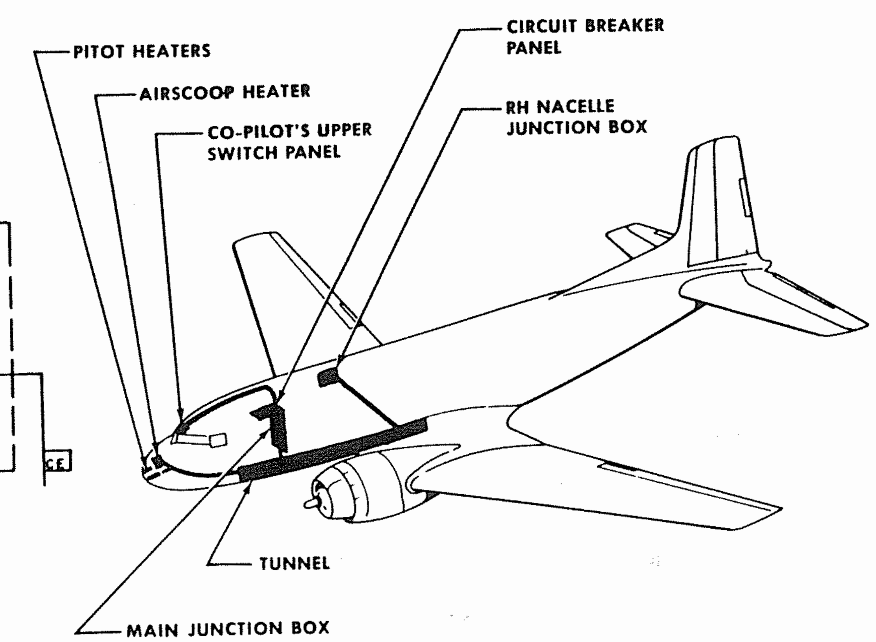


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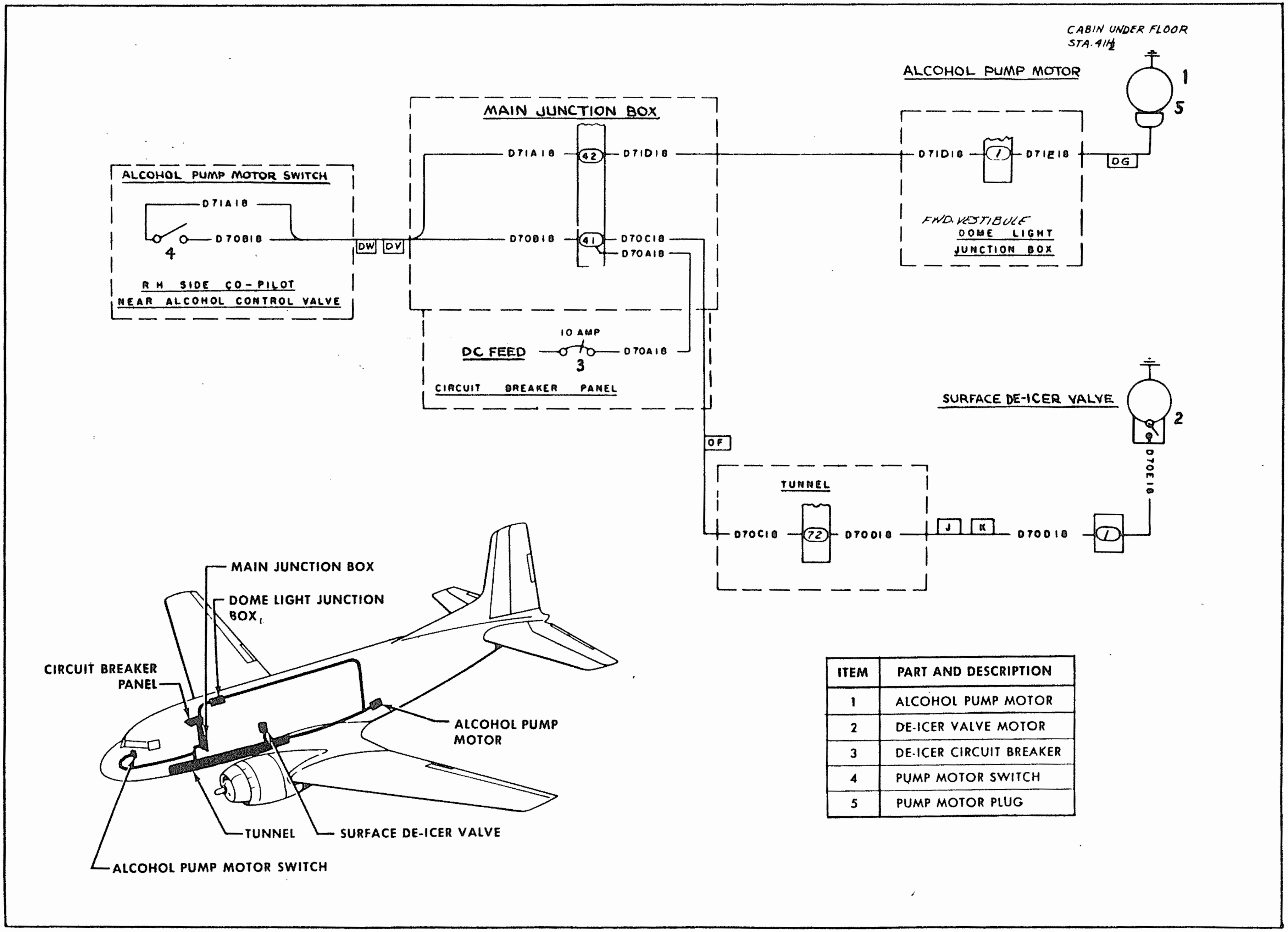
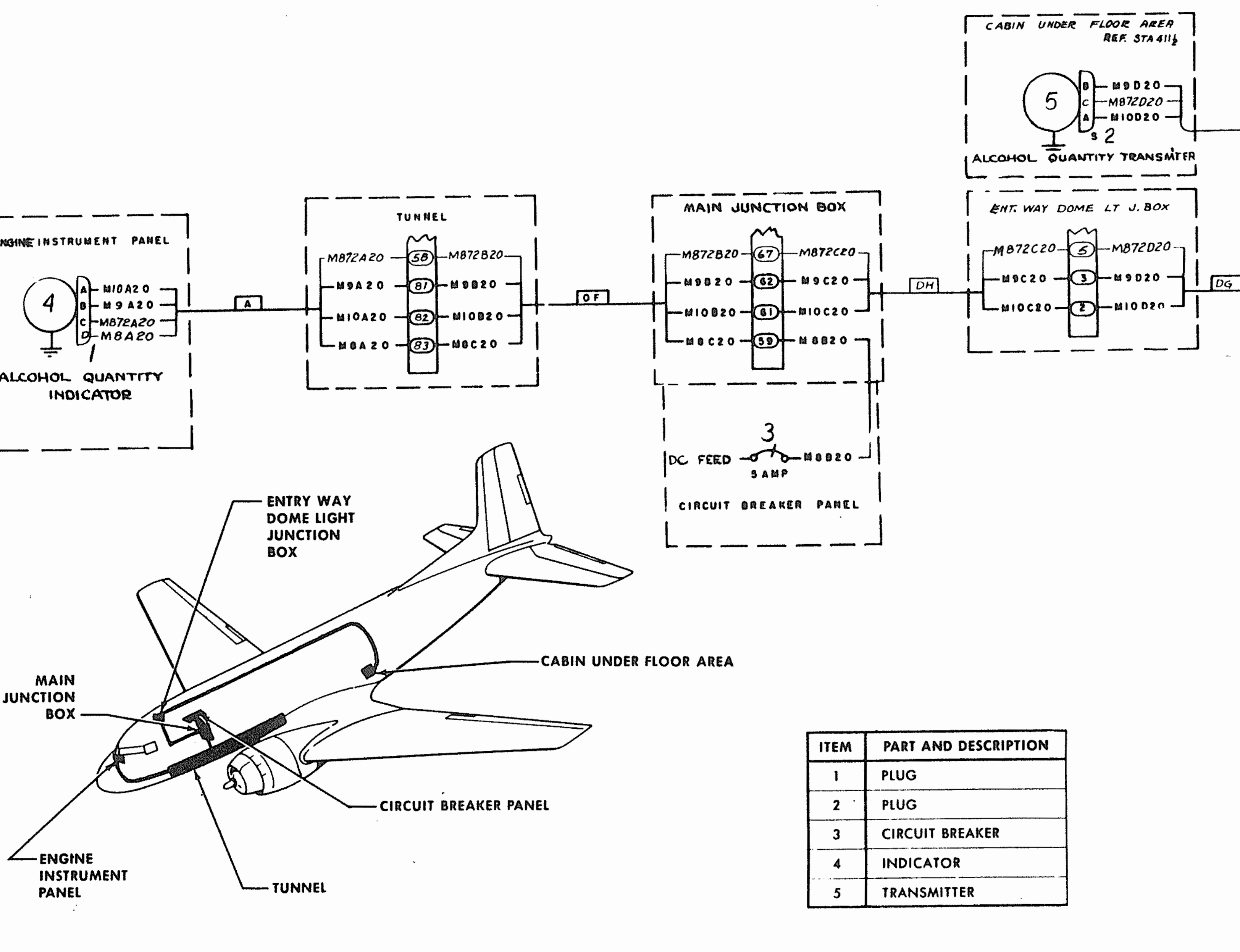
Figure 10-4. Airscoop and Pitot Metering and Control (Aircraft A, B, and 1 through 78)



ITEM	PART AND DESCRIPTION
1	ALCOHOL PUMP MOTOR
2	DE-ICER VALVE MOTOR
3	DE-ICER CIRCUIT BREAKER
4	PUMP MOTOR SWITCH
5	PUMP MOTOR PLUG

Figure 10-5. Alcohol Pump and Surface De-Icer Motors



ITEM	PART AND DESCRIPTION
1	PLUG
2	PLUG
3	CIRCUIT BREAKER
4	INDICATOR
5	TRANSMITTER

Figure 10-6. Alcohol Quantity System

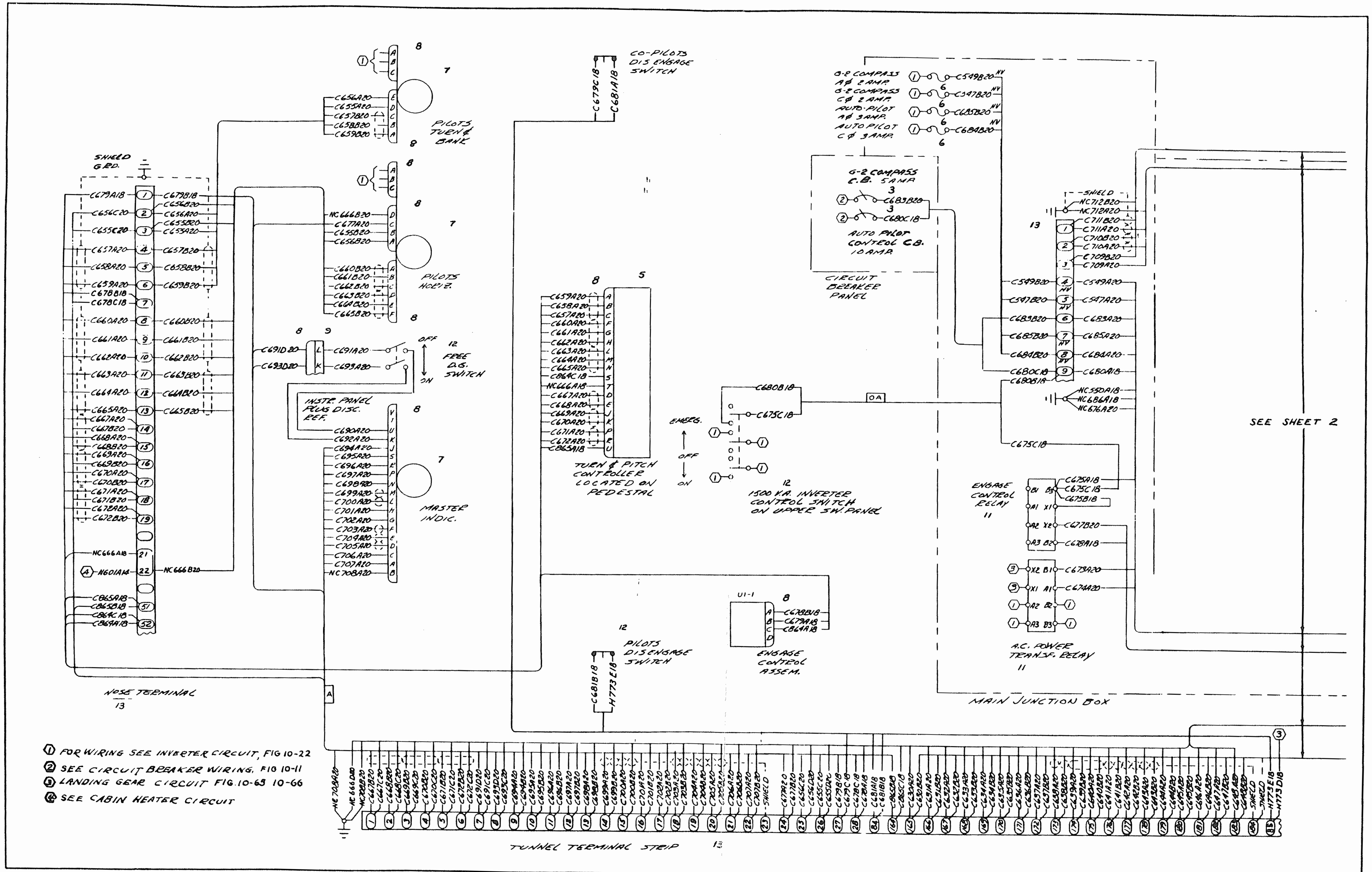


Figure 10-7 (Sheet 1 of 2 Sheets). Automatic Pilot and G-2 Compass (Aircraft A, B, and 1 through 68)

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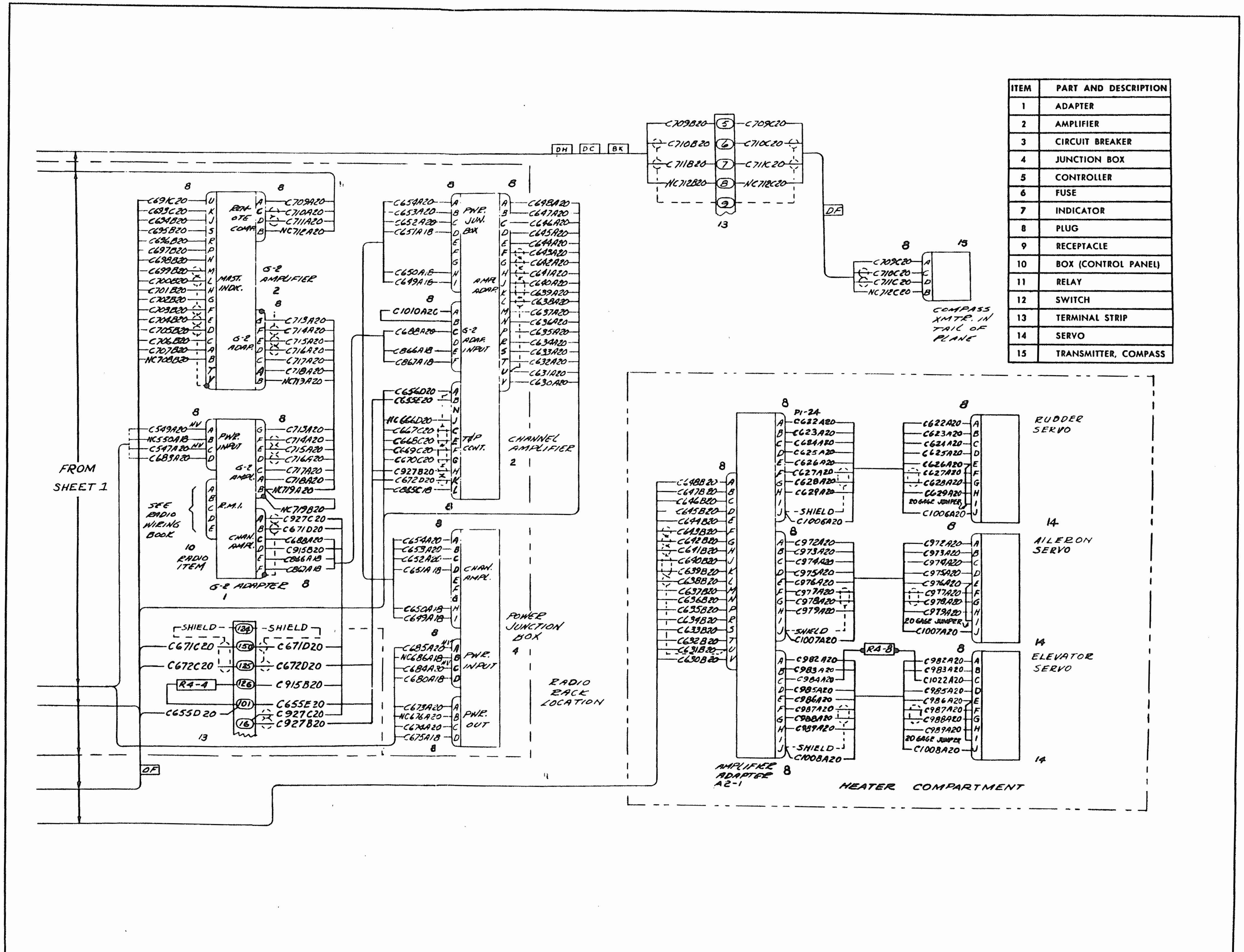

ITEM	PART AND DESCRIPTION
1	ADAPTER
2	AMPLIFIER
3	CIRCUIT BREAKER
4	JUNCTION BOX
5	CONTROLLER
6	FUSE
7	INDICATOR
8	PLUG
9	RECEPTACLE
10	BOX (CONTROL PANEL)
11	RELAY
12	SWITCH
13	TERMINAL STRIP
14	SERVO
15	TRANSMITTER, COMPASS

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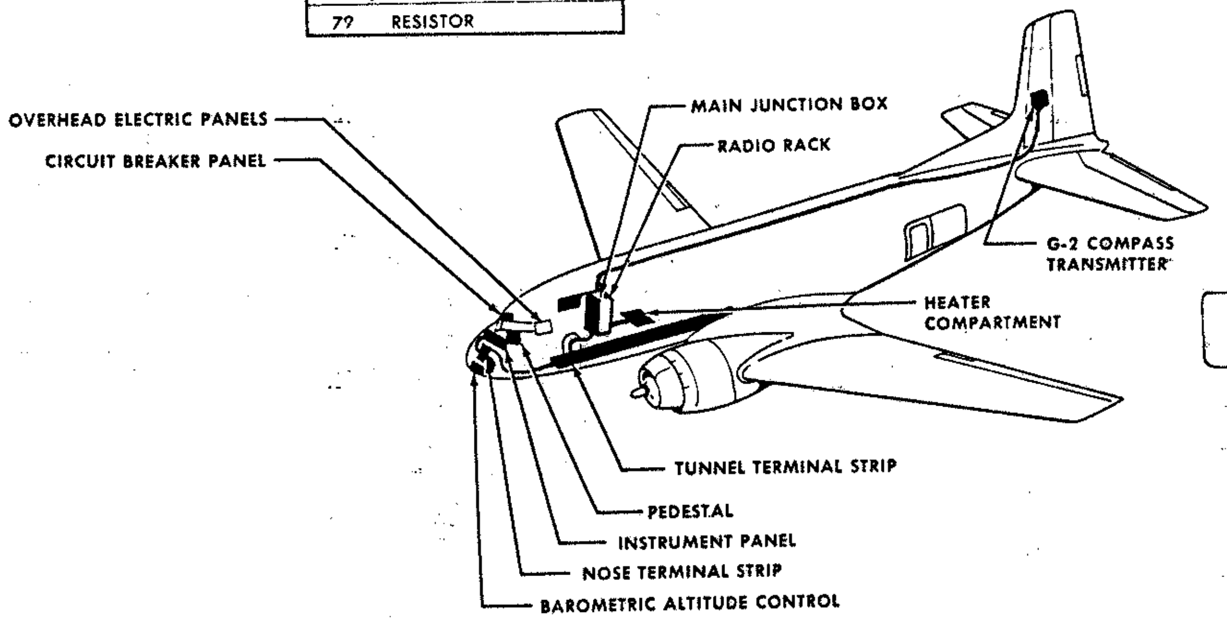
Figure 10-7 (Sheet 2 of 2 Sheets). Automatic Pilot and G-2 Compass (Aircraft A, B, and 1 through 68)

ITEM	PART AND DESCRIPTION
1	PLUG
2	UNIT
3	PLUG
4	RECEPTACLE
5	SWITCH
6	PLUG
7	INDICATOR
8	PLUG
9	PLUG
10	PLUG
11	INDICATOR
12	PLUG
13	SWITCH
14	PLUG
15	INDICATOR
16	INDICATOR
17	SWITCH
18	SWITCH
19	PEDESTAL RECEPTACLE
20	PLUG
21	LAMP
22	LAMP ASSEMBLY
23	SWITCH
24	SWITCH
25	LAMP
26	LAMP ASSEMBLY
27	CONTROLLER
28	PLUG
29	FUSE
30	FUSE
31	CIRCUIT BREAKER
32	CIRCUIT BREAKER
33	SWITCH
34	RELAY
35	PLUG
36	UNIT
37	PLUG
38	PLUG
39	PLUG
40	PLUG
41	PLUG
42	PLUG
43	ADAPTOR
44	PLUG
45	PLUG
46	BOX
47	RELAY
48	RELAY
49	PLUG

ITEM	PART AND DESCRIPTION
50	AMPLIFIER
51	PLUG
52	PLUG
53	PLUG
54	RECEPTACLE
55	UNIT
56	RHEOSTAT
57	PLUG
58	RECEPTACLE
59	ADAPTOR
60	PLUG
61	PLUG
62	PLUG
63	PLUG
64	ADAPTOR
65	TRANSMITTER
66	PLUG
67	PLUG
68	PLUG
69	PLUG
70	ADAPTOR
71	PLUG
72	SERVO
73	PLUG
74	SERVO
75	PLUG
76	SERVO
77	PLUG
78	AMPLIFIER
79	RESISTOR



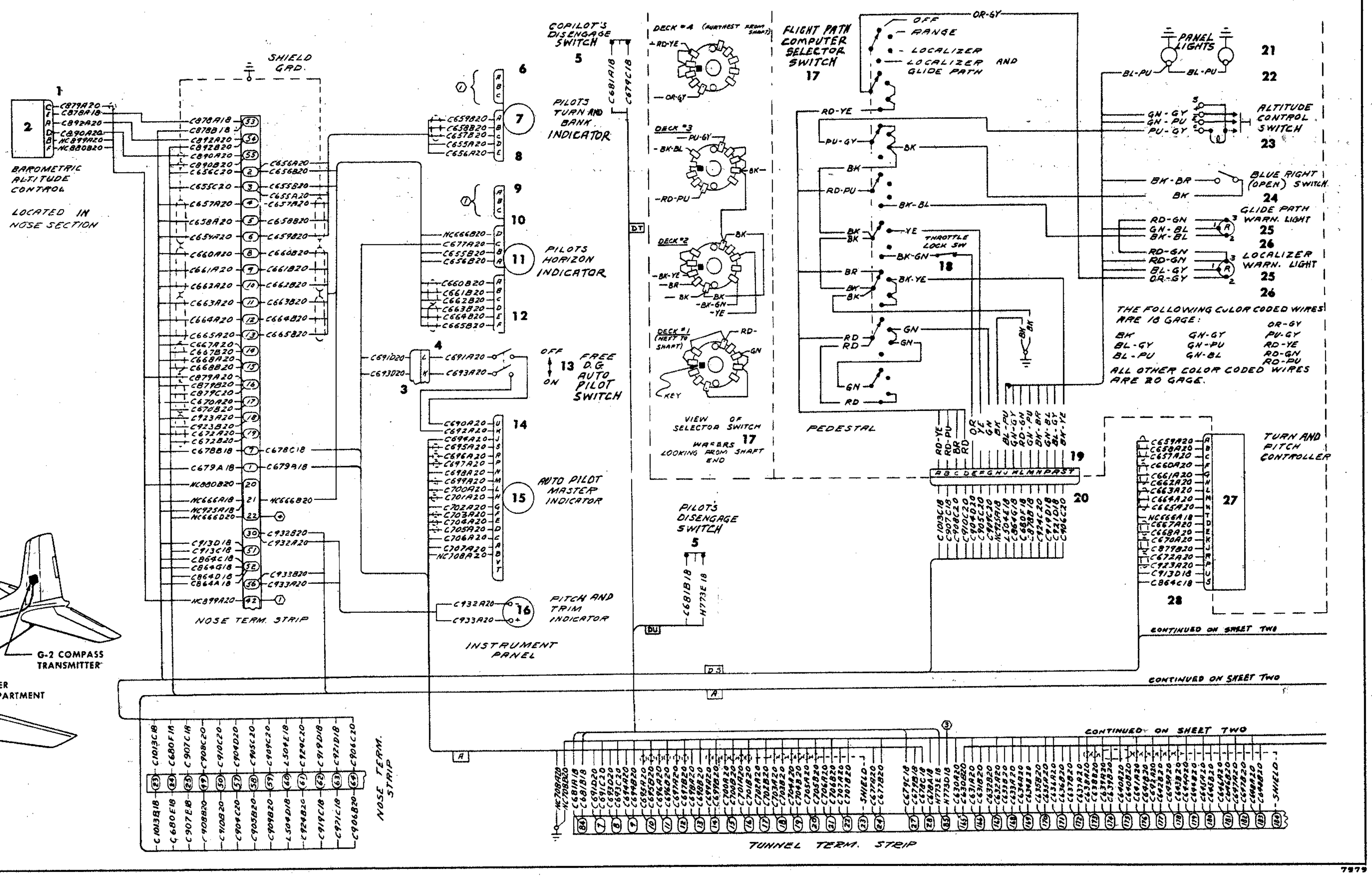
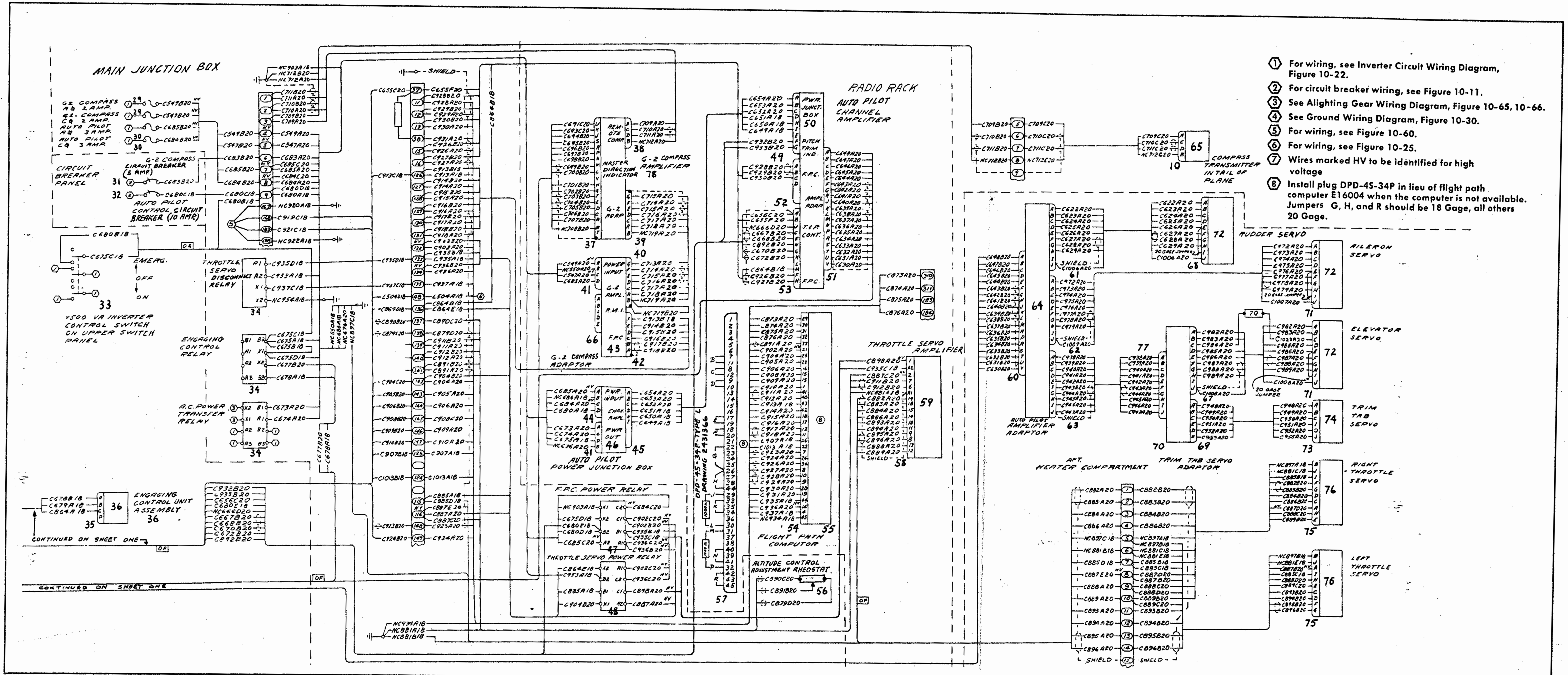
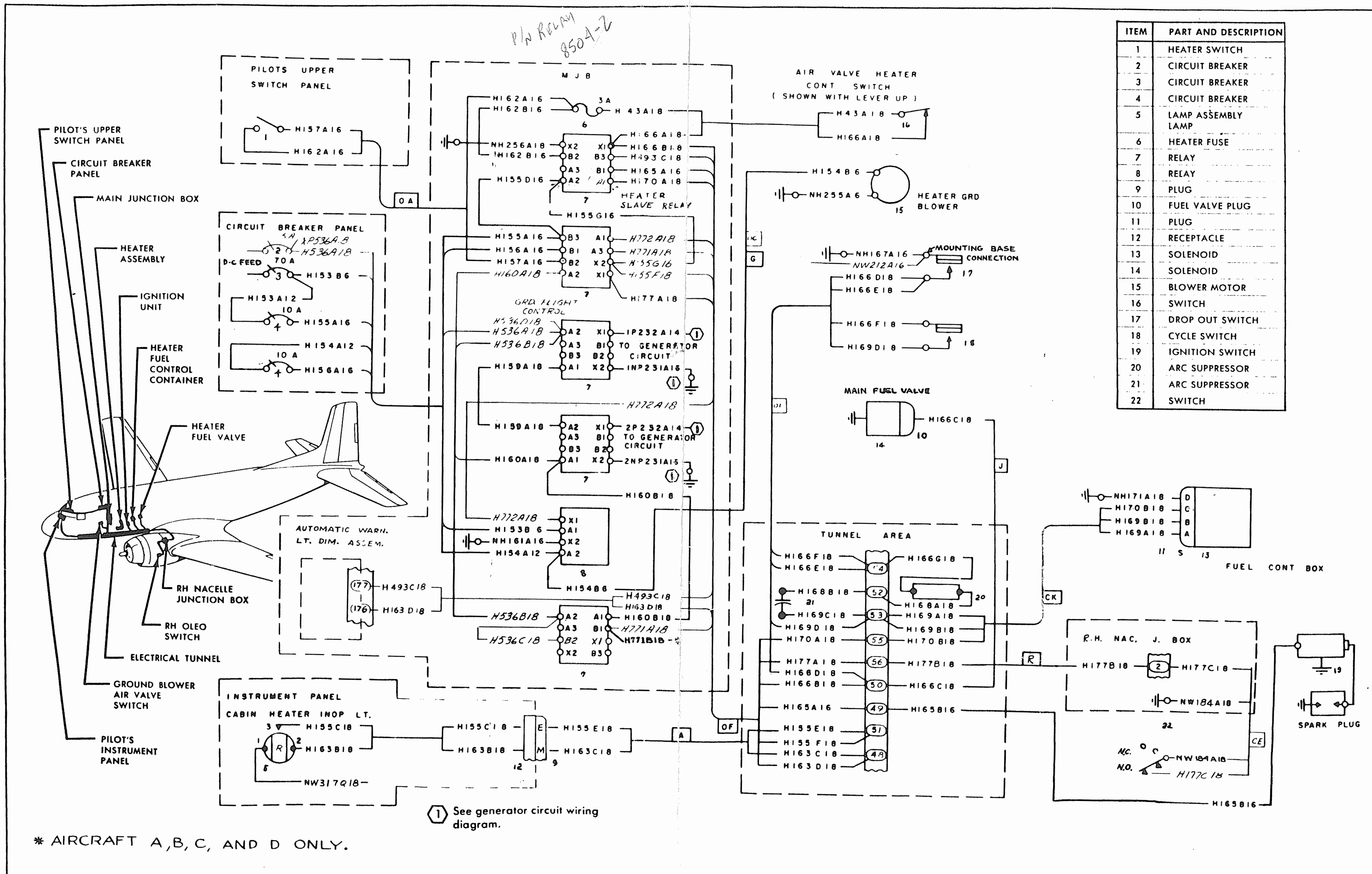


Figure 10-8 (Sheet 1 of 2 Sheets). Automatic Pilot, G-2 Compass, and Flight Path Computer (Aircraft C, D, and 69 through 96)



7980

Figure 10-8 (Sheet 2 of 2 Sheets). Automatic Pilot, G-2 Compass, and Flight Path Computer (Aircraft C, D, and 69 through 96)



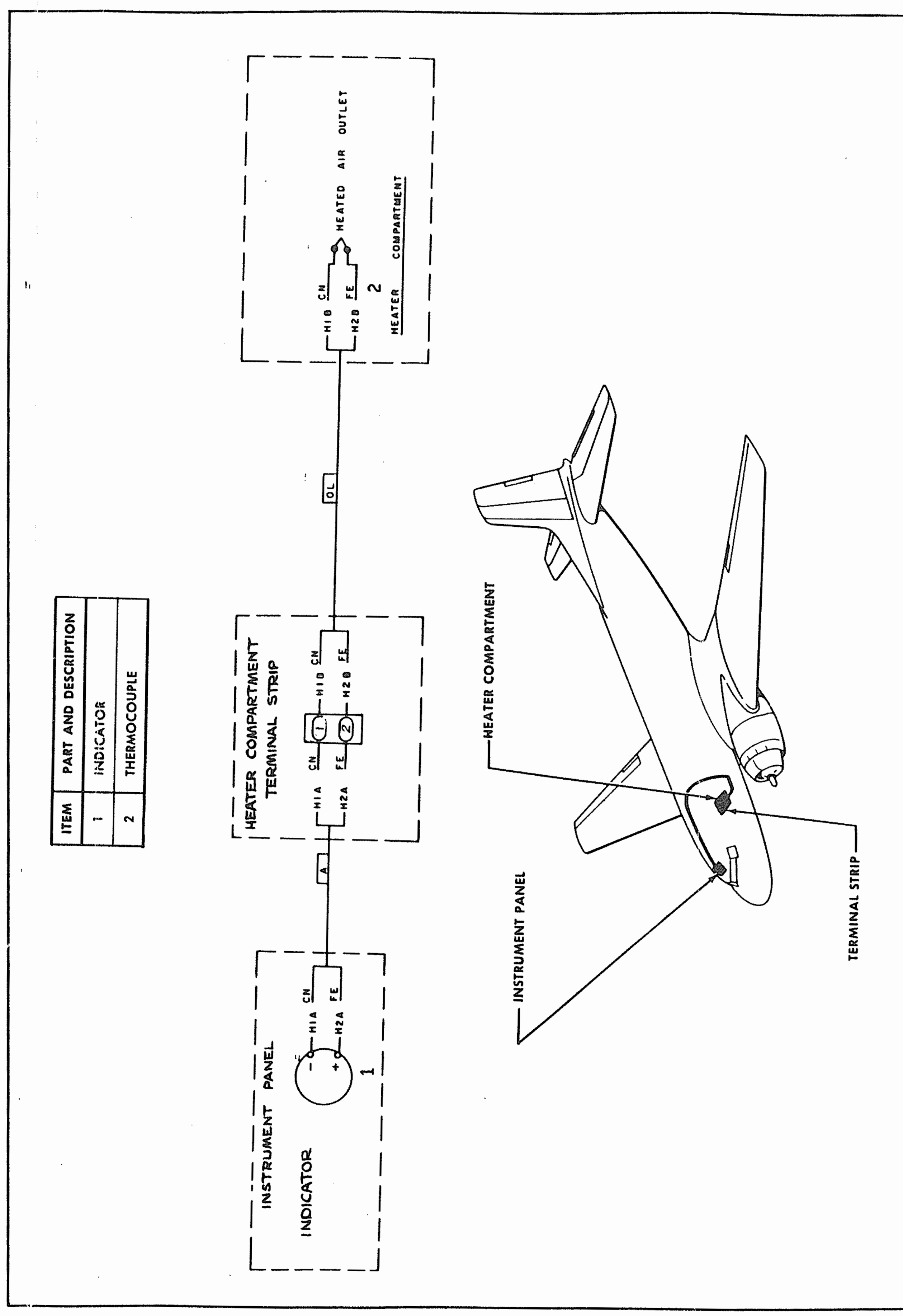
ITEM	PART AND DESCRIPTION
1	HEATER SWITCH
2	CIRCUIT BREAKER
3	CIRCUIT BREAKER
4	CIRCUIT BREAKER
5	LAMP ASSEMBLY LAMP
6	HEATER FUSE
7	RELAY
8	RELAY
9	PLUG
10	FUEL VALVE PLUG
11	PLUG
12	RECEPTACLE
13	SOLENOID
14	SOLENOID
15	BLOWER MOTOR
16	SWITCH
17	DROP OUT SWITCH
18	CYCLE SWITCH
19	IGNITION SWITCH
20	ARC SUPPRESSOR
21	ARC SUPPRESSOR
22	SWITCH

(1) See generator circuit wiring diagram.

* AIRCRAFT A, B, C, AND D ONLY.

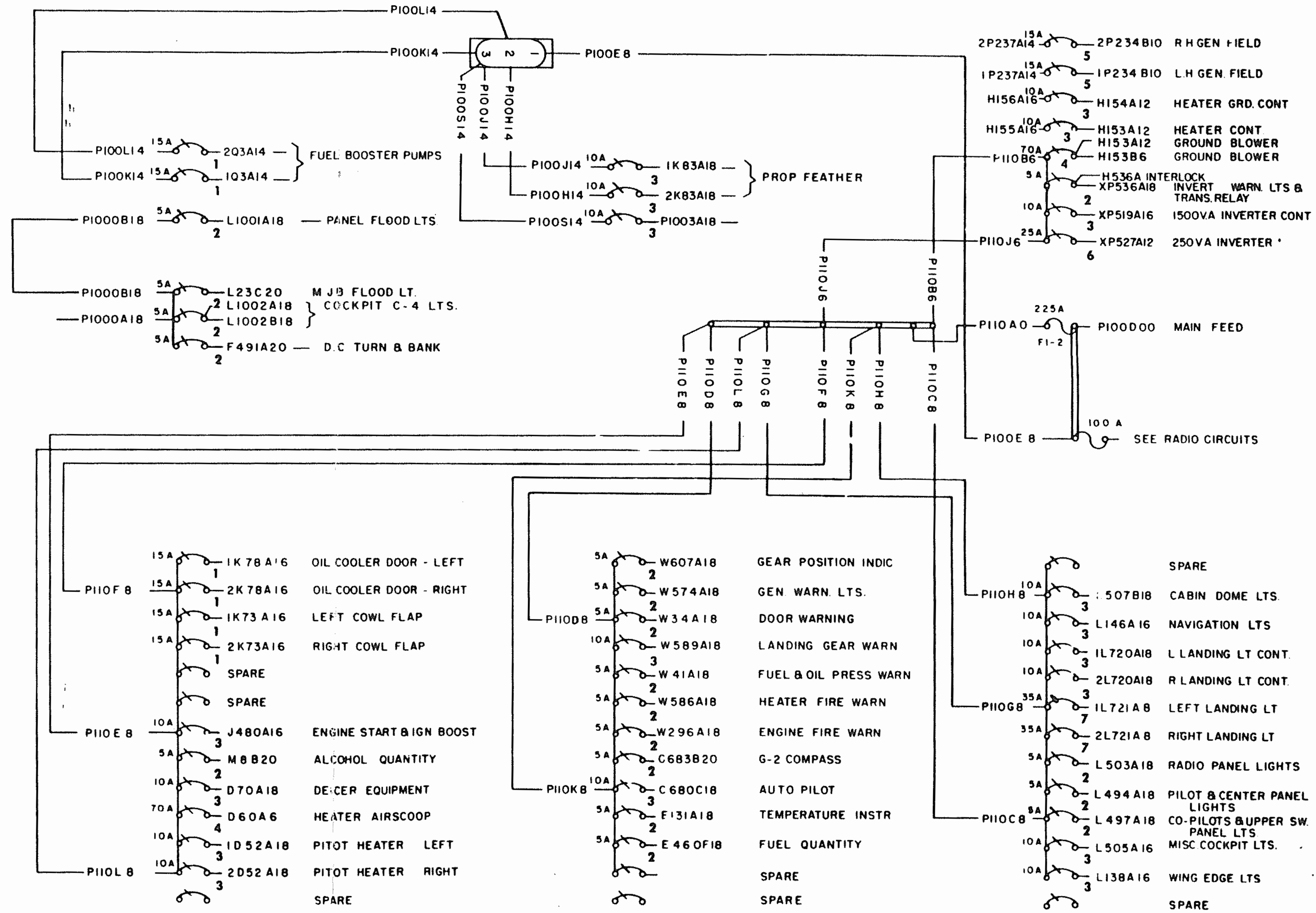
7982

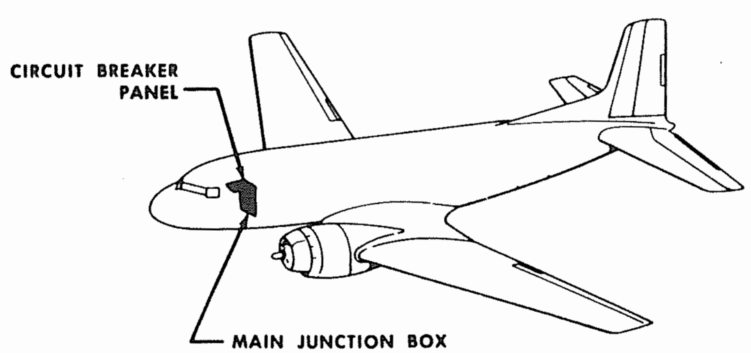
Figure 10-9. Cabin Heater



ITEM	PART AND DESCRIPTION
1	INDICATOR
2	THERMOCOUPLE

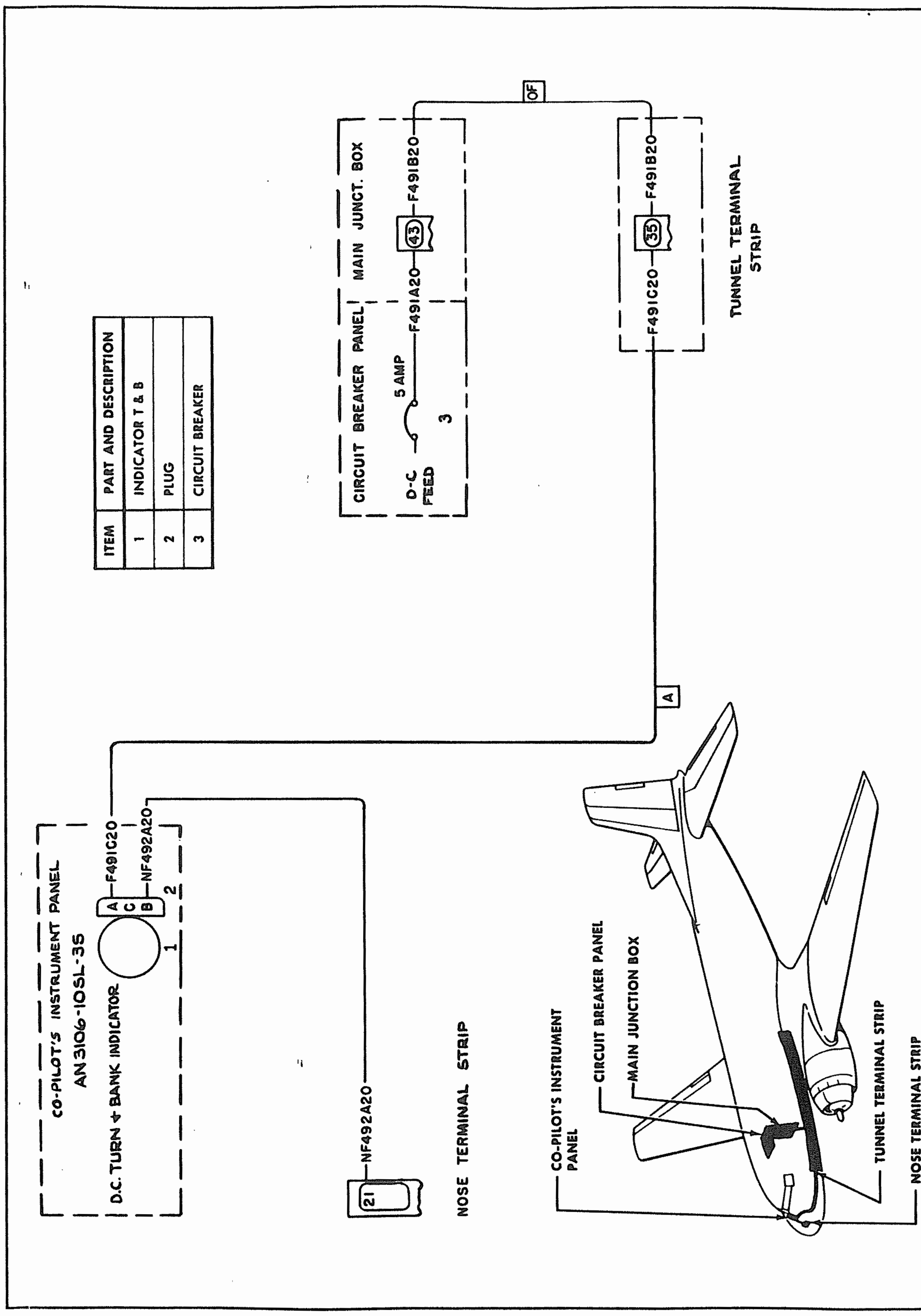
ITEM	PART AND DESCRIPTION
1	CIRCUIT BREAKER
2	CIRCUIT BREAKER
3	CIRCUIT BREAKER
4	CIRCUIT BREAKER
5	CIRCUIT BREAKER
6	CIRCUIT BREAKER
7	CIRCUIT BREAKER



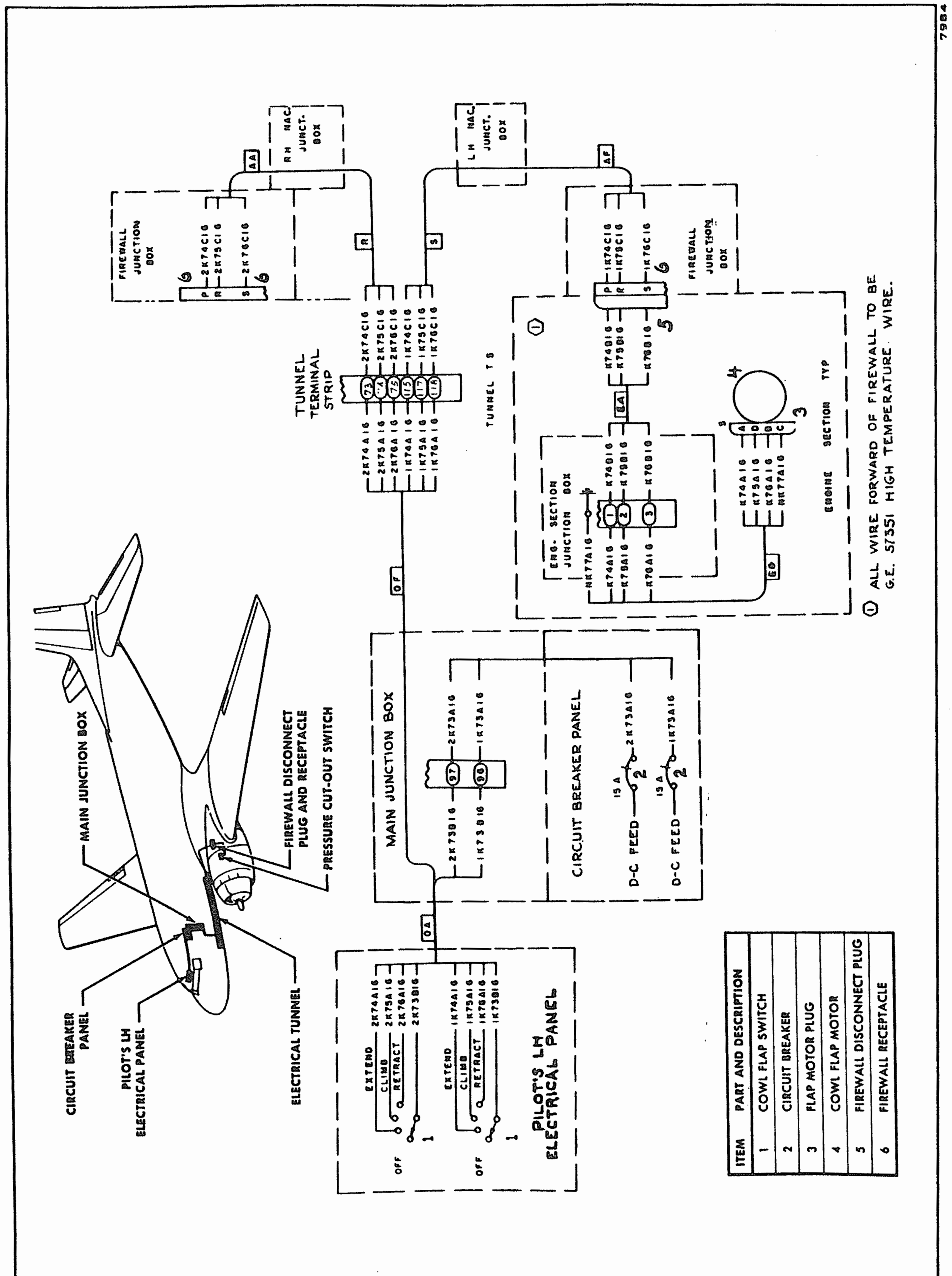


8974

Figure 10-11. Circuit Breaker Bus System

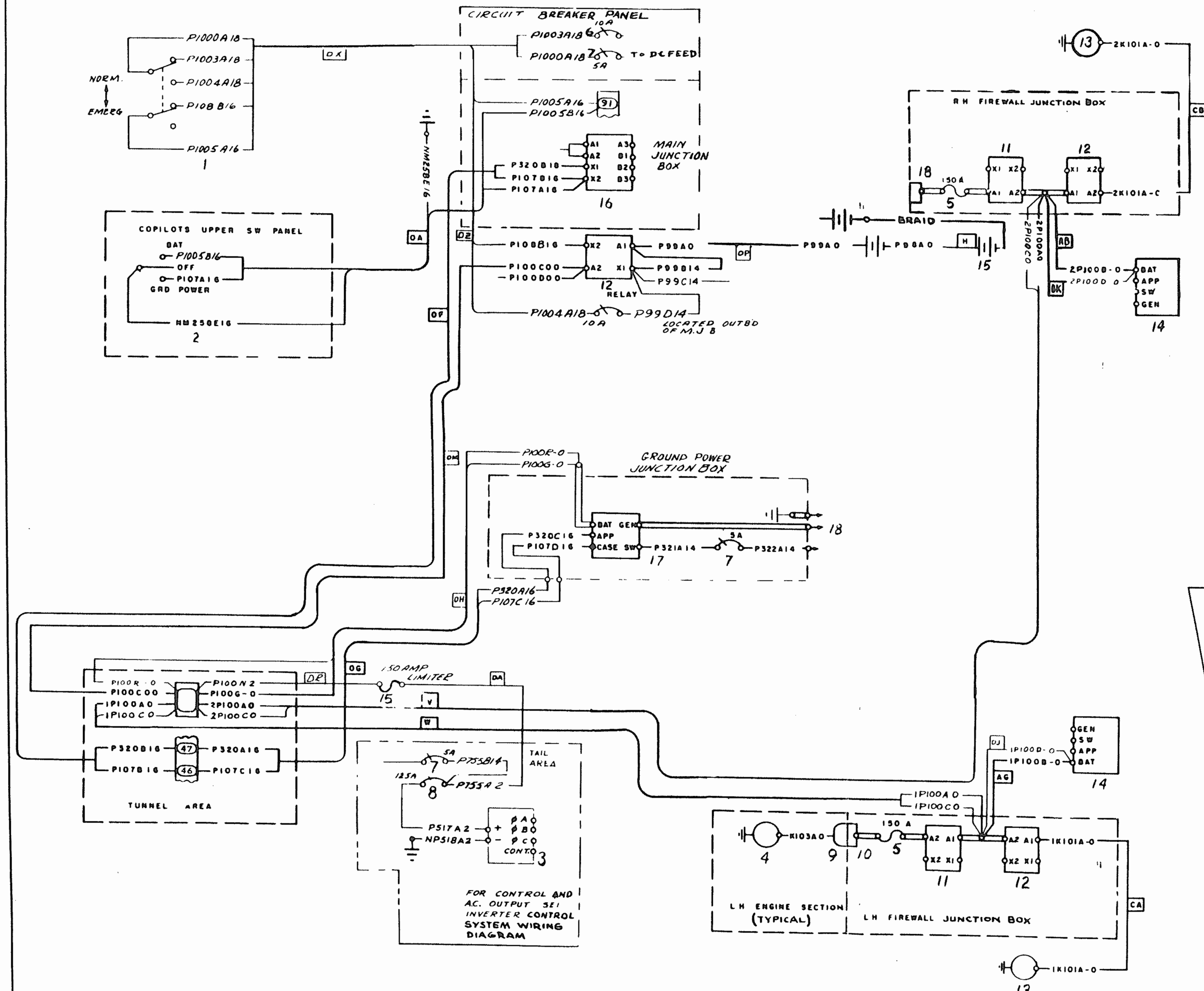


ITEM	PART AND DESCRIPTION
1	INDICATOR T & B
2	PLUG
3	CIRCUIT BREAKER

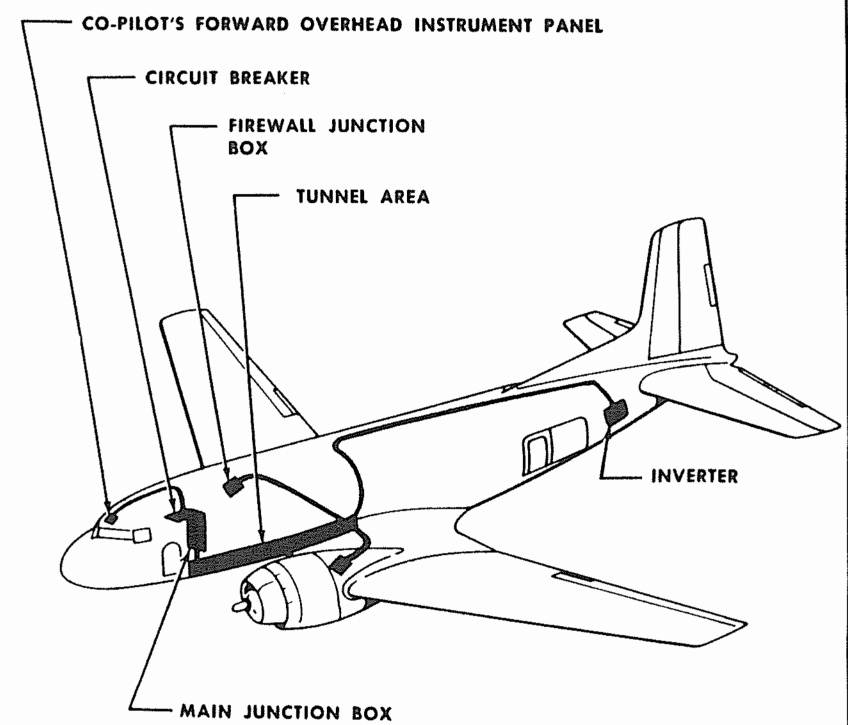


ITEM	PART AND DESCRIPTION
1	COWL FLAP SWITCH
2	CIRCUIT BREAKER
3	FLAP MOTOR PLUG
4	COWL FLAP MOTOR
5	FIREWALL DISCONNECT PLUG
6	FIREWALL RECEPTACLE

Figure 10-13. Cowl Flap Control



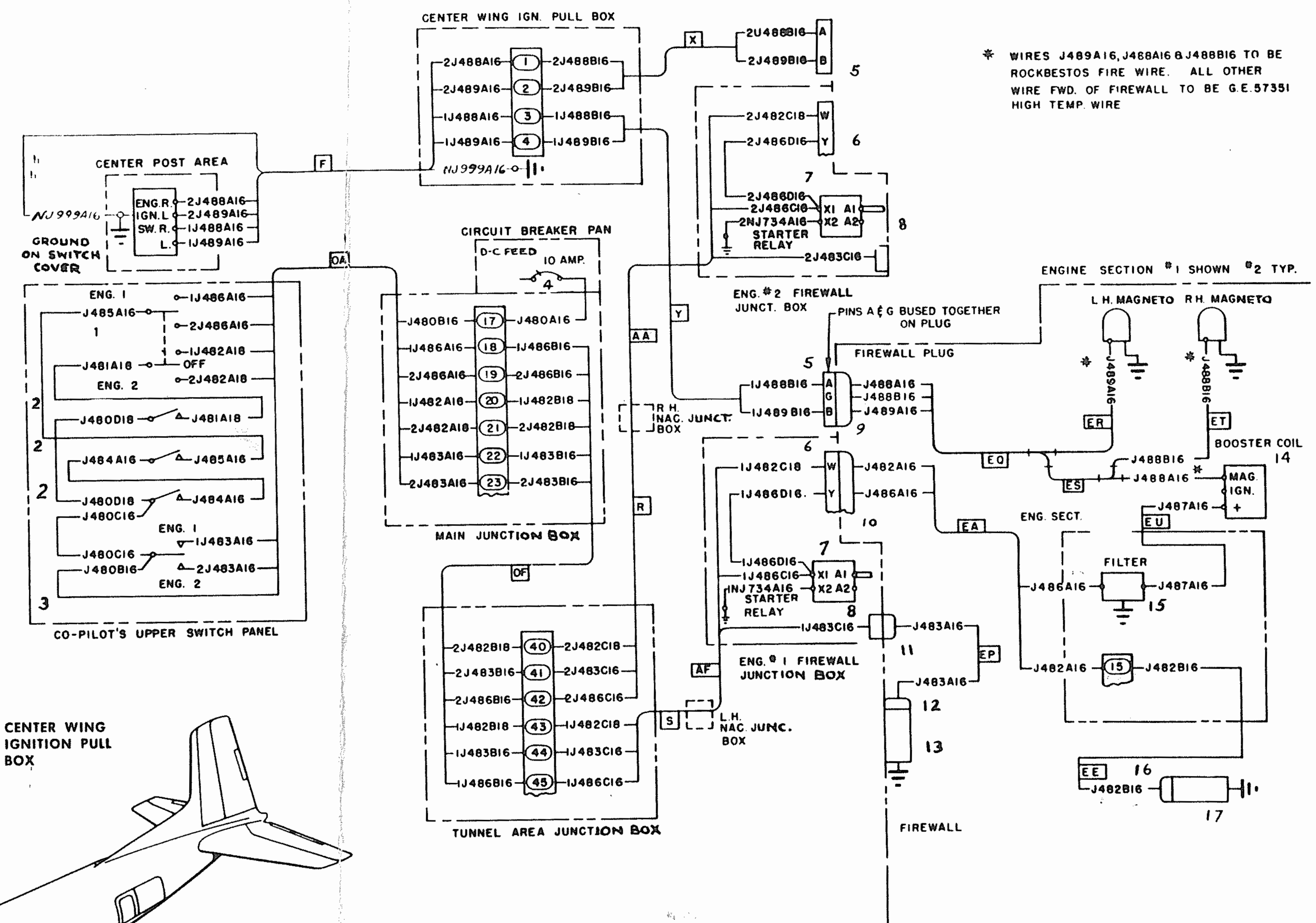
ITEM	PART AND DESCRIPTION
1	SWITCH
2	SWITCH
3	1500 VA INVERTER
4	STARTER
5	FUSE
6	CIRCUIT BREAKER
7	CIRCUIT BREAKER
8	CIRCUIT BREAKER
9	PLUG
10	RECEPTACLE
11	RELAY
12	RELAY
13	MOTOR
14	RELAY
15	BATTERY
16	RELAY
17	RELAY
18	RECEPTACLE

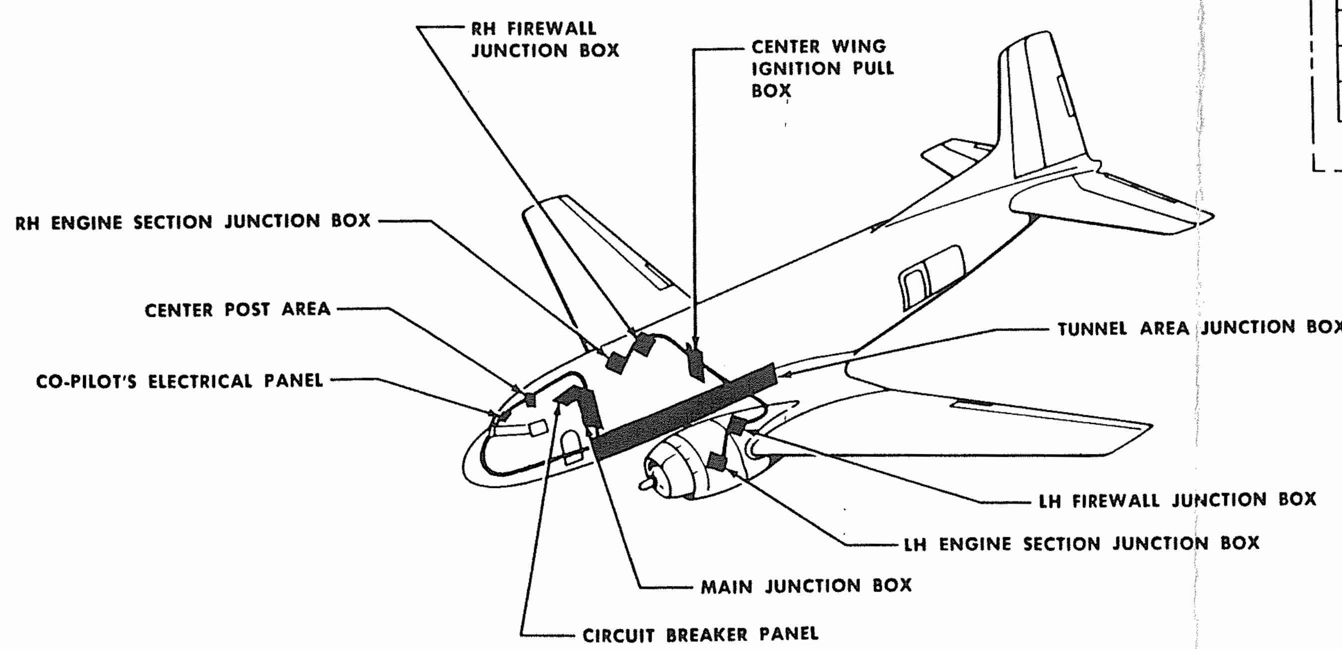


7985

Figure 10-14. D-C Bus System

ITEM	PART AND DESCRIPTION
1	SWITCH
2	SWITCH
3	SWITCH
4	CIRCUIT BREAKER (10A)
5	RECEPTACLE
6	RECEPTACLE
7	RELAY
8	RECEPTACLE
9	PLUG
10	PLUG
11	PLUG
12	PLUG
13	SOLENOID
14	INDUCTION UNIT
15	NOISE FILTER
16	PRIMER PLUG
17	PRIMER SOLENOID

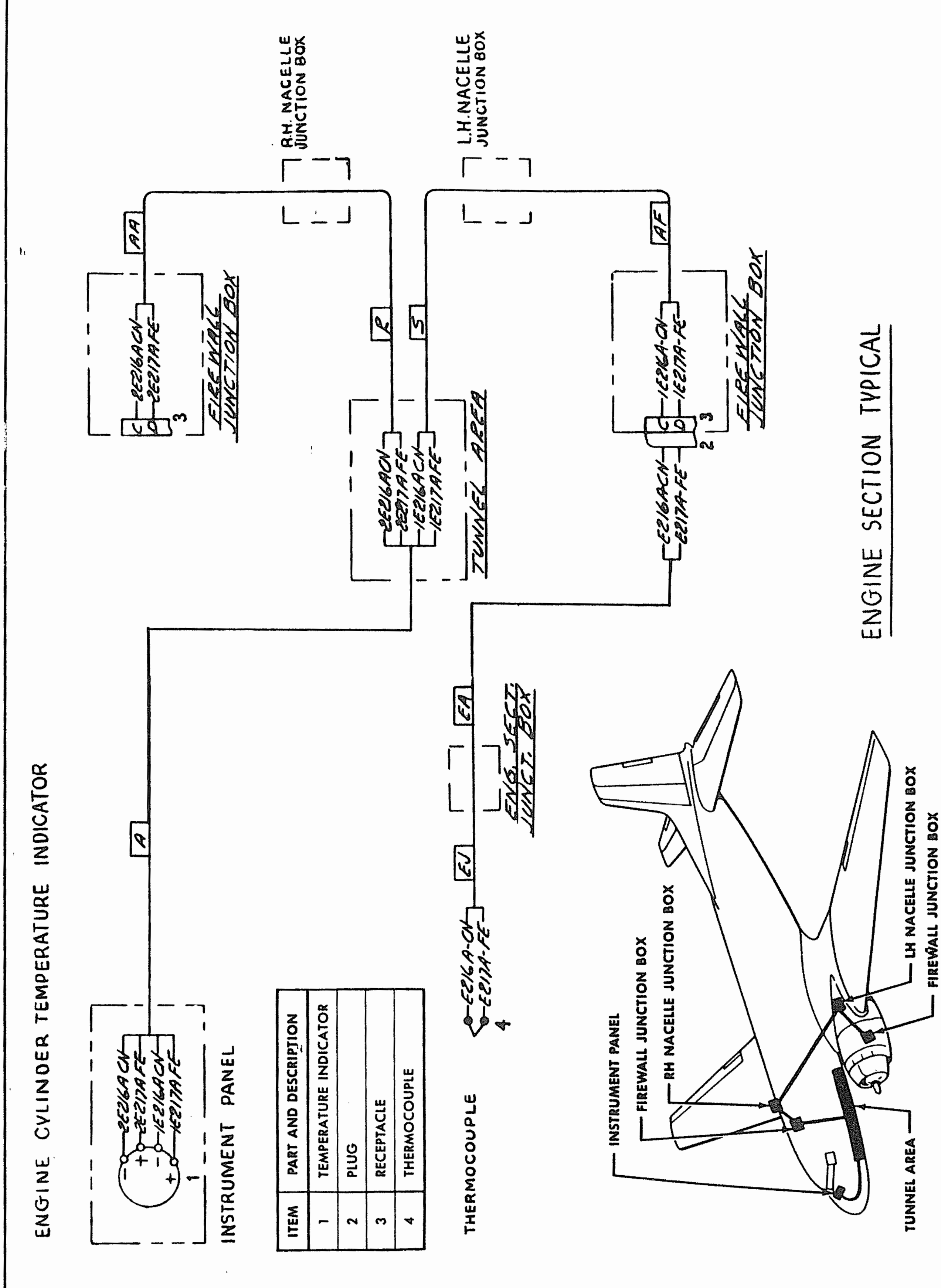




7986

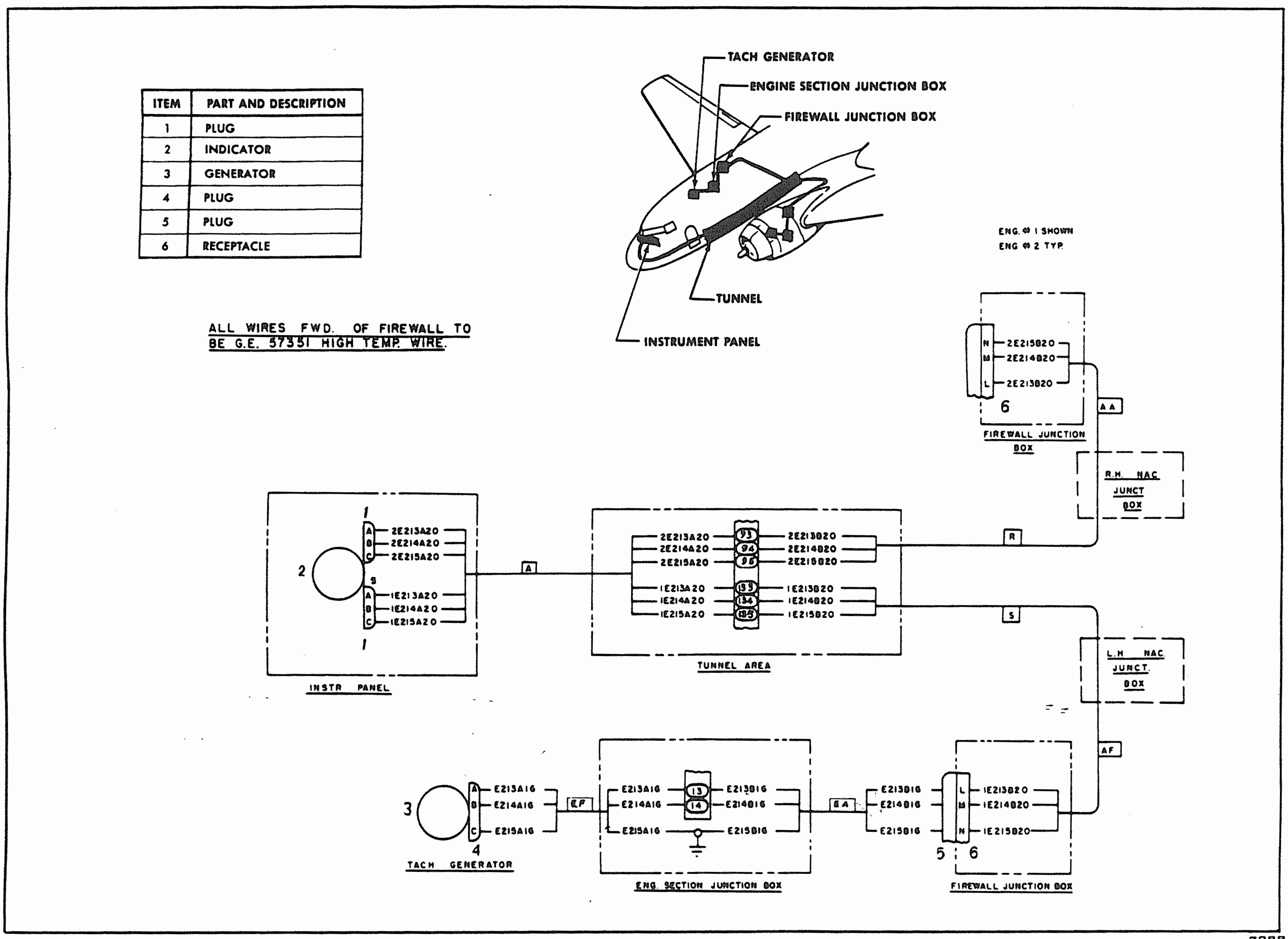
Figure 10-15. Engine Control Circuit

ENGINE CYLINDER TEMPERATURE INDICATOR



ITEM	PART AND DESCRIPTION
1	TEMPERATURE INDICATOR
2	PLUG
3	RECEPTACLE
4	THERMOCOUPLE

ENGINE SECTION TYPICAL



ITEM	PART AND DESCRIPTION
1	PLUG
2	INDICATOR
3	GENERATOR
4	PLUG
5	PLUG
6	RECEPTACLE

Figure 10-17. Engine Tachometer System

ITEM	PART AND DESCRIPTION
1	FUEL CONTROL SWITCH
2	CIRCUIT BREAKER
3	FUEL BOOSTER PUMP
4	FUEL BOOSTER PUMP PLUG

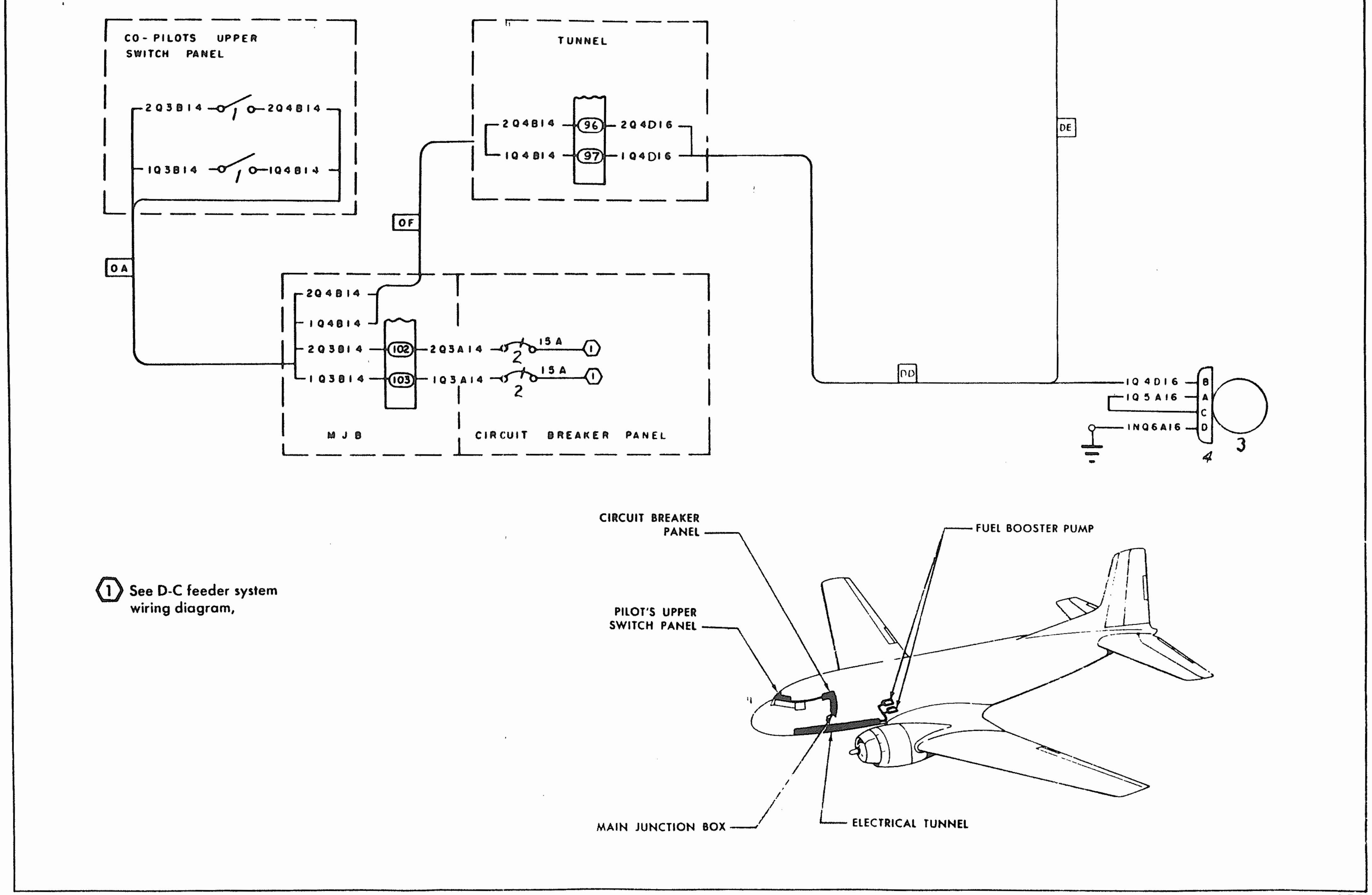


Figure 10-18. Fuel Booster Pumps

ITEM	PART AND DESCRIPTION
1	INDICATOR
2	INDICATOR
3	PLUG
4	PLUG
5	CIRCUIT BREAKER
6	PLUG
7	TRANSMITTER
8	TRANSMITTER
9	TRANSMITTER

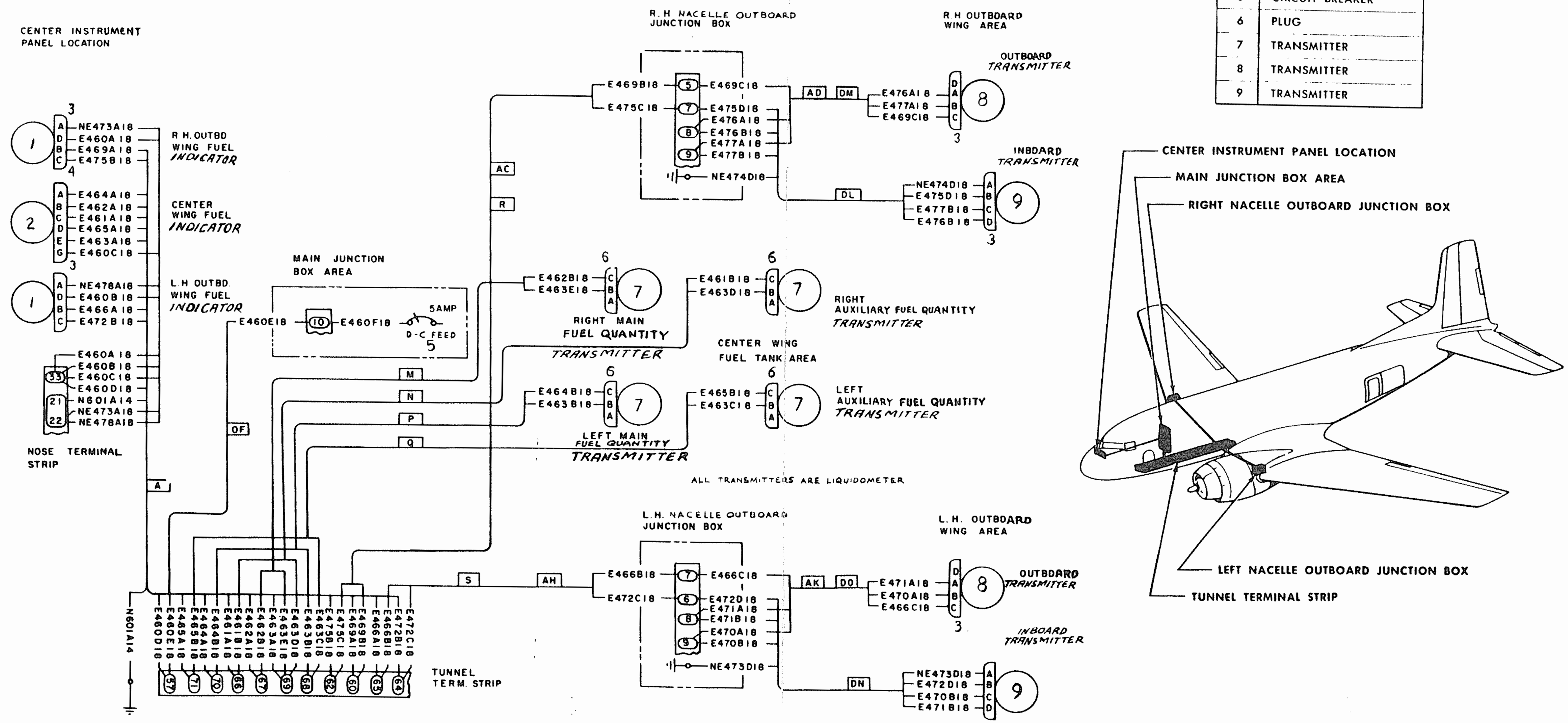
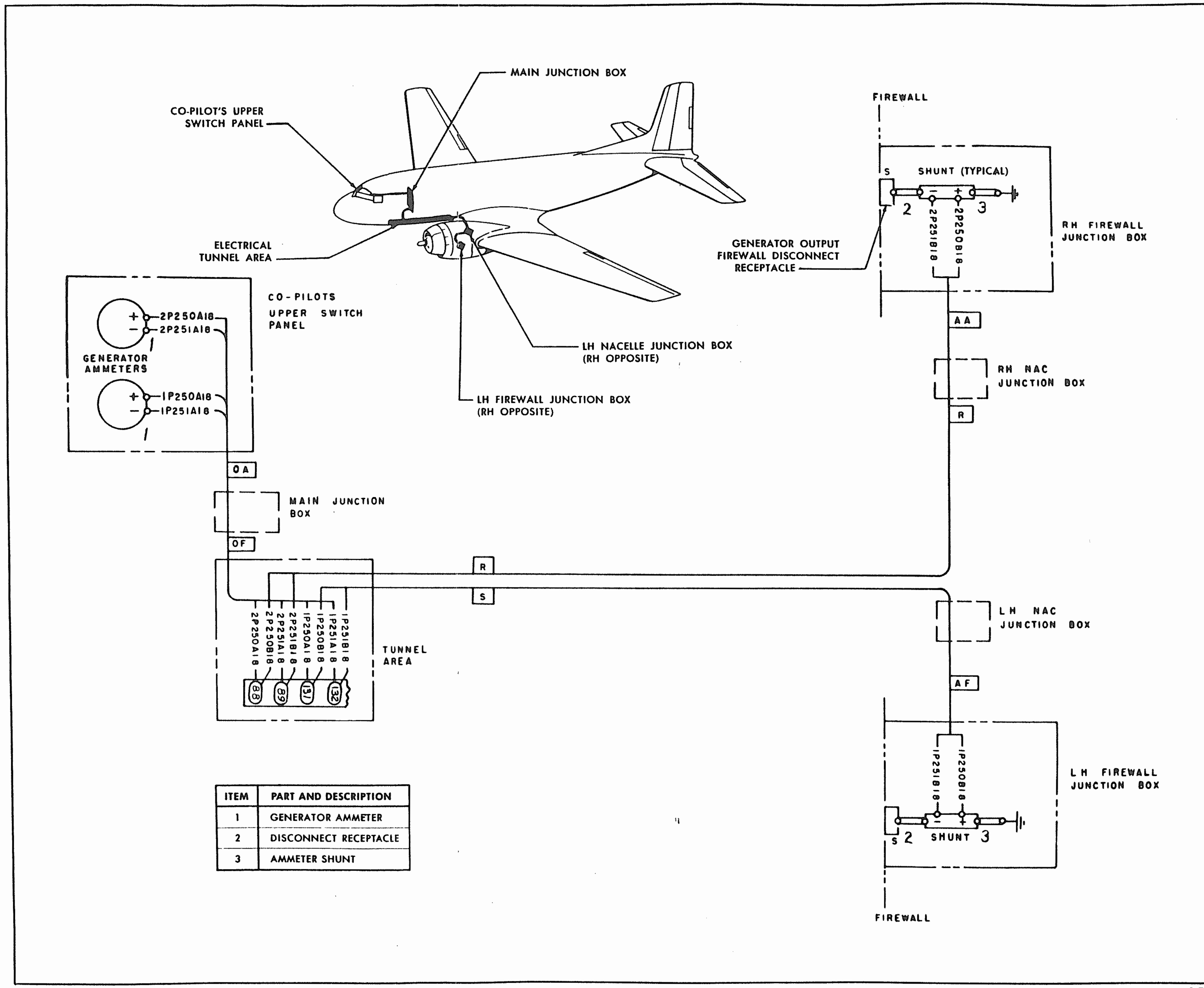


Figure 10-19. Fuel Quantity System

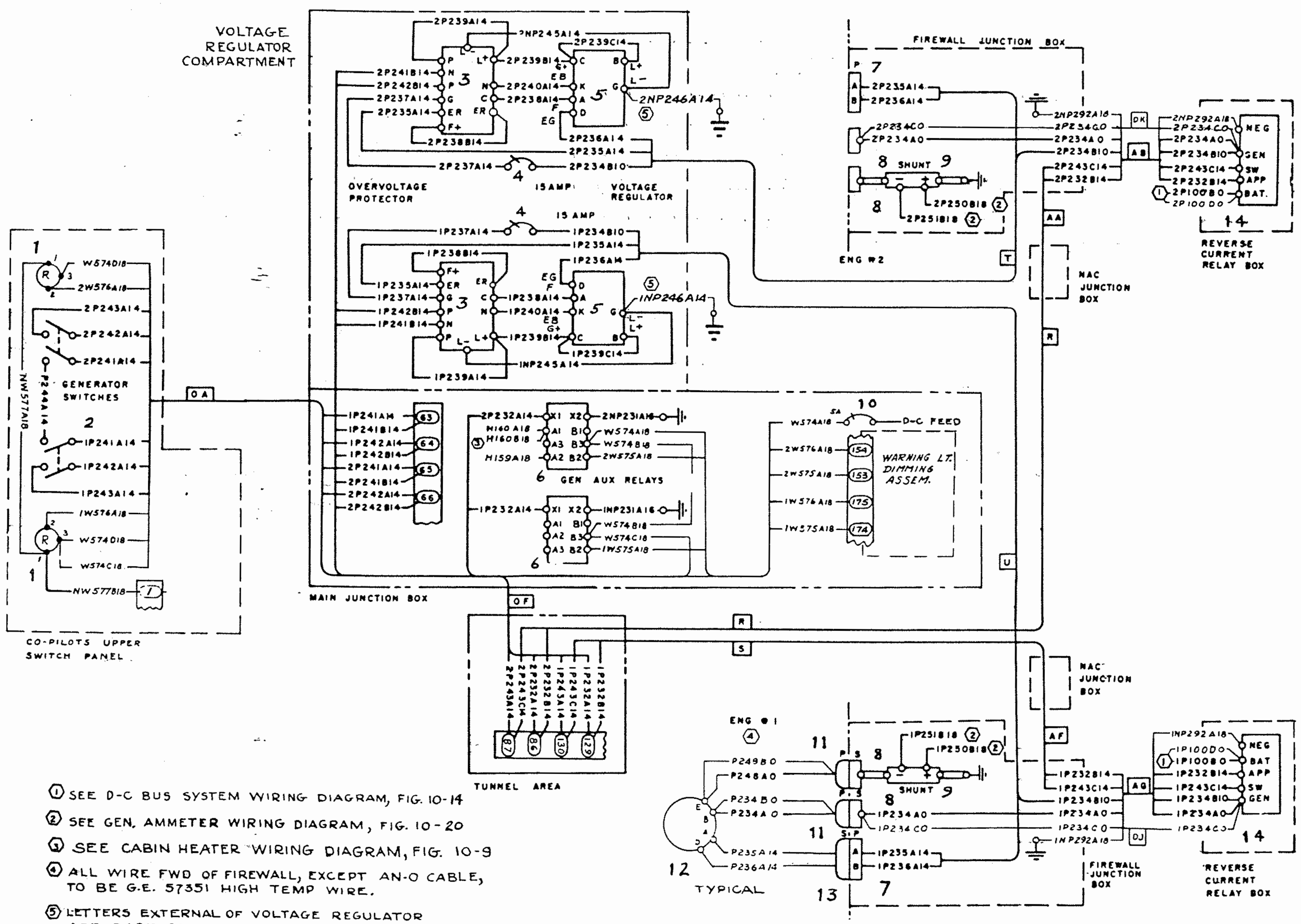
7989



ITEM	PART AND DESCRIPTION
1	GENERATOR AMMETER
2	DISCONNECT RECEPTACLE
3	AMMETER SHUNT

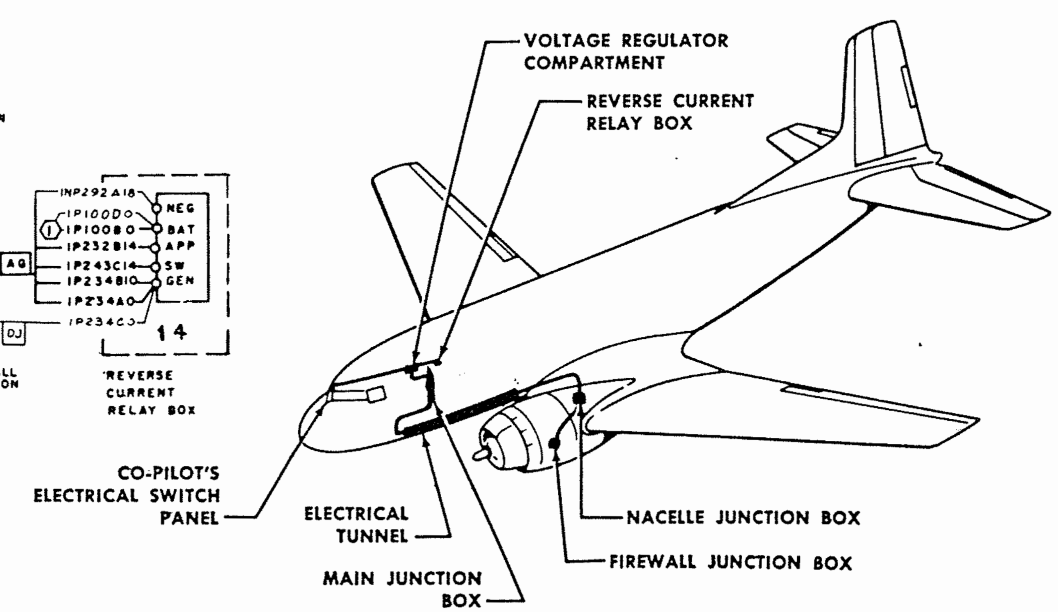
1.532

Figure 10-20. Generator Ammeter



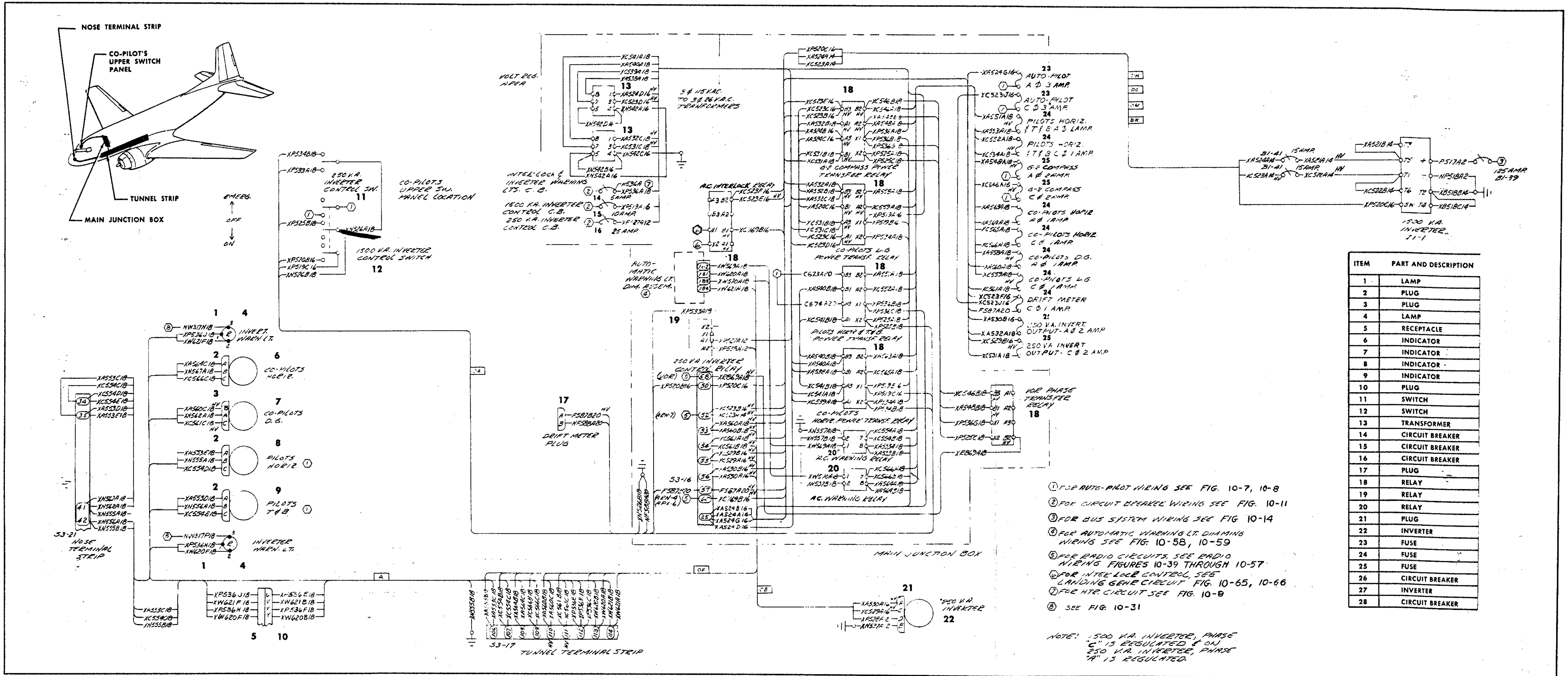
ITEM	PART AND DESCRIPTION
1	LAMP ASSEMBLY LAMP
2	SWITCH
3	RELAY
4	CIRCUIT BREAKER (15A)
5	REGULATOR
6	RELAY
7	RECEPTACLE
8	RECEPTACLE
9	SHUNT
10	CIRCUIT BREAKER (5A)
11	PLUG
12	300 AMP GENERATOR
13	PLUG
14	RELAY

① SEE D-C BUS SYSTEM WIRING DIAGRAM, FIG. 10-14
② SEE GEN. AMMETER WIRING DIAGRAM, FIG. 10-20
③ SEE CABIN HEATER WIRING DIAGRAM, FIG. 10-9
④ ALL WIRE FWD OF FIREWALL, EXCEPT AN-O CABLE, TO BE G.E. 57351 HIGH TEMP WIRE.
⑤ LETTERS EXTERNAL OF VOLTAGE REGULATOR ARE BASE CONNECTIONS



7990

Figure 10-21. Generator Circuit

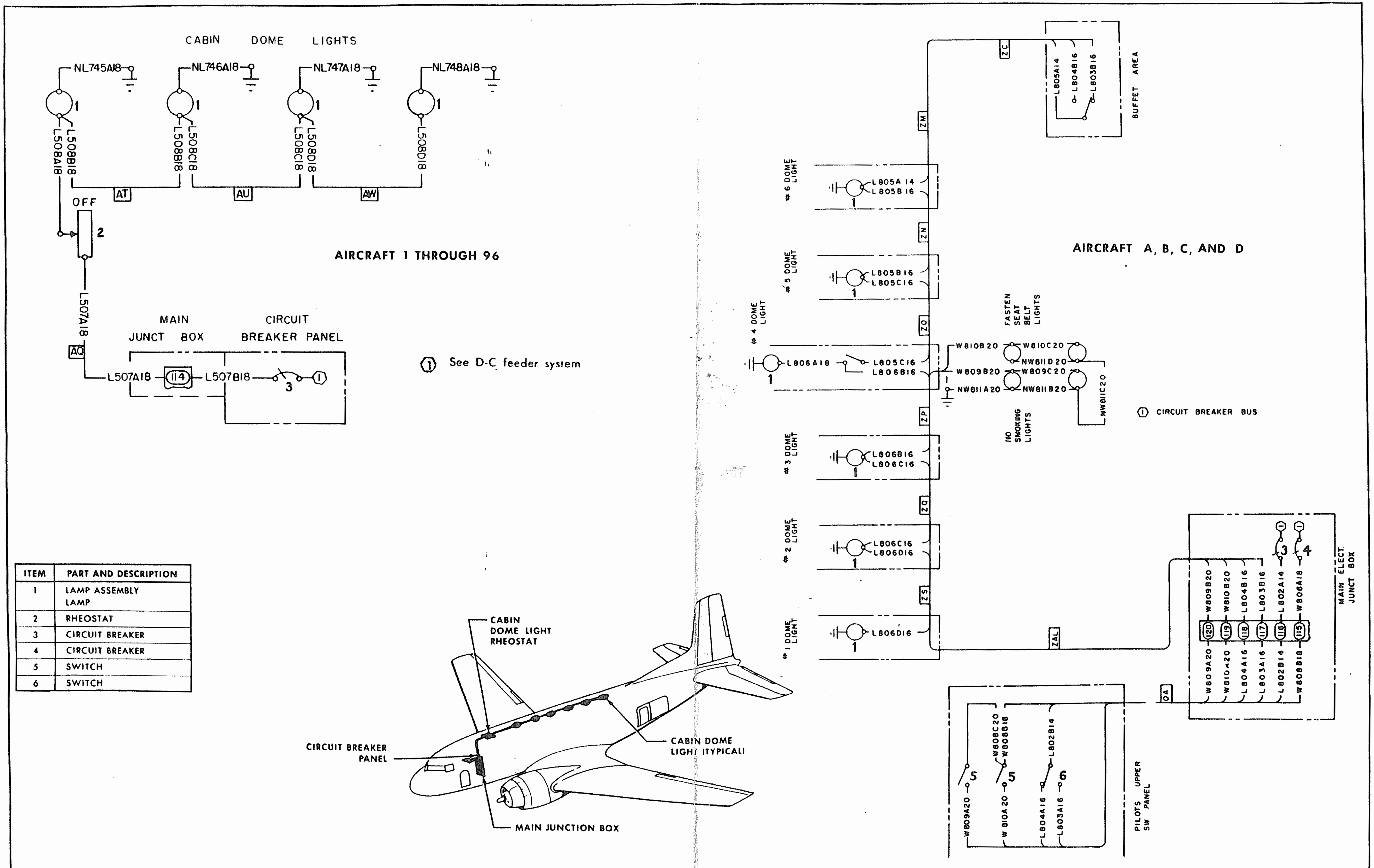


ITEM	PART AND DESCRIPTION
1	LAMP
2	PLUG
3	PLUG
4	LAMP
5	RECEPTACLE
6	INDICATOR
7	INDICATOR
8	INDICATOR
9	INDICATOR
10	PLUG
11	SWITCH
12	SWITCH
13	TRANSFORMER
14	CIRCUIT BREAKER
15	CIRCUIT BREAKER
16	CIRCUIT BREAKER
17	PLUG
18	RELAY
19	RELAY
20	RELAY
21	PLUG
22	INVERTER
23	FUSE
24	FUSE
25	FUSE
26	CIRCUIT BREAKER
27	INVERTER
28	CIRCUIT BREAKER

①FOR AUTO-PILOT WIRING SEE FIG. 10-7, 10-8
②FOR CIRCUIT BREAKER WIRING SEE FIG. 10-11
③FOR BUS SYSTEM WIRING SEE FIG 10-14
④FOR AUTOMATIC WARNING LT. DIMMING WIRING SEE FIG. 10-58, 10-59
⑤FOR RADIO CIRCUITS, SEE RADIO WIRING FIGURES 10-39 THROUGH 10-57
⑥FOR INTERLOCK CONTROL, SEE LANDING GEAR CIRCUIT FIG. 10-65, 10-66
⑦FOR HTR. CIRCUIT SEE FIG. 10-8
⑧ SEE FIG. 10-31

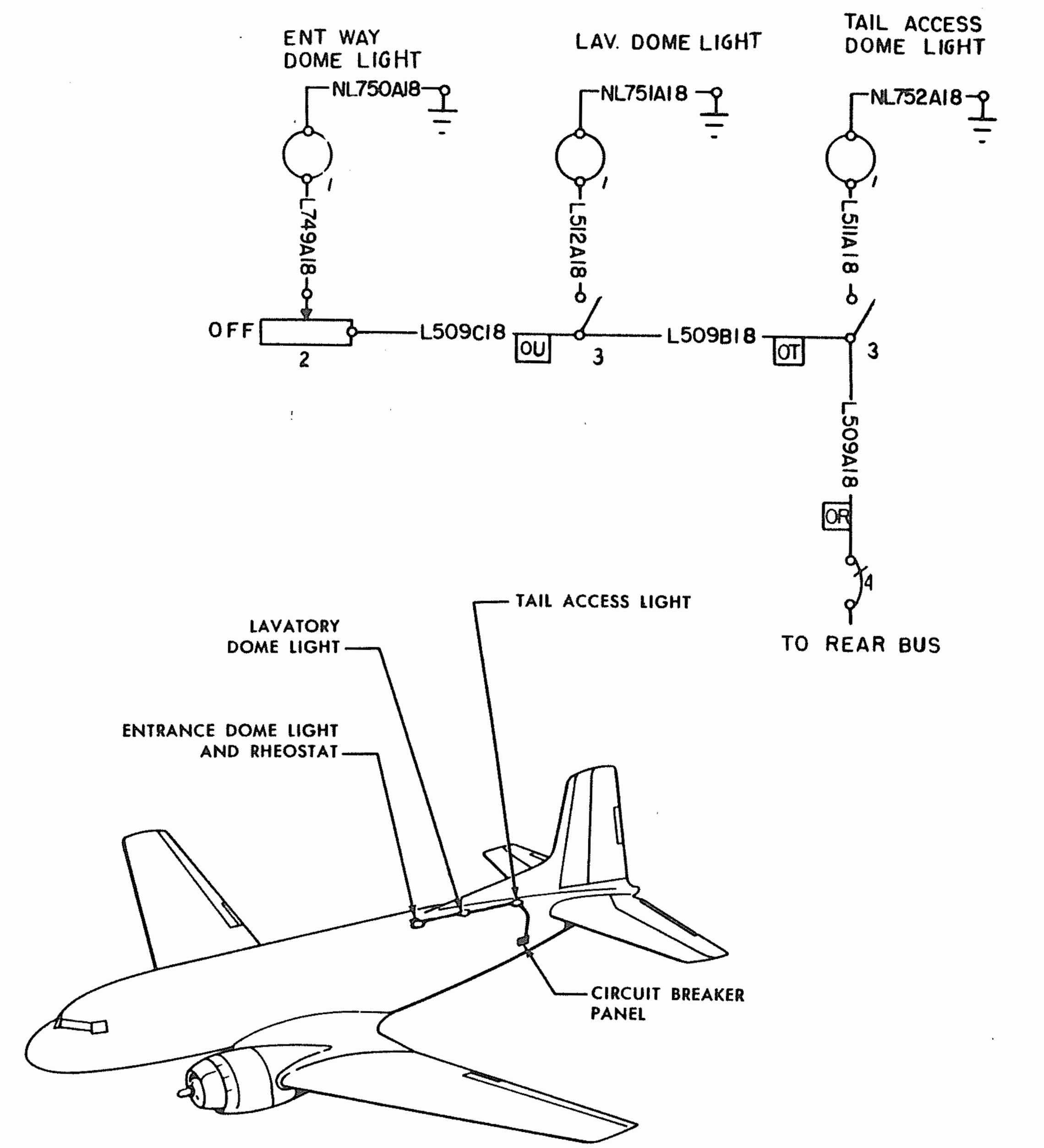
NOTE: 1500 V.A. INVERTER, PHASE "C" IS REGULATED & ON 250 V.A. INVERTER, PHASE "A" IS REGULATED.

Figure 10-22. Inverter Control System



ITEM	PART AND DESCRIPTION
1	LAMP ASSEMBLY LAMP
2	RHEOSTAT
3	CIRCUIT BREAKER
4	CIRCUIT BREAKER
5	SWITCH
6	SWITCH

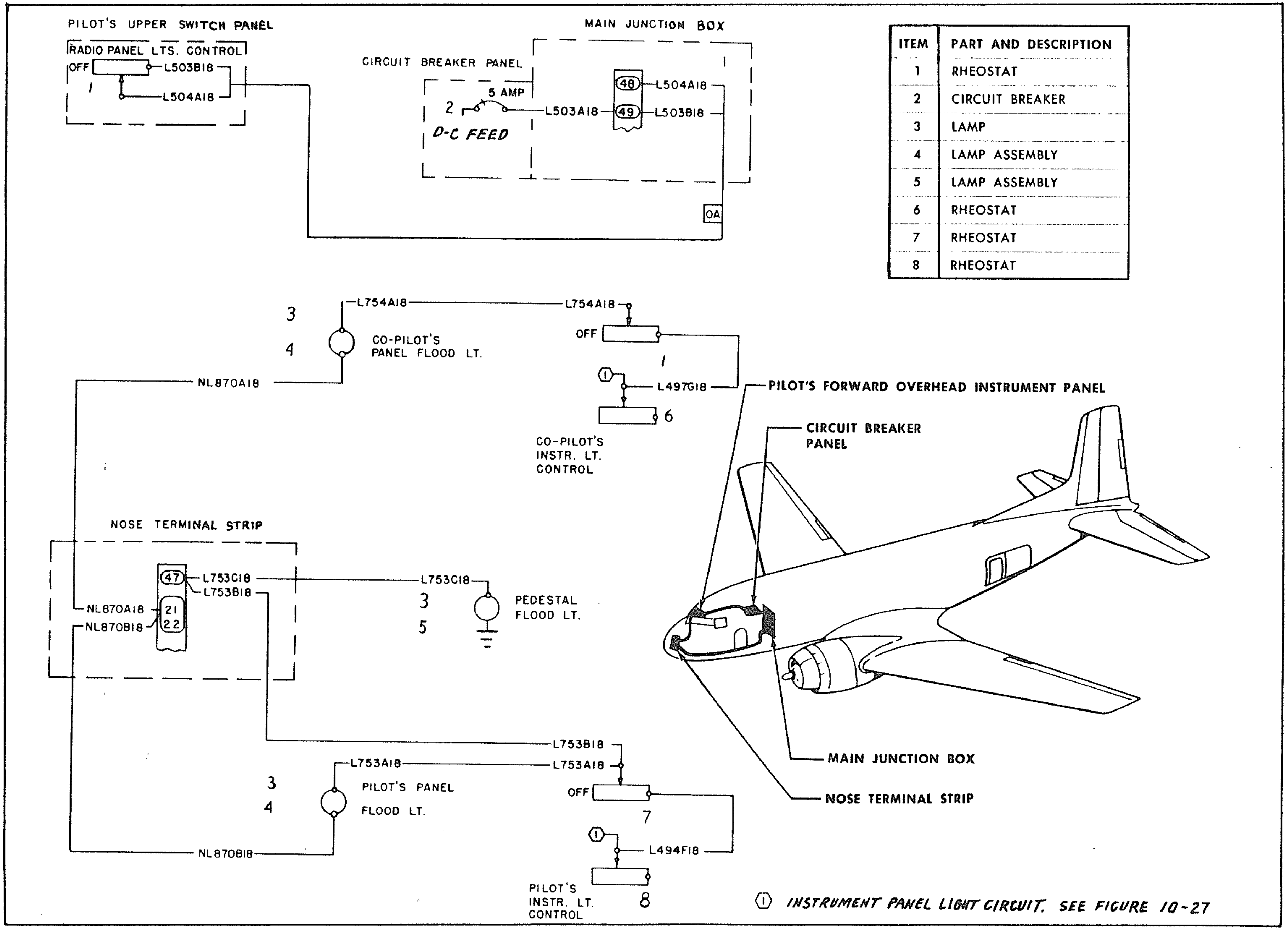
Figure 10-23. Lights — Cabin Dome



ITEM	PART AND DESCRIPTION
1	LAMP LAMP ASSEMBLY
2	RHEOSTAT
3	SWITCH
4	CIRCUIT BREAKER (5A)

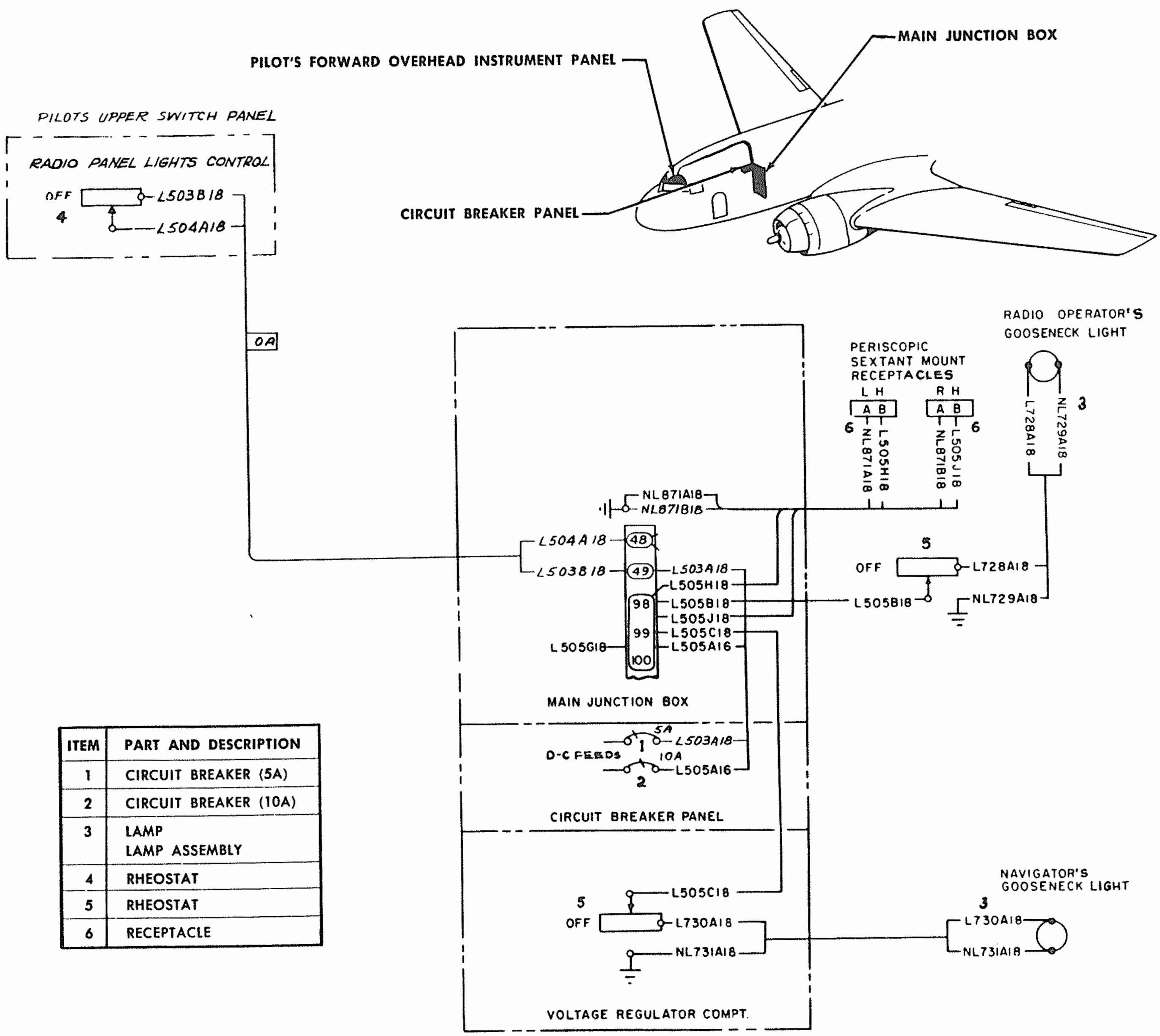
1,534

Figure 10-24. Lights — Cabin Entrance, Lavatory, and Tail Access Dome (Aircraft 1 through 96)



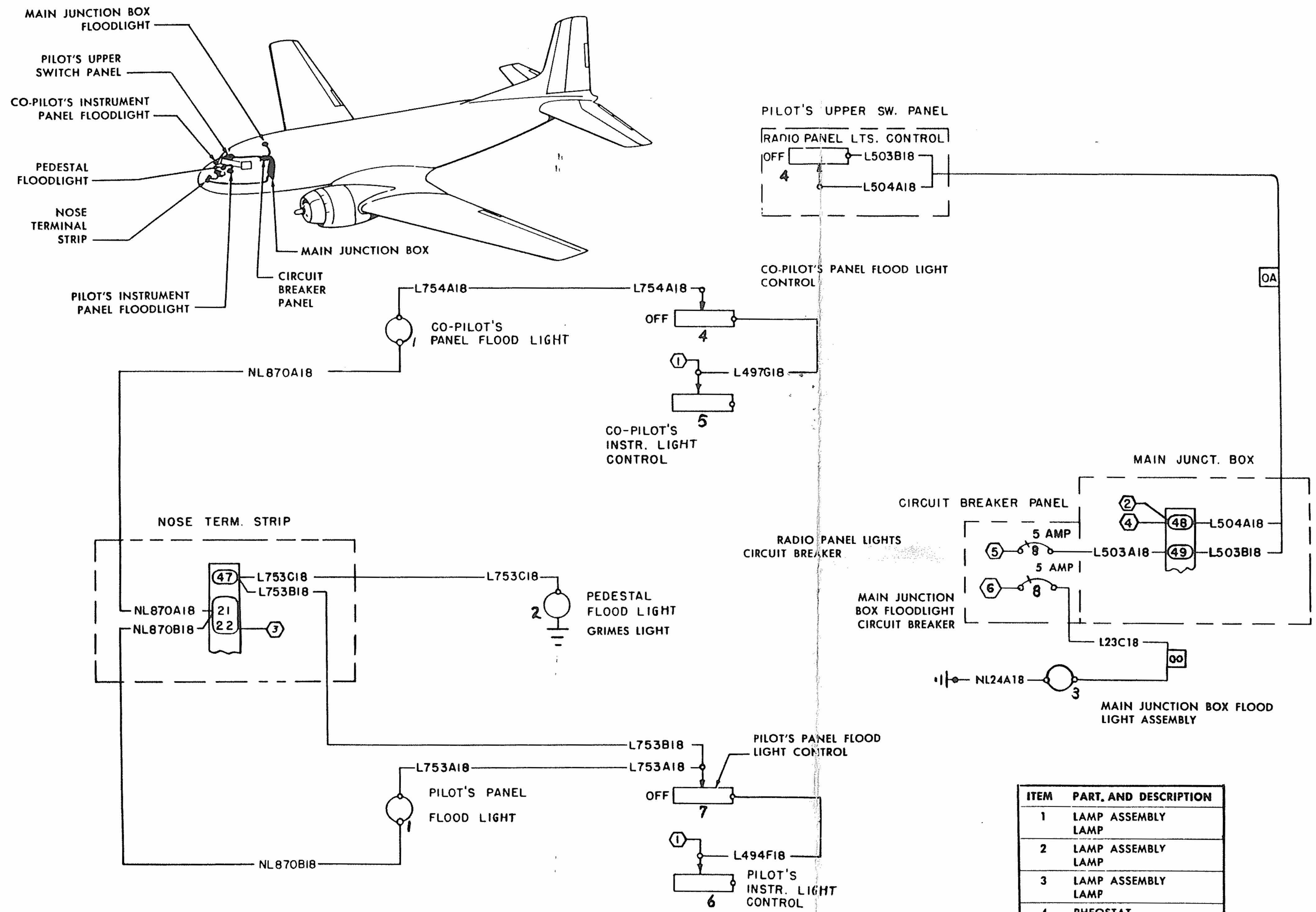
ITEM	PART AND DESCRIPTION
1	RHEOSTAT
2	CIRCUIT BREAKER
3	LAMP
4	LAMP ASSEMBLY
5	LAMP ASSEMBLY
6	RHEOSTAT
7	RHEOSTAT
8	RHEOSTAT

Figure 10-25. Lights — Cockpit Flood



ITEM	PART AND DESCRIPTION
1	CIRCUIT BREAKER (5A)
2	CIRCUIT BREAKER (10A)
3	LAMP LAMP ASSEMBLY
4	RHEOSTAT
5	RHEOSTAT
6	RECEPTACLE

Figure 10-26. Lights — Cockpit Miscellaneous



ITEM	PART. AND DESCRIPTION
1	LAMP ASSEMBLY LAMP
2	LAMP ASSEMBLY LAMP
3	LAMP ASSEMBLY LAMP
4	RHEOSTAT
5	RHEOSTAT
6	RHEOSTAT
7	RHEOSTAT
8	CIRCUIT BREAKER (5A)

(1) See instrument panel light circuit wiring diagram.

(2) See radio power feed circuit wiring diagram.

(3) See fuel quantity wiring diagram.

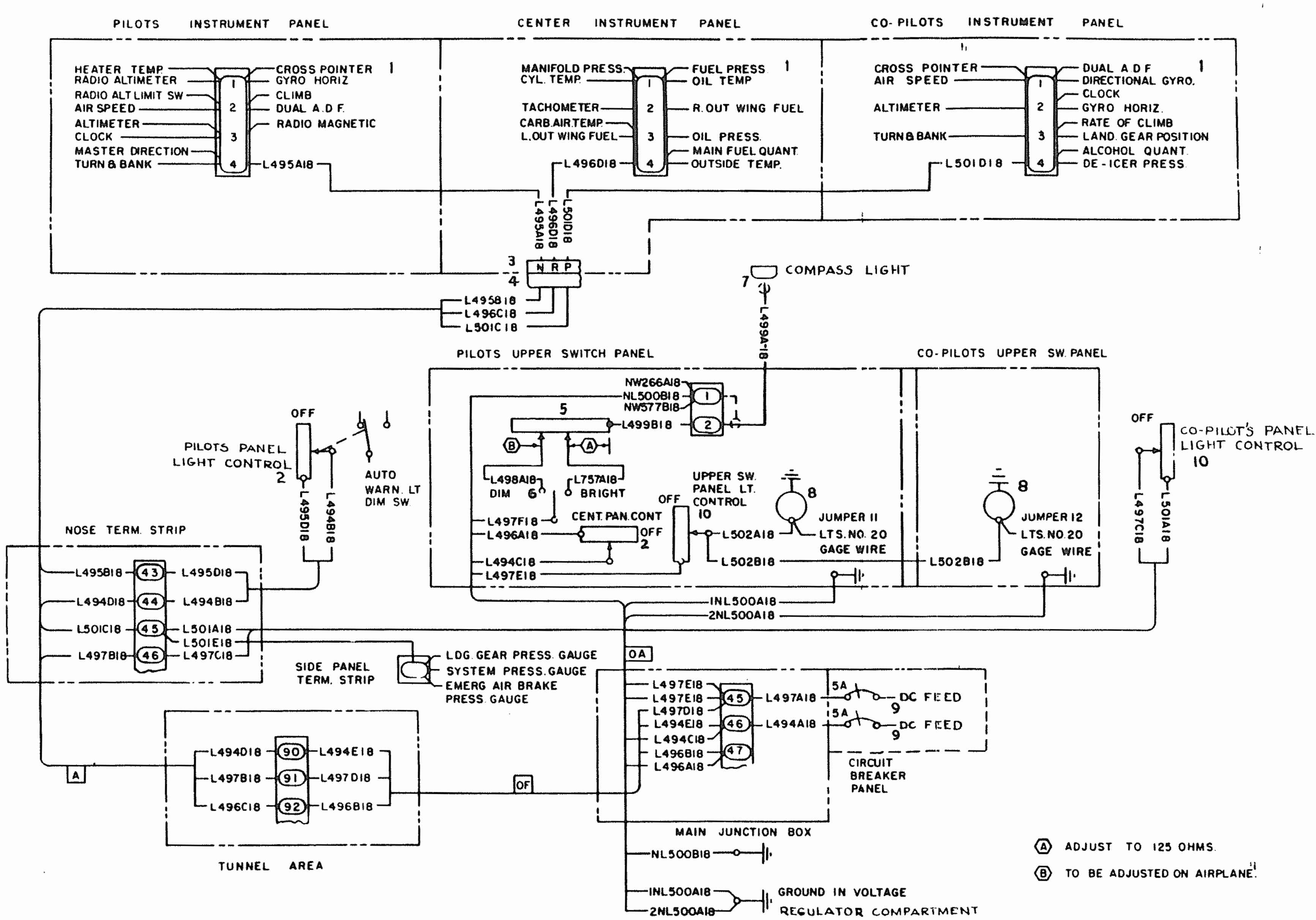
(4) See automatic pilot, G-2 compass, and flight path circuit wiring diagram.

(5) See circuit breaker bus wiring diagram.

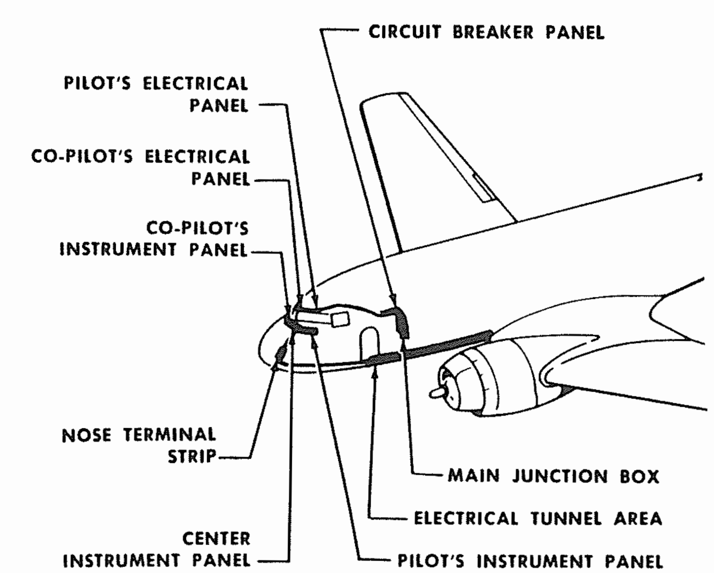
(6) See bus system wiring diagram.

1,540

Figure 10-27. Lights — Flood, Main Junction Box, and Panel

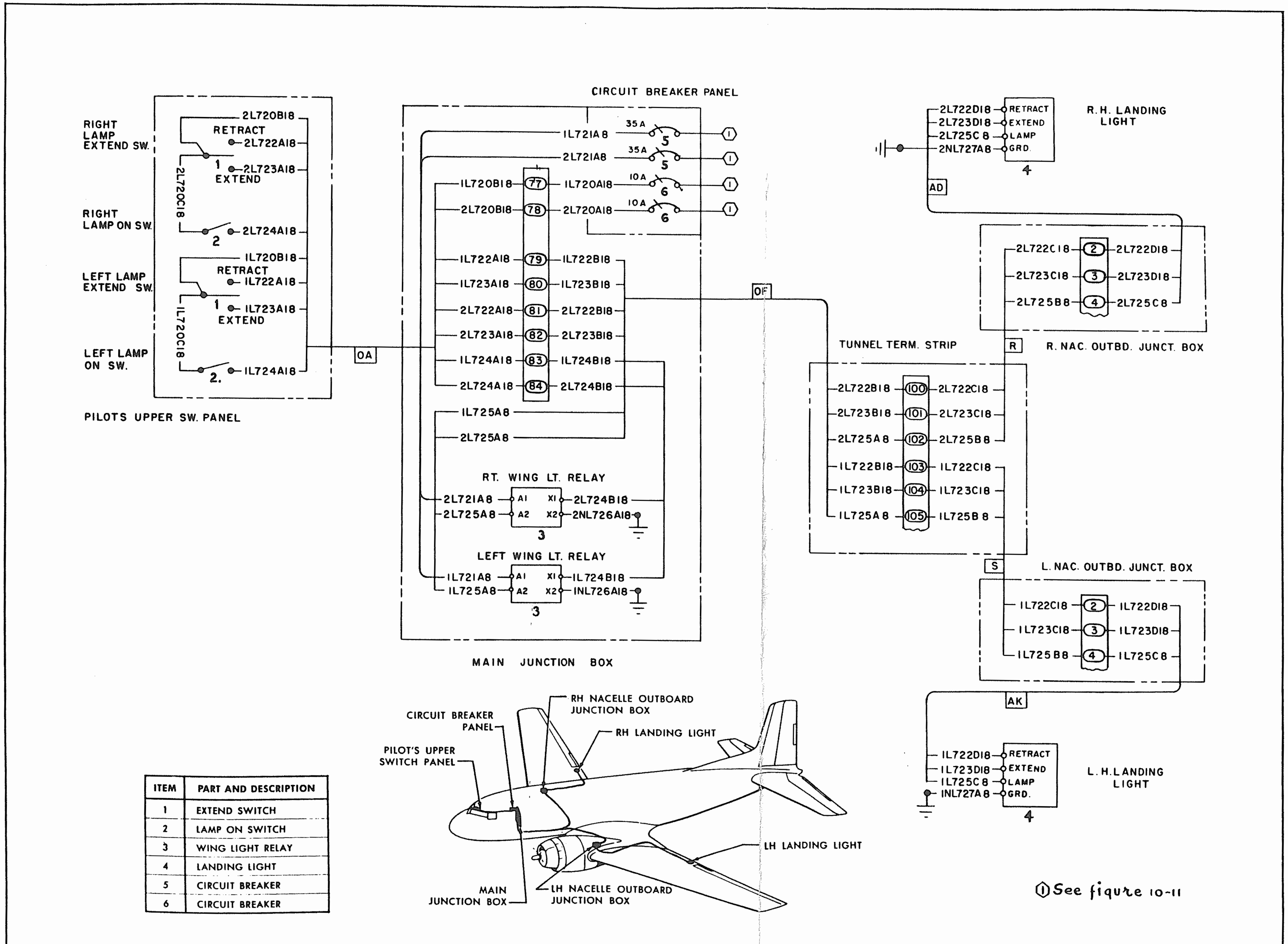


ITEM	PART AND DESCRIPTION
1	LAMP ASSEMBLY LAMP
2	RHEOSTAT
3	RECEPTACLE
4	PLUG
5	RESISTOR
6	SWITCH
7	PLUG
8	LAMP ASSEMBLY LAMP
9	CIRCUIT BREAKER (5A)
10	RHEOSTAT



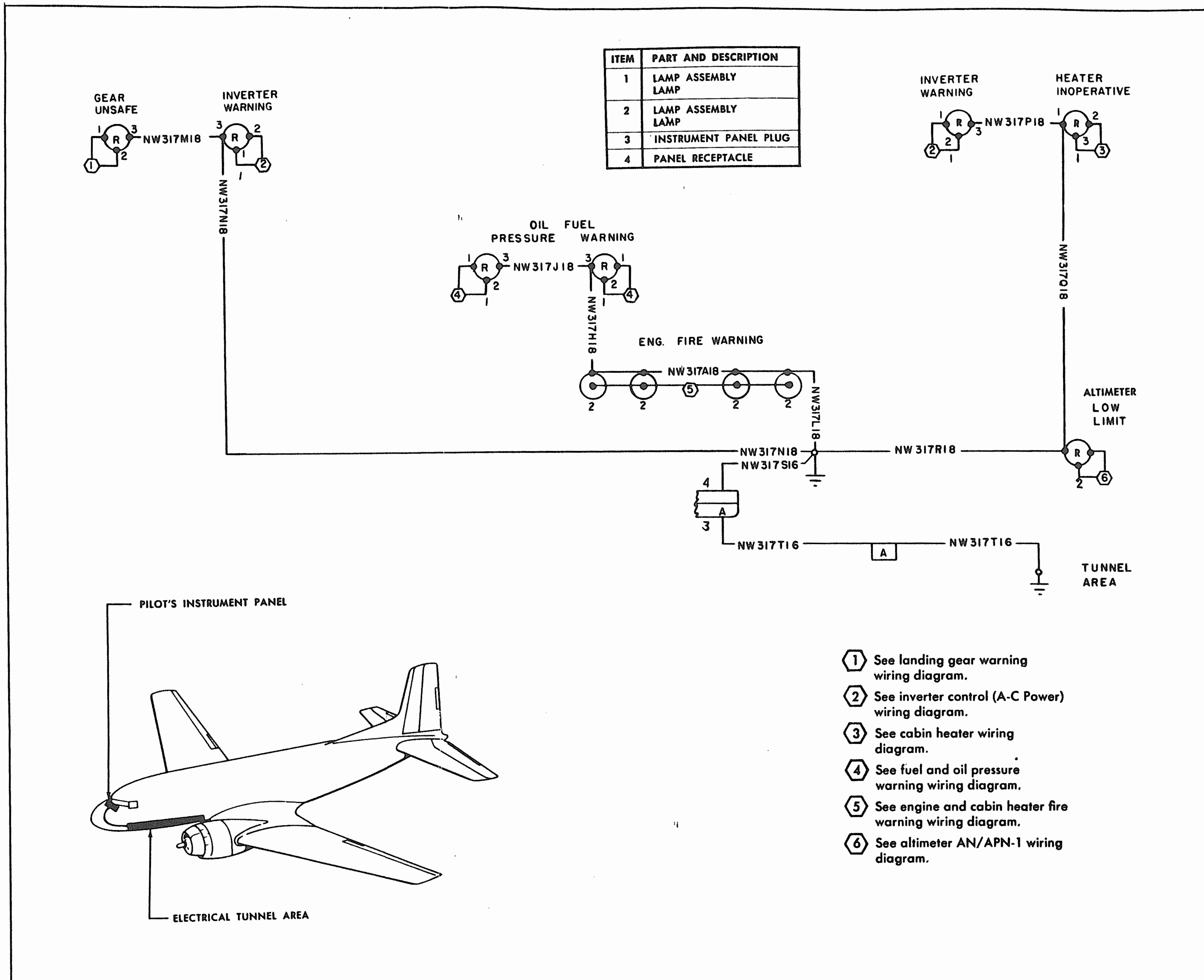
7995

Figure 10-28. Lights — Instrument and Upper Switch Panels



ITEM	PART AND DESCRIPTION
1	EXTEND SWITCH
2	LAMP ON SWITCH
3	WING LIGHT RELAY
4	LANDING LIGHT
5	CIRCUIT BREAKER
6	CIRCUIT BREAKER

Figure 10-29. Lights — Landing



ITEM	PART AND DESCRIPTION
1	LAMP ASSEMBLY LAMP
2	LAMP ASSEMBLY LAMP
3	INSTRUMENT PANEL PLUG
4	PANEL RECEPTACLE

1. See landing gear warning wiring diagram.
2. See inverter control (A-C Power) wiring diagram.
3. See cabin heater wiring diagram.
4. See fuel and oil pressure warning wiring diagram.
5. See engine and cabin heater fire warning wiring diagram.
6. See altimeter AN/APN-1 wiring diagram.

Figure 10-30. Lights — Main Instrument Panel Warning

ITEM	PART AND DESCRIPTION
1	LAMP ASSEMBLY LAMP
2	LAMP ASSEMBLY LAMP
3	LAMP ASSEMBLY LAMP
4	LAMP ASSEMBLY LAMP
5	LAMP ASSEMBLY LAMP
6	LIGHT SWITCH
7	CIRCUIT BREAKER
8	FLASHER PLUG
9	LIGHT FLASHER

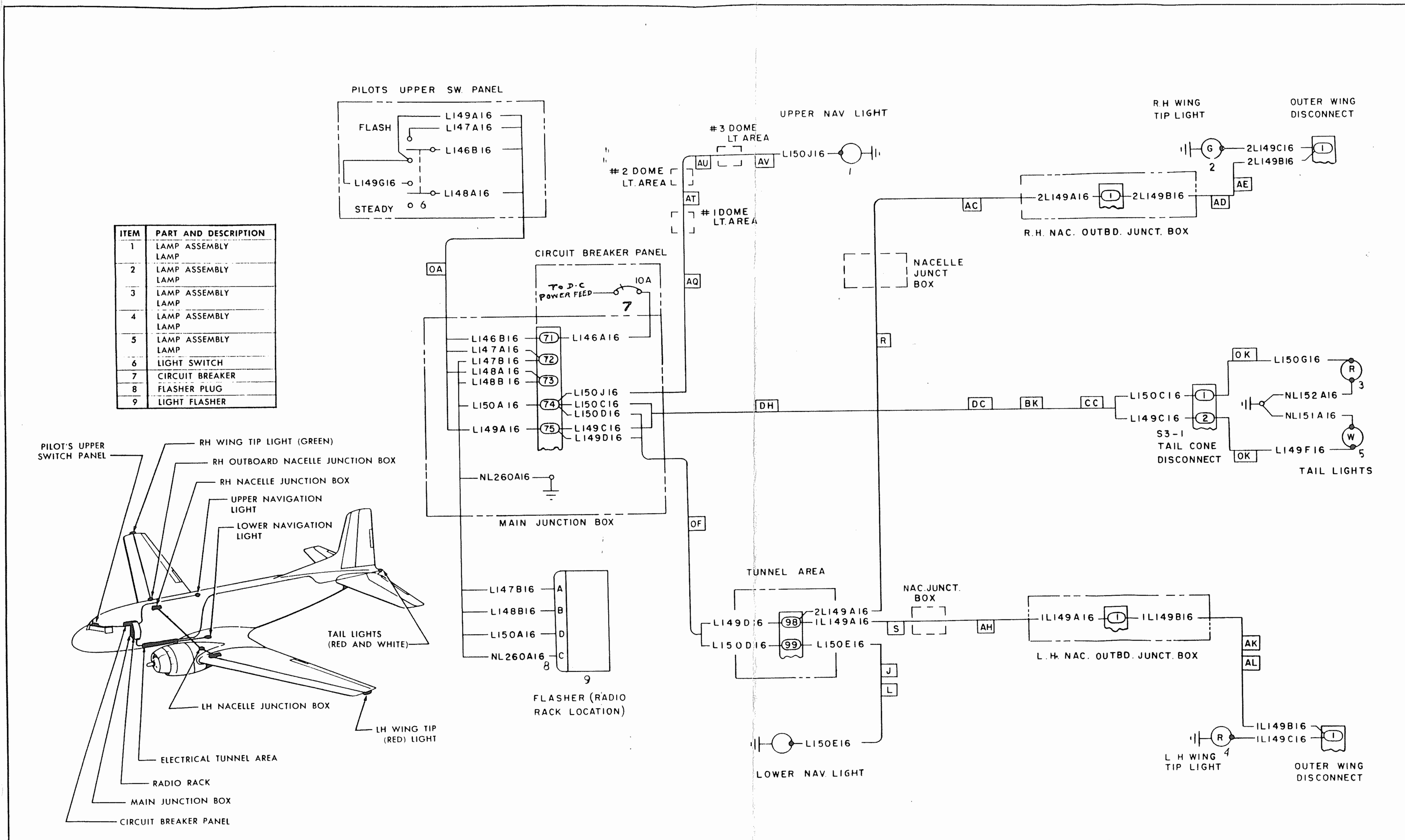
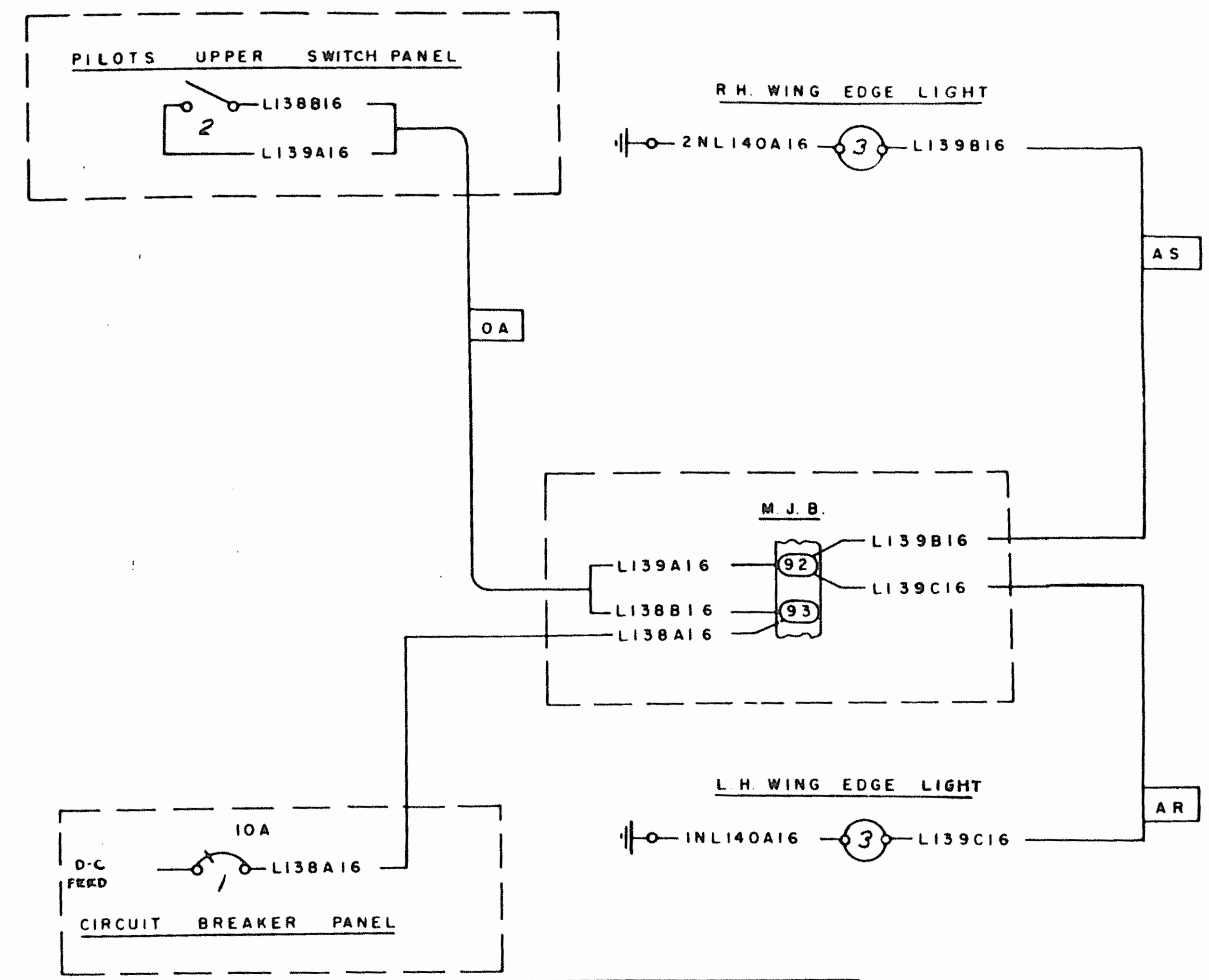
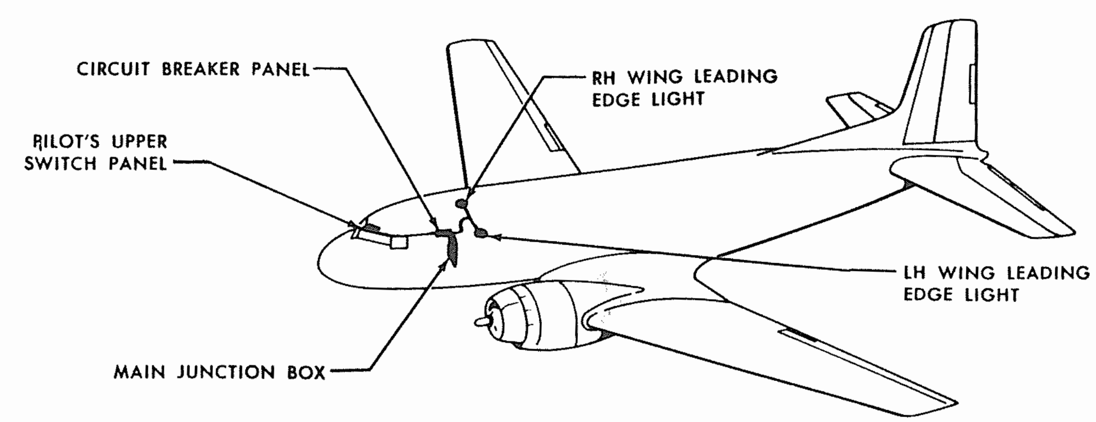


Figure 10-31. Lights — Navigation

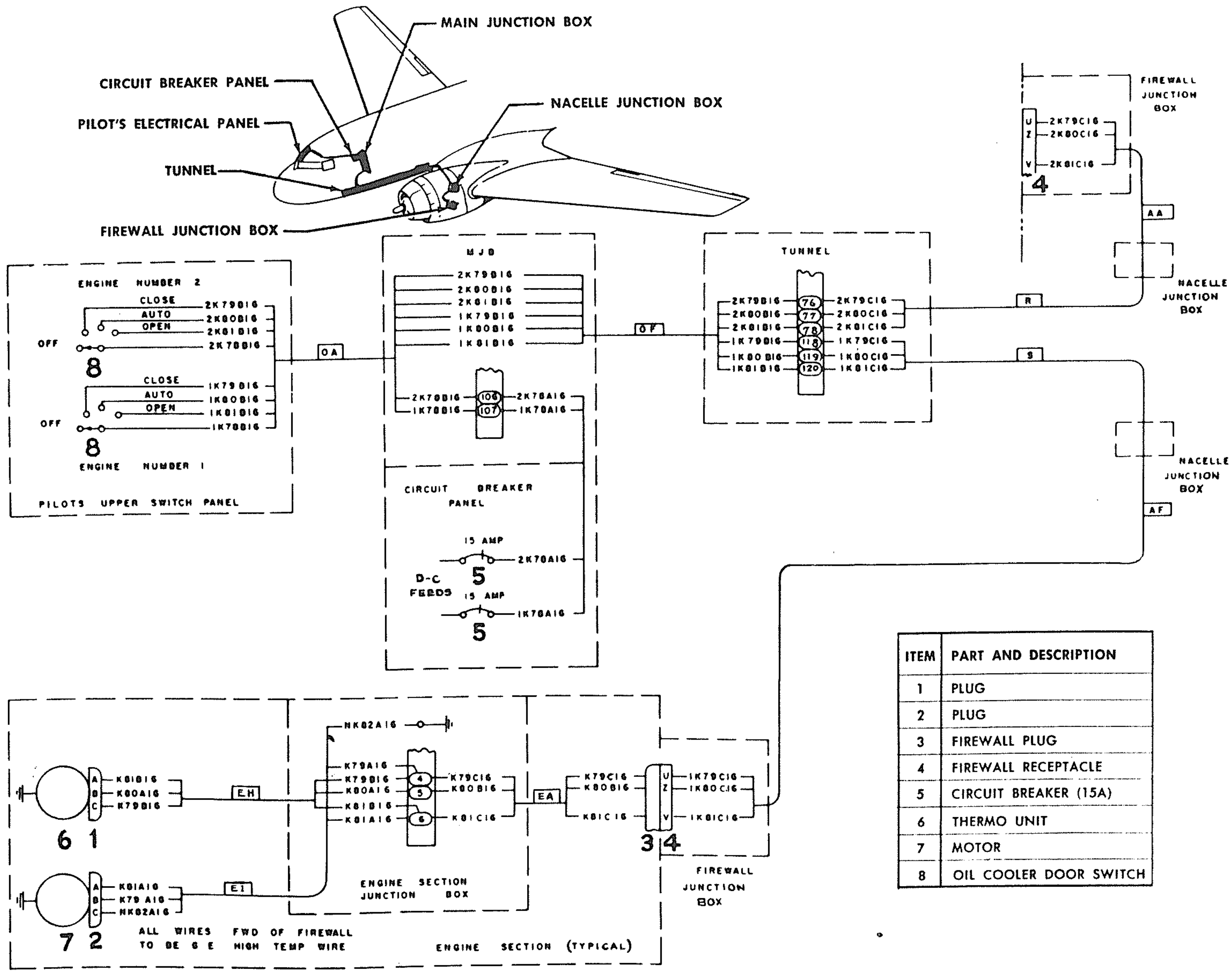


ITEM	PART AND DESCRIPTION
1	CIRCUIT BREAKER
2	LIGHT SWITCH
3	WING LIGHT



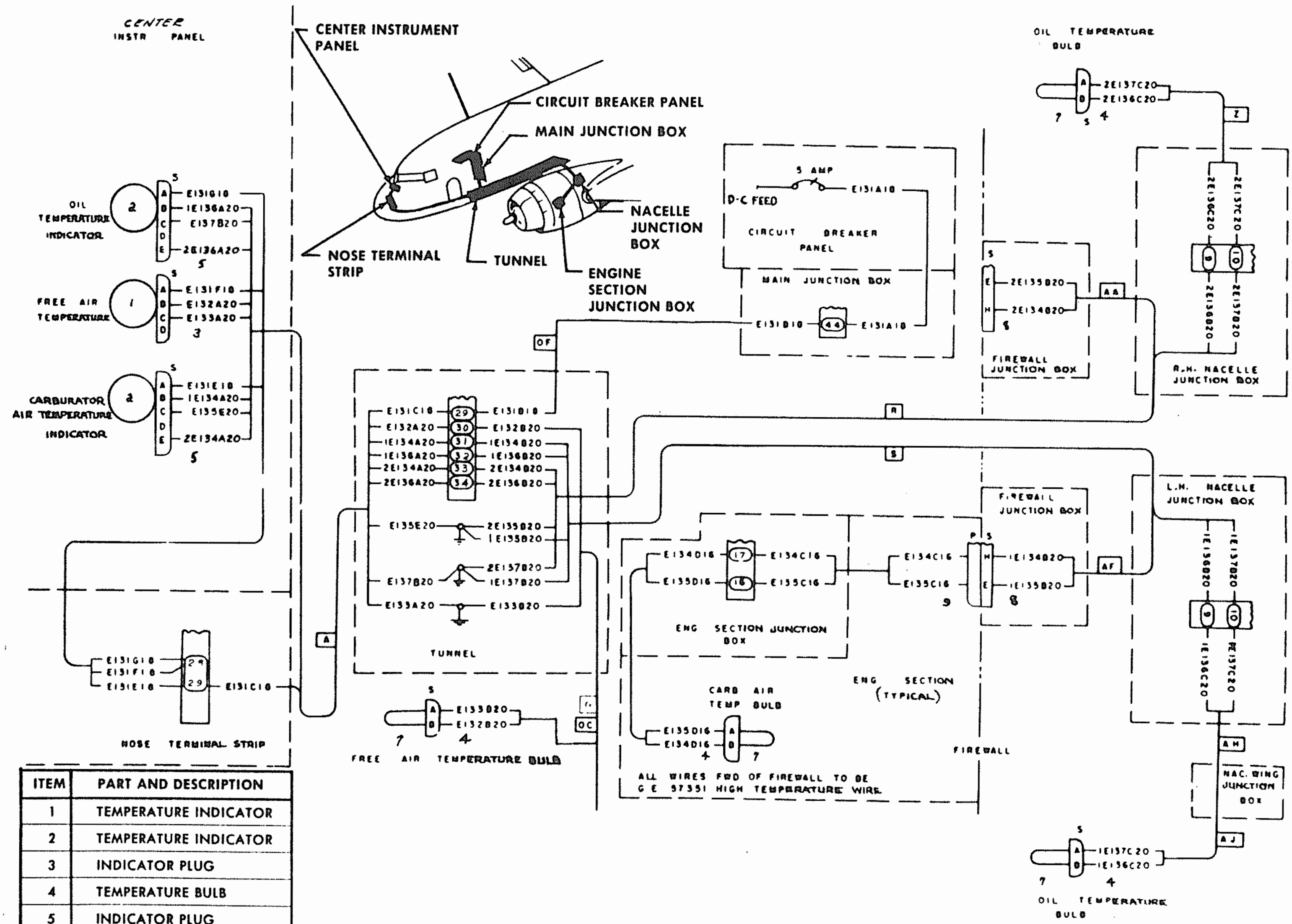
1,538

Figure 10-32. Lights — Wing Leading Edge



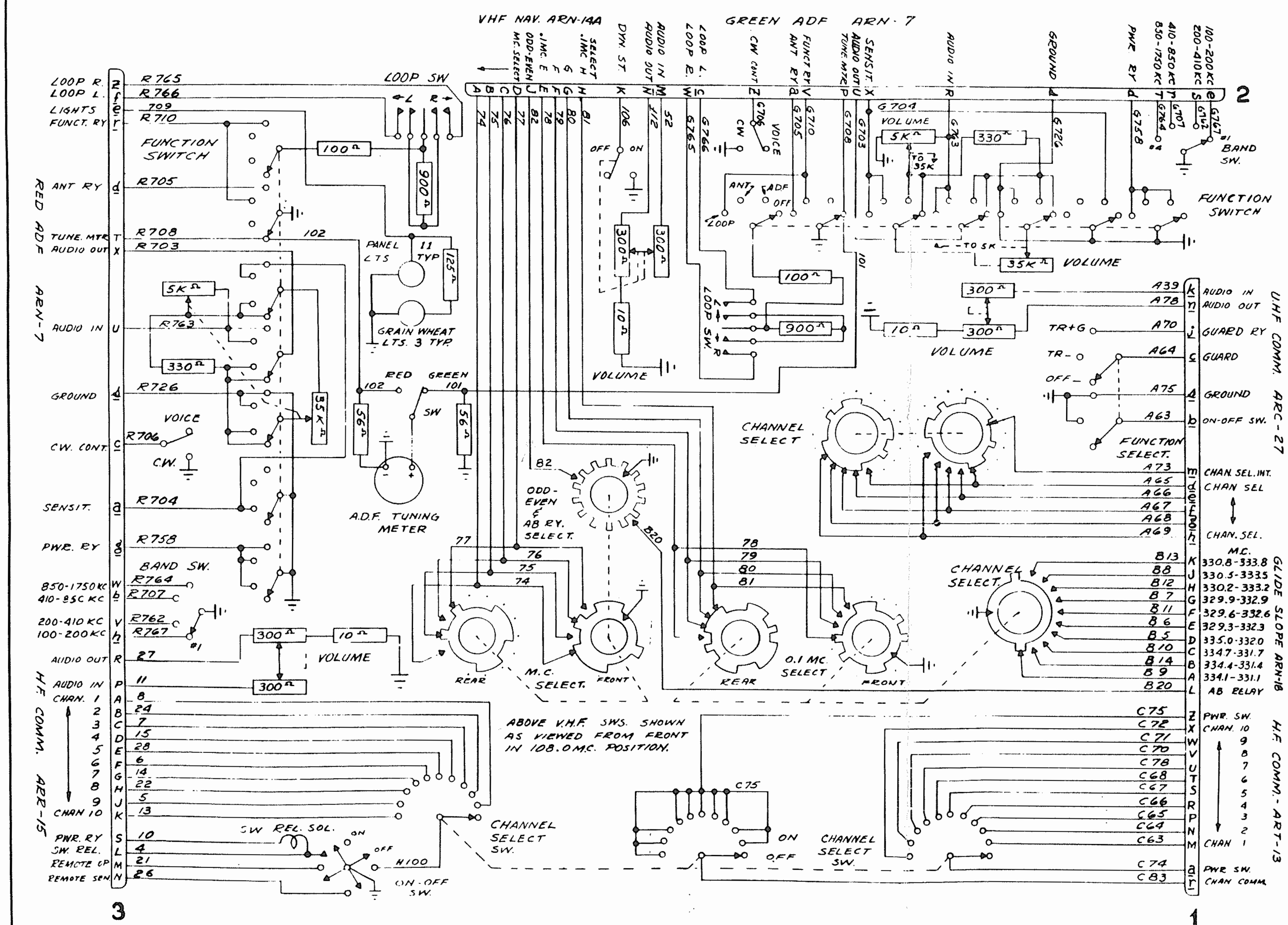
ITEM	PART AND DESCRIPTION
1	PLUG
2	PLUG
3	FIREWALL PLUG
4	FIREWALL RECEPTACLE
5	CIRCUIT BREAKER (15A)
6	THERMO UNIT
7	MOTOR
8	OIL COOLER DOOR SWITCH

Figure 10-33. Oil Cooler Door Control



ITEM	PART AND DESCRIPTION
1	TEMPERATURE INDICATOR
2	TEMPERATURE INDICATOR
3	INDICATOR PLUG
4	TEMPERATURE BULB
5	INDICATOR PLUG
6	CIRCUIT BREAKER (5A)
7	TEMPERATURE BULB
8	FIREWALL RECEPTACLE
9	FIREWALL PLUG

Figure 10-34. Oil and Air Temperature Indicating System



ITEM	PART AND DESCRIPTION
1	PLUG
2	PLUG
3	PLUG

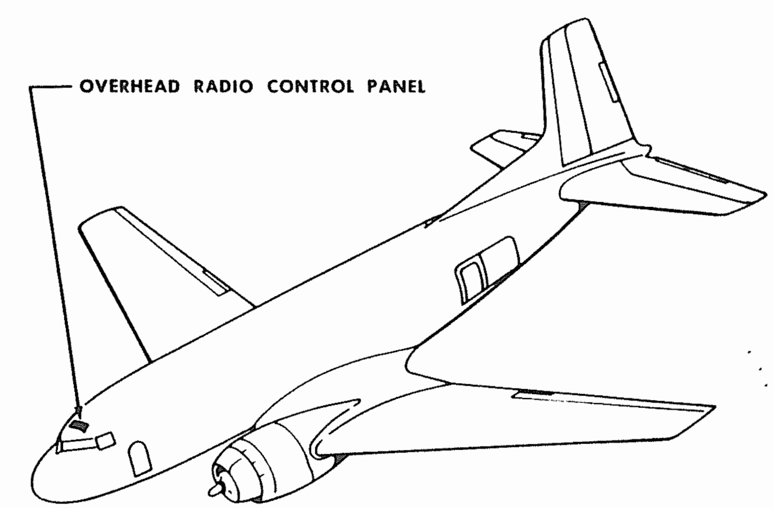
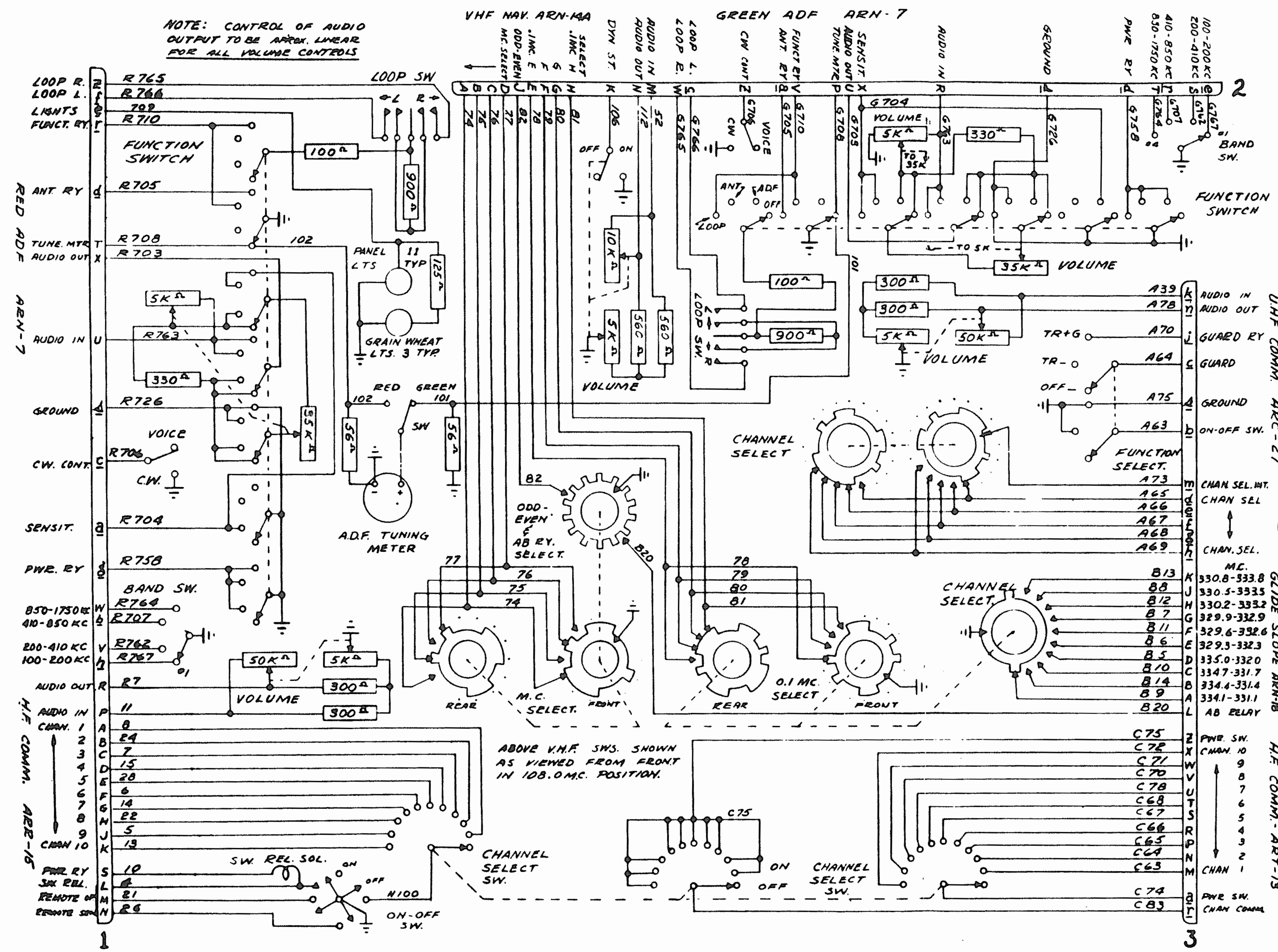


Figure 10-35. Overhead Radio Control Panel (Aircraft A, B, and 1 through 18)

7998



ITEM	PART AND DESCRIPTION
1	PLUG
2	PLUG
3	PLUG

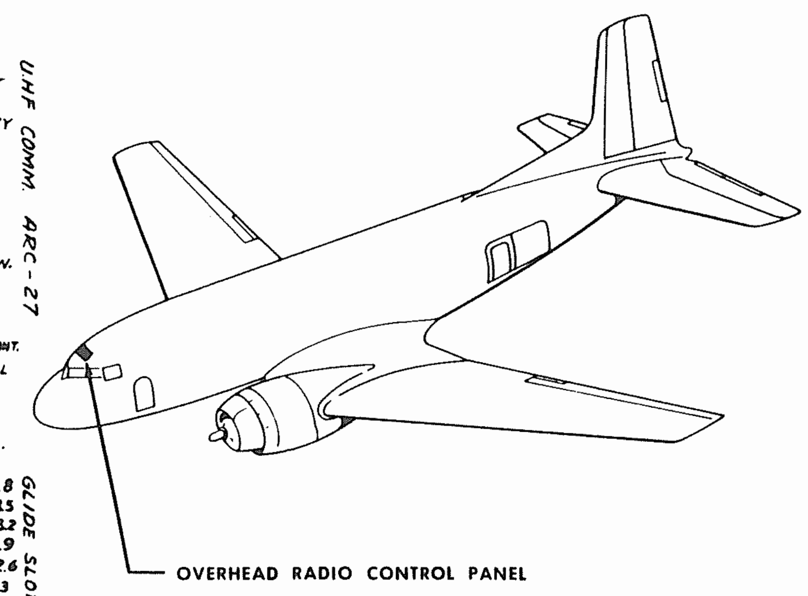
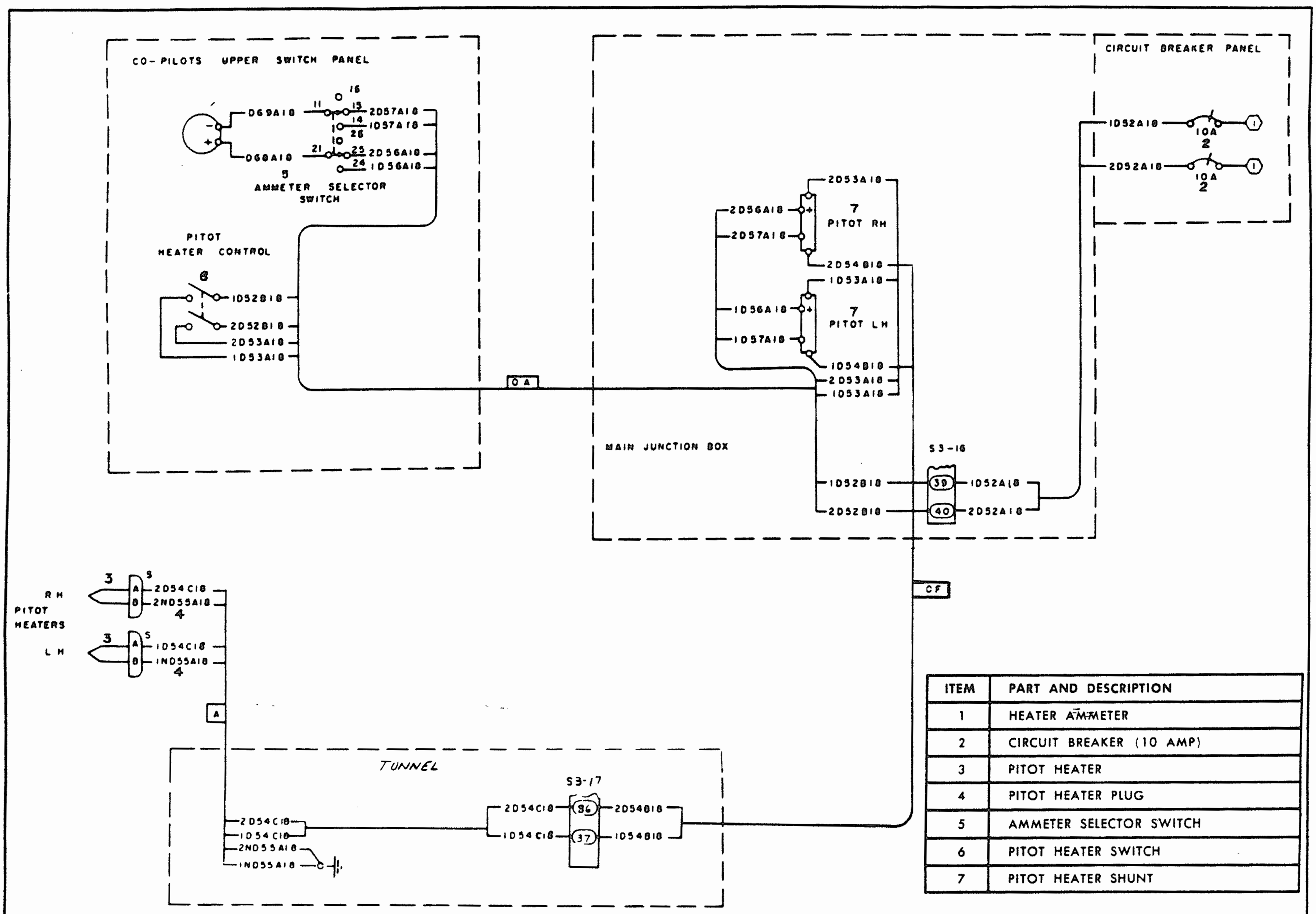


Figure 10-36. Overhead Radio Control Panel (Aircraft C, D, and 19 through 96)

7999



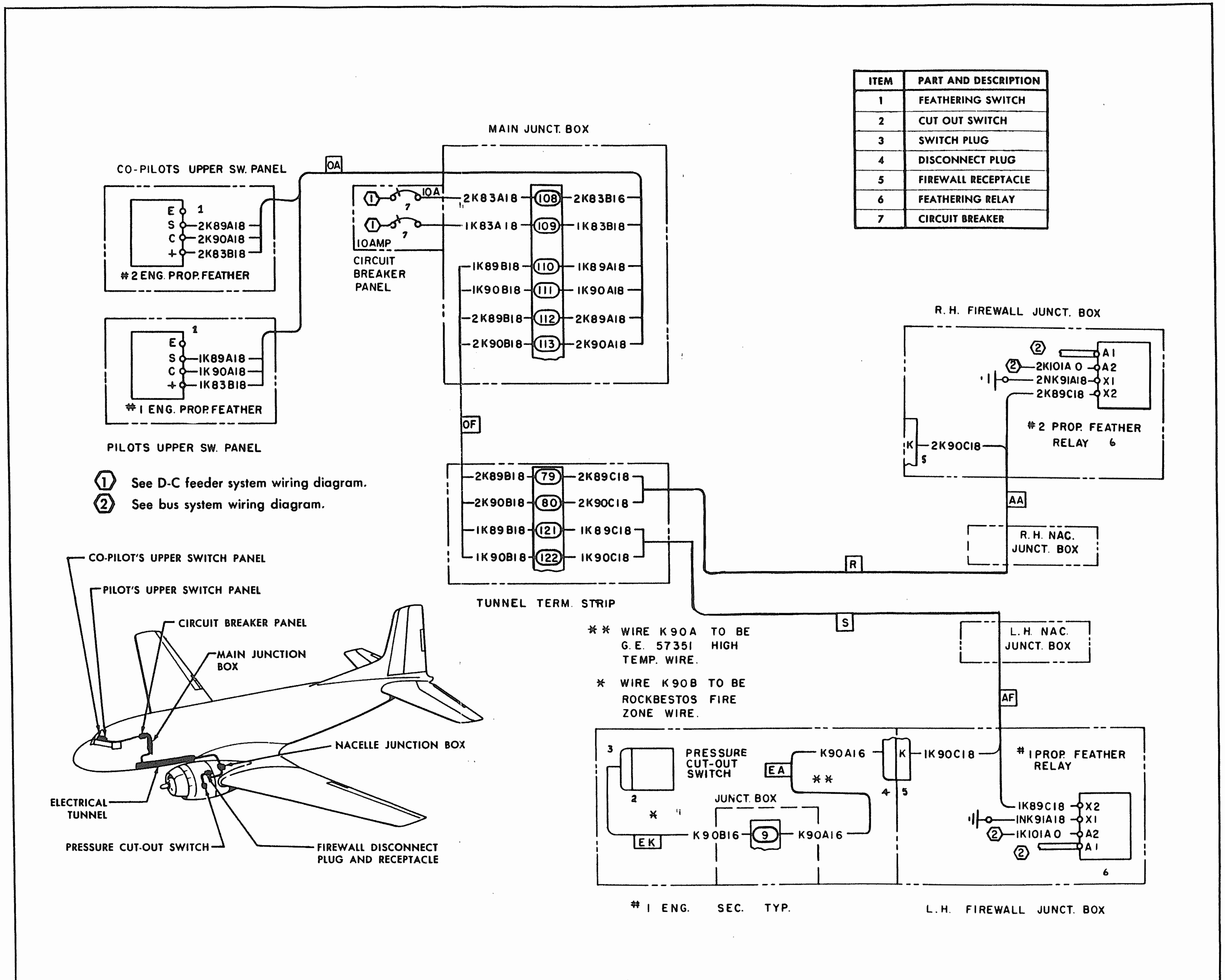
ITEM	PART AND DESCRIPTION
1	HEATER AMMETER
2	CIRCUIT BREAKER (10 AMP)
3	PITOT HEATER
4	PITOT HEATER PLUG
5	AMMETER SELECTOR SWITCH
6	PITOT HEATER SWITCH
7	PITOT HEATER SHUNT

(1) FOR D C FEEDS SEE PAGE 64P2.

11,367

Figure 10-37. Pitot Metering and Control (Aircraft C, D, and 79 through 96)

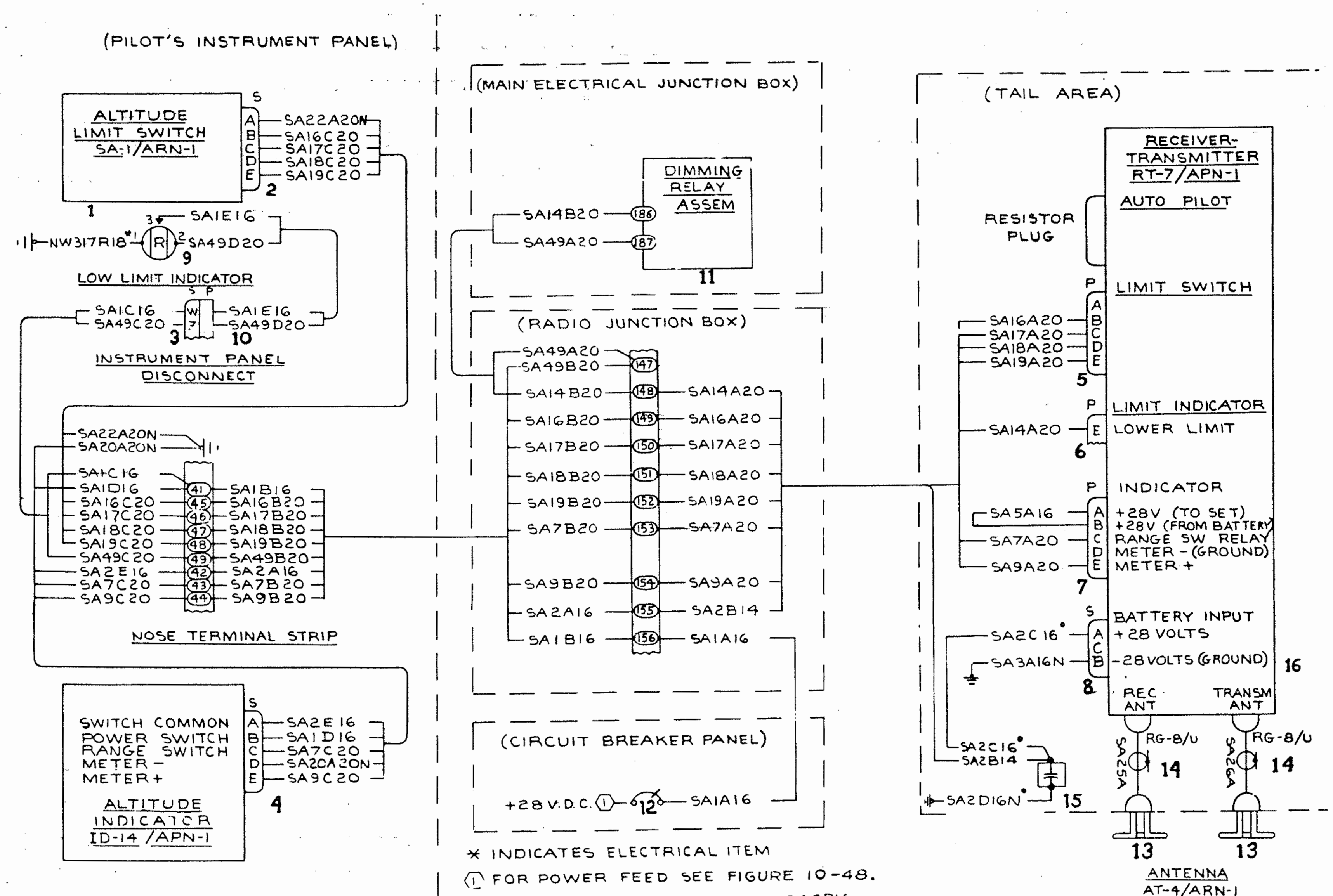
ITEM	PART AND DESCRIPTION
1	FEATHERING SWITCH
2	CUT OUT SWITCH
3	SWITCH PLUG
4	DISCONNECT PLUG
5	FIREWALL RECEPTACLE
6	FEATHERING RELAY
7	CIRCUIT BREAKER



(1) See D-C feeder system wiring diagram.
(2) See bus system wiring diagram.

1,541

Figure 10-38. Propeller Feathering

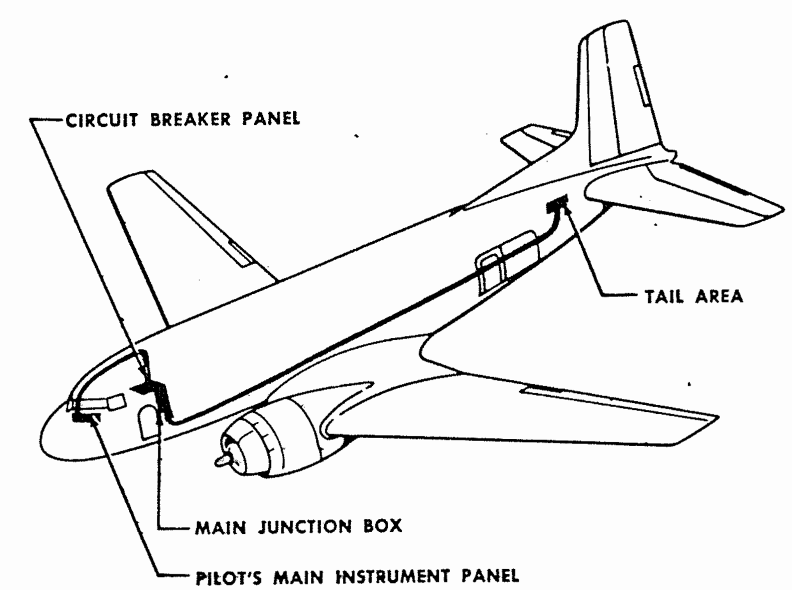


✻ INDICATES ELECTRICAL ITEM

(1) FOR POWER FEED SEE FIGURE 10-48.

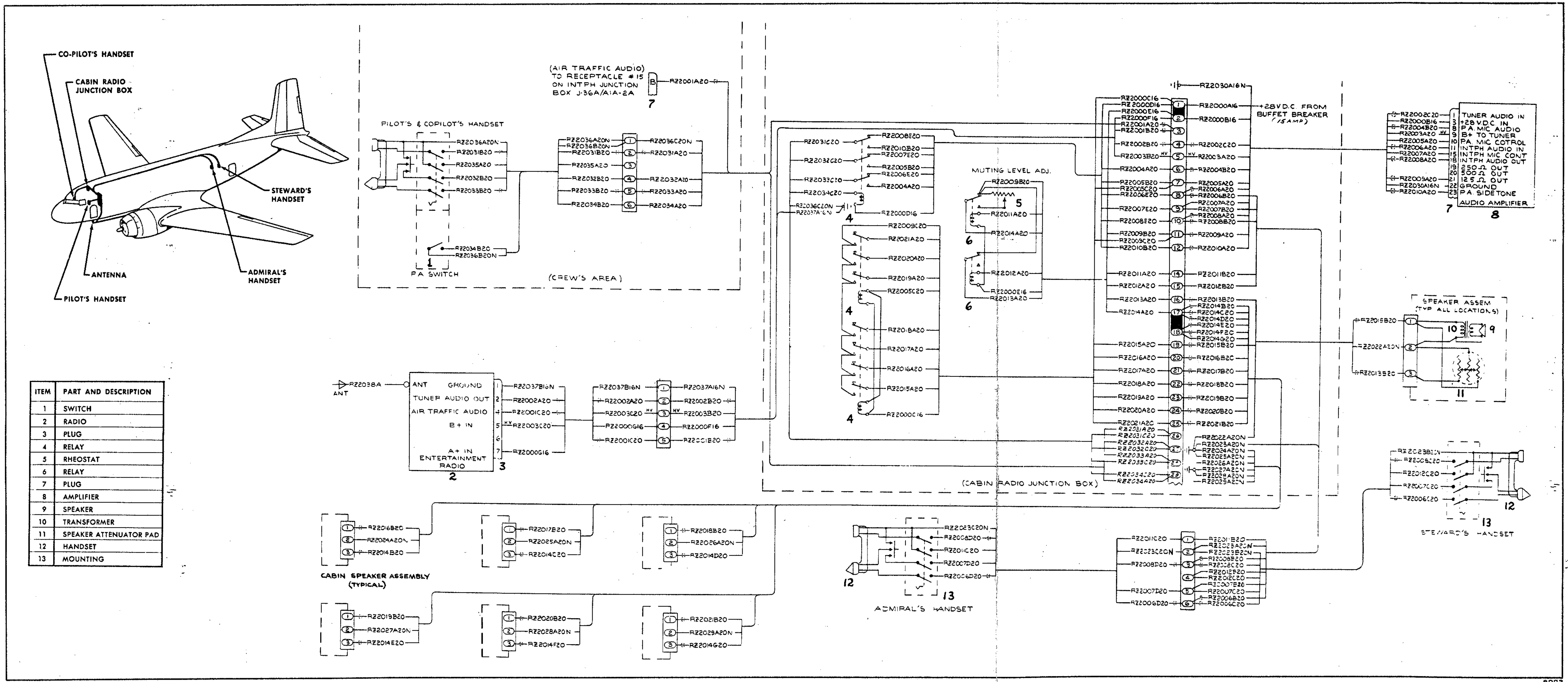
• THE TOTAL LENGTH OF SA2C16 + SA2D16 MUST NOT EXCEED 6 INCHES

ITEM	PART AND DESCRIPTION
1	ALTITUDE LIMIT SWITCH
2	LIMIT SWITCH PLUG
3	INSTRUMENT PANEL PLUG
4	INDICATOR PLUG
5	LIMIT SWITCH PLUG
6	LIMIT INDICATOR PLUG
7	INDICATOR PLUG
8	BATTERY INPUT PLUG
9	ALTIMETER LIMIT LAMP LAMP ASSEMBLY
10	PANEL RECEPTACLE
11	ROTARY RELAY
12	CIRCUIT BREAKER (10A)
13	ALTIMETER ANTENNA
14	ANTENNA CABLE
15	POWER FILTER
16	RECEIVER TRANSMITTER



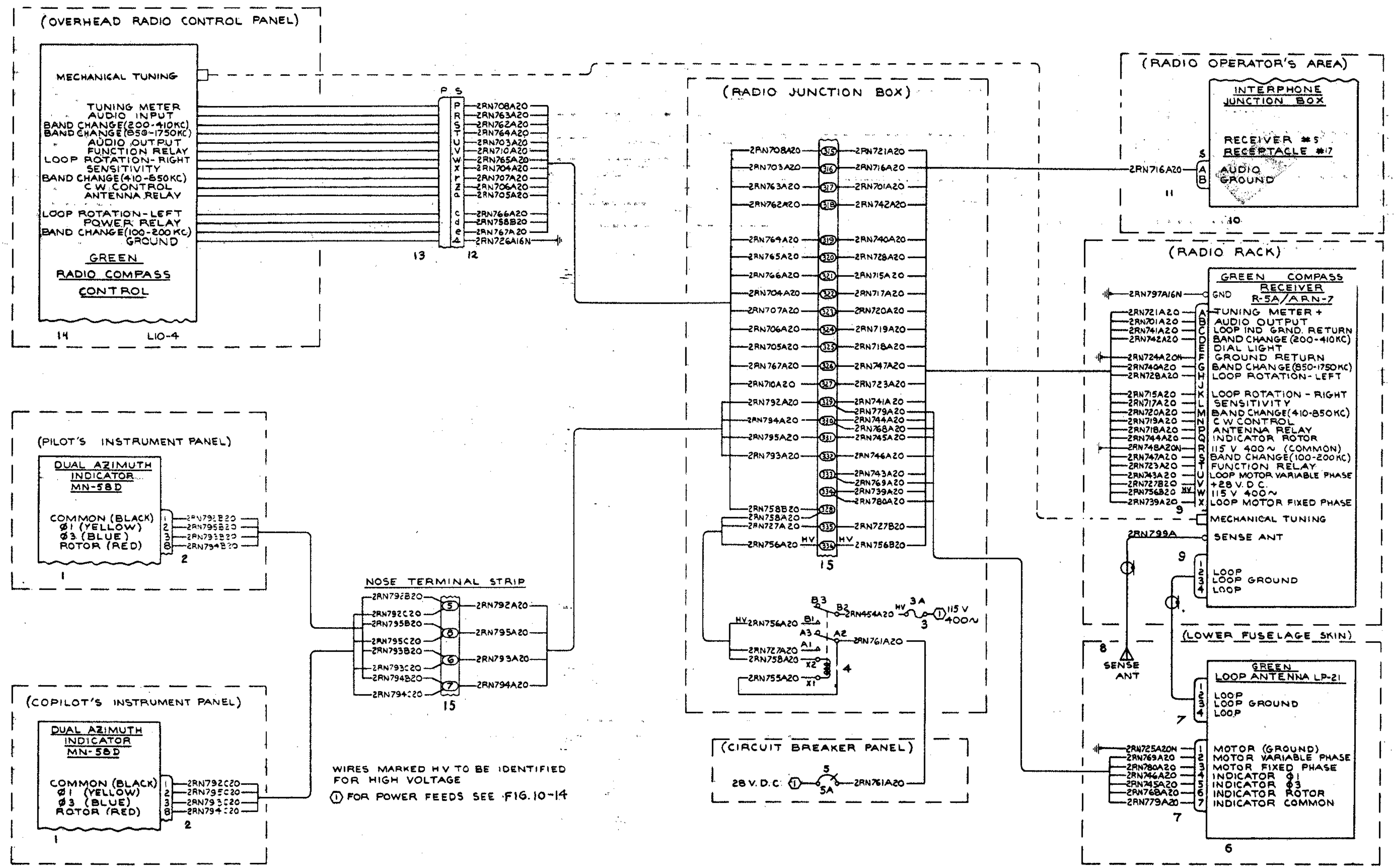
8002

Figure 10-39. Radio Altimeter



ITEM	PART AND DESCRIPTION
1	SWITCH
2	RADIO
3	PLUG
4	RELAY
5	RHEOSTAT
6	RELAY
7	PLUG
8	AMPLIFIER
9	SPEAKER
10	TRANSFORMER
11	SPEAKER ATTENUATOR PAD
12	HANDSET
13	MOUNTING

Figure 10-40. Radio, Cabin Interphone, and Public Address System (Aircraft A through D)

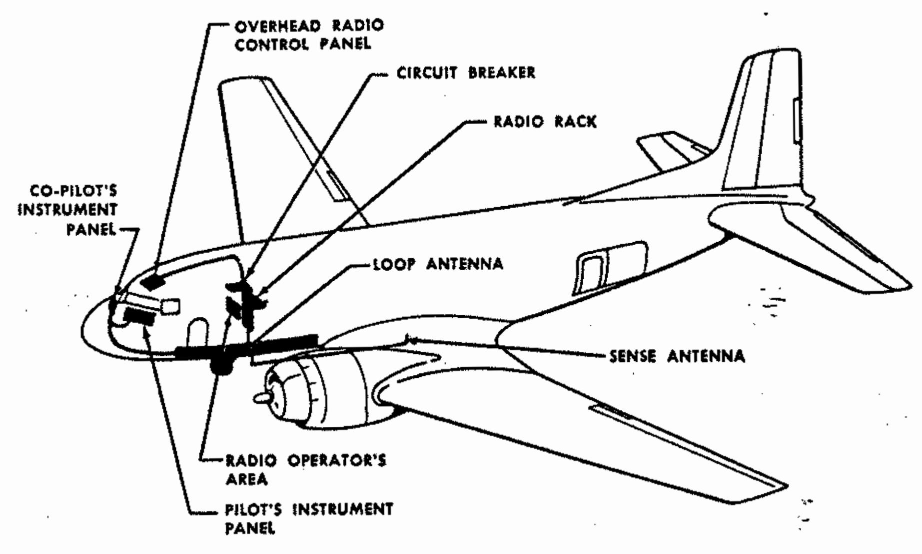


ITEM	PART AND DESCRIPTION
1	INDICATOR
2	PLUG
3	FUSE (3 AMP)
4	RELAY
5	CIRCUIT BREAKER
6	GREEN ADF LOOP ANTENNA
7	PLUG
8	GREEN ADF SENSE ANTENNA
9	PLUG
10	ICS JUNCTION BOX UNIT
11	PLUG
12	RECEPTACLE
13	PLUG
14	CONTROL PANEL
15	TERMINAL STRIP

11,388

Figure 10-41. Radio Compass (Green) (Aircraft A, B, and 1 through 68)

ITEM	PART AND DESCRIPTION
1	OVERHEAD CONT PANEL
2	INDICATOR
3	LAMP
4	PLUG
5	RECEPTACLE
6	PLUG
7	PLUG
8	RECEPTACLE
9	RELAY
10	CIRCUIT BREAKER
11	FUSE
12	PLUG
13	ICS JUNCTION BOX
14	PLUG
15	CABLE
16	GREED ADF REC
17	ANTENNA
18	CABLE
19	PLUG
20	ANTENNA
21	SWITCH



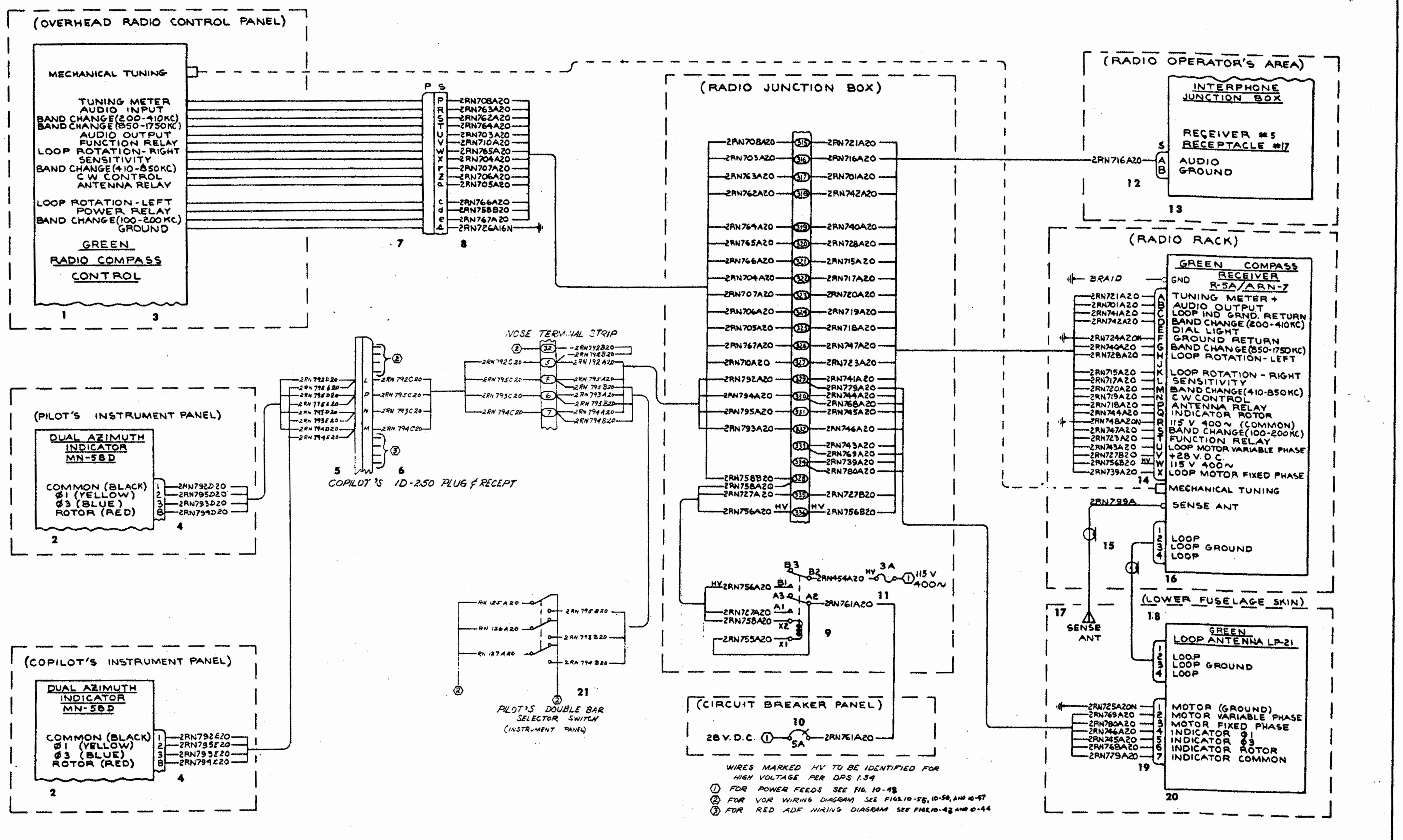
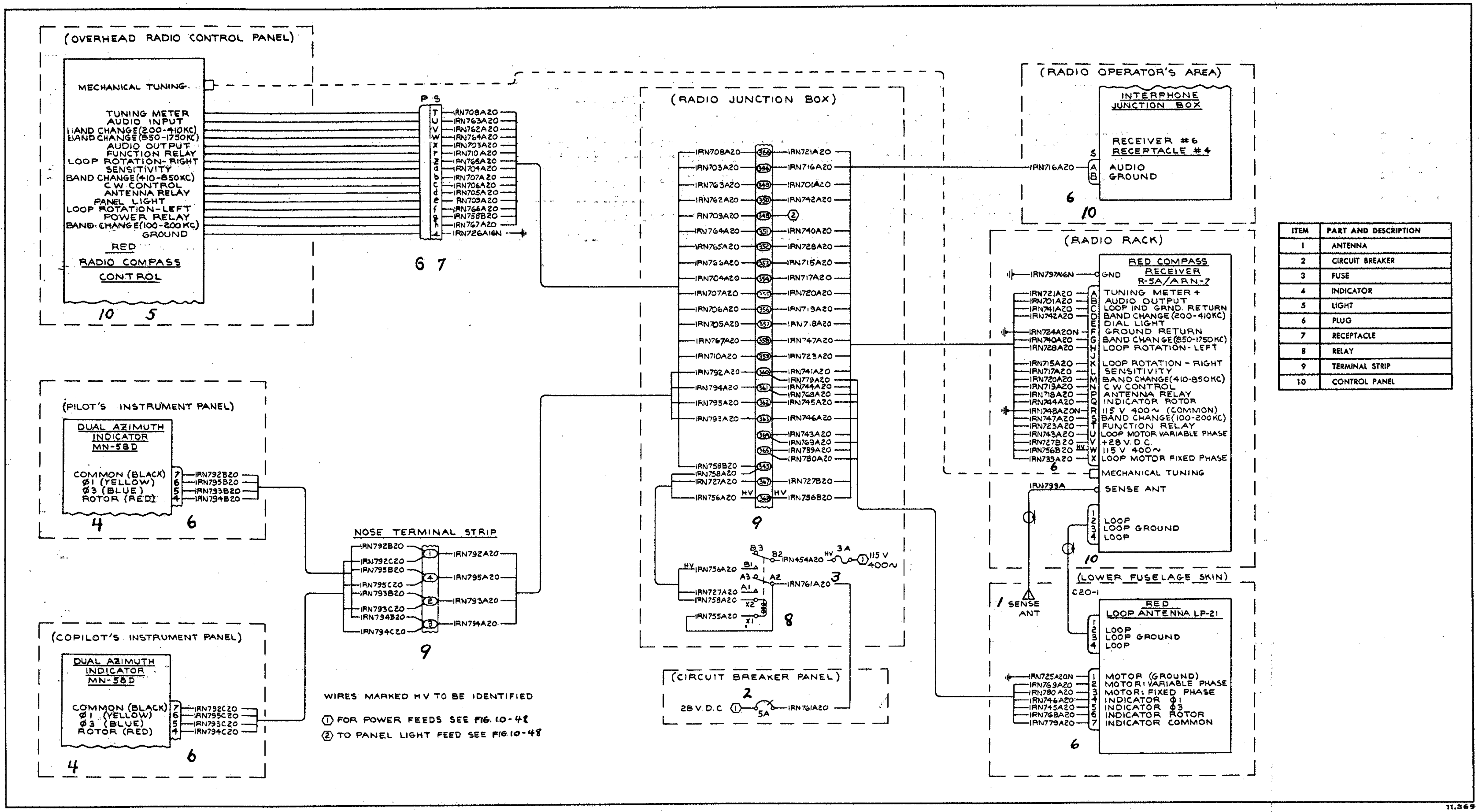


Figure 10-42. Radio Compass (Green) (Aircraft C, D, and 69 through 96)



ITEM	PART AND DESCRIPTION
1	ANTENNA
2	CIRCUIT BREAKER
3	FUSE
4	INDICATOR
5	LIGHT
6	PLUG
7	RECEPTACLE
8	RELAY
9	TERMINAL STRIP
10	CONTROL PANEL

Figure 10-43. Radio Compass (Red) (Aircraft A, B, and 1 through 68)

ITEM	PART AND DESCRIPTION
1	OVERHEAD CONT PANEL
2	INDICATOR
3	LAMP
4	PLUG
5	RECEPTACLE
6	PLUG
7	PLUG
8	RECEPTACLE
9	RELAY
10	CIRCUIT BREAKER
11	FUSE
12	PLUG
13	ICS JUNCTION BOX
14	PLUG
15	CABLE
16	RED ADF REC
17	ANTENNA
18	CABLE
19	PLUG
20	ANTENNA
21	SWITCH

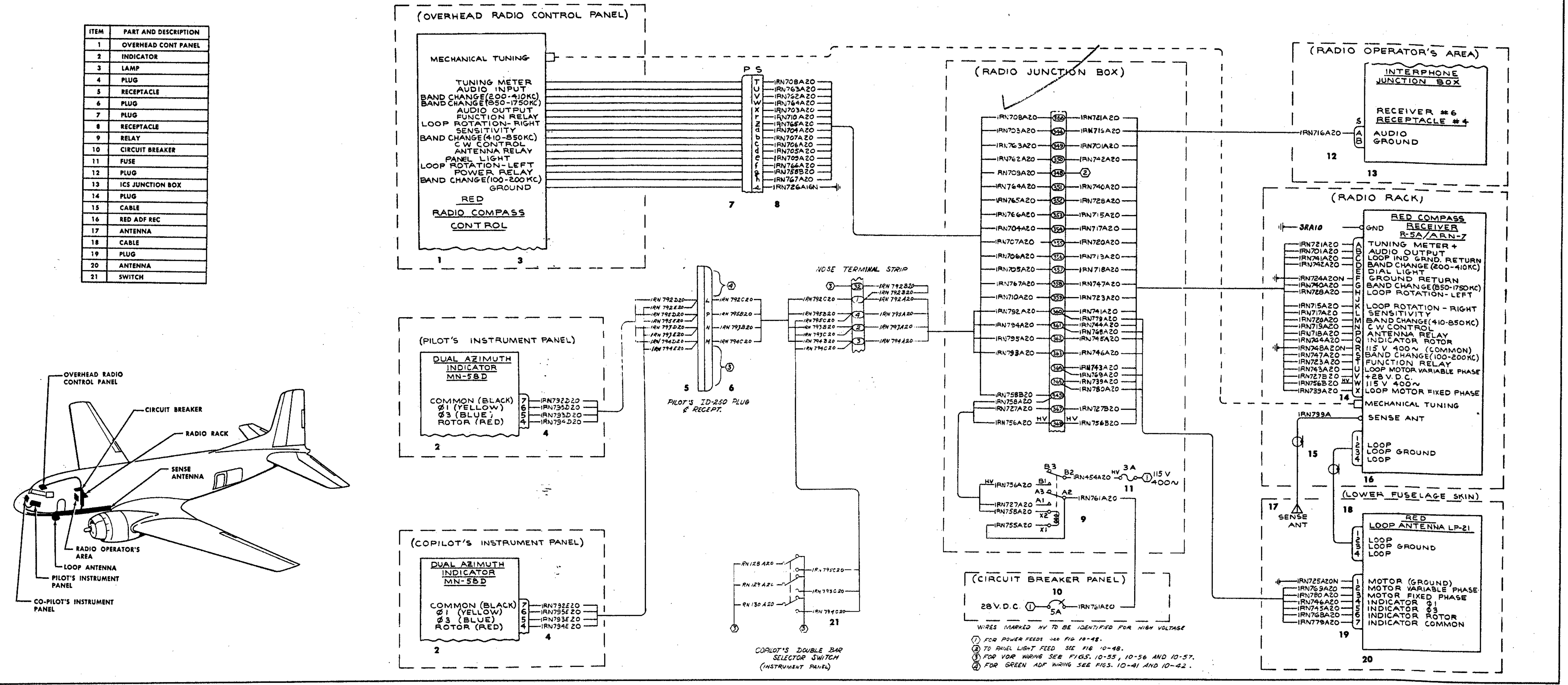
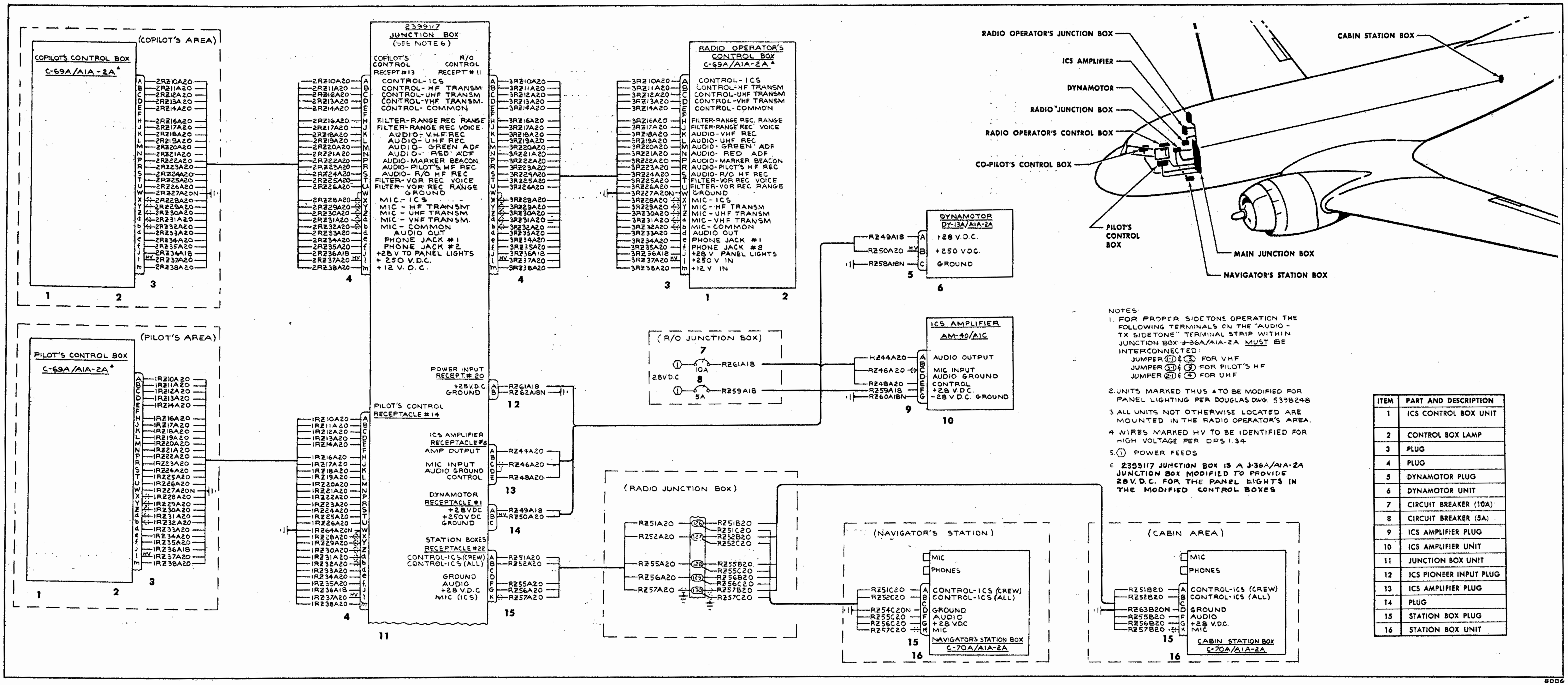


Figure 10-44. Radio Compass (Red) (Aircraft C, D, and 69 through 96)



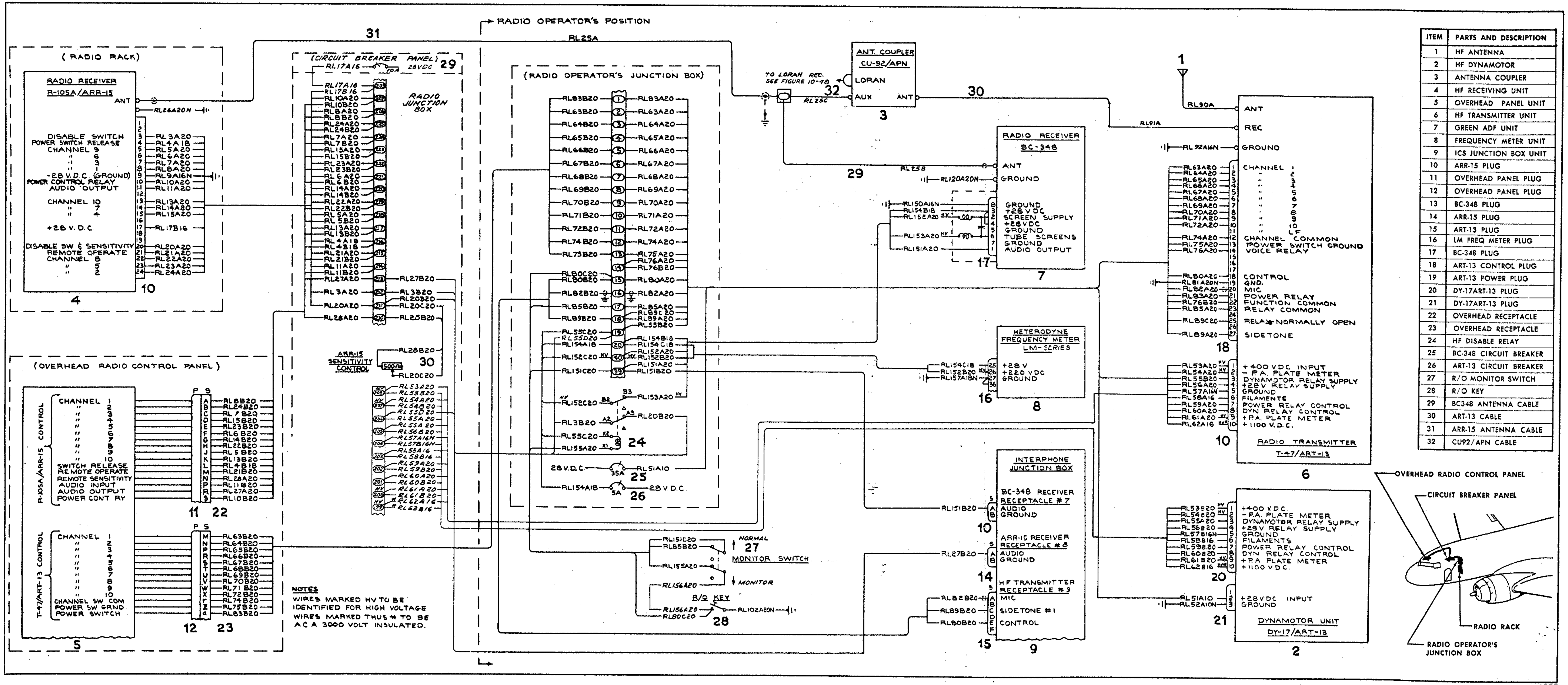
NOTES:

1. FOR PROPER SIDETONE OPERATION THE FOLLOWING TERMINALS ON THE "AUDIO - TX SIDETONE" TERMINAL STRIP WITHIN JUNCTION BOX J-36A/AIA-2A MUST BE INTERCONNECTED:
   - JUMPER (1-1) & (3) FOR VHF
   - JUMPER (3-1) & (9) FOR PILOT'S HF
   - JUMPER (2-1) & (4) FOR UHF
2. UNITS MARKED THUS ▲ TO BE MODIFIED FOR PANEL LIGHTING PER DOUGLAS DWG. 5398248
3. ALL UNITS NOT OTHERWISE LOCATED ARE MOUNTED IN THE RADIO OPERATOR'S AREA.
4. WIRES MARKED HV TO BE IDENTIFIED FOR HIGH VOLTAGE PER DPS 1.34
5. (1) POWER FEEDS
6. 2399117 JUNCTION BOX IS A J-36A/AIA-2A JUNCTION BOX MODIFIED TO PROVIDE 28V. D.C. FOR THE PANEL LIGHTS IN THE MODIFIED CONTROL BOXES

ITEM	PART AND DESCRIPTION
1	ICS CONTROL BOX UNIT
2	CONTROL BOX LAMP
3	PLUG
4	PLUG
5	DYNAMOTOR PLUG
6	DYNAMOTOR UNIT
7	CIRCUIT BREAKER (10A)
8	CIRCUIT BREAKER (5A)
9	ICS AMPLIFIER PLUG
10	ICS AMPLIFIER UNIT
11	JUNCTION BOX UNIT
12	ICS PIONEER INPUT PLUG
13	ICS AMPLIFIER PLUG
14	PLUG
15	STATION BOX PLUG
16	STATION BOX UNIT

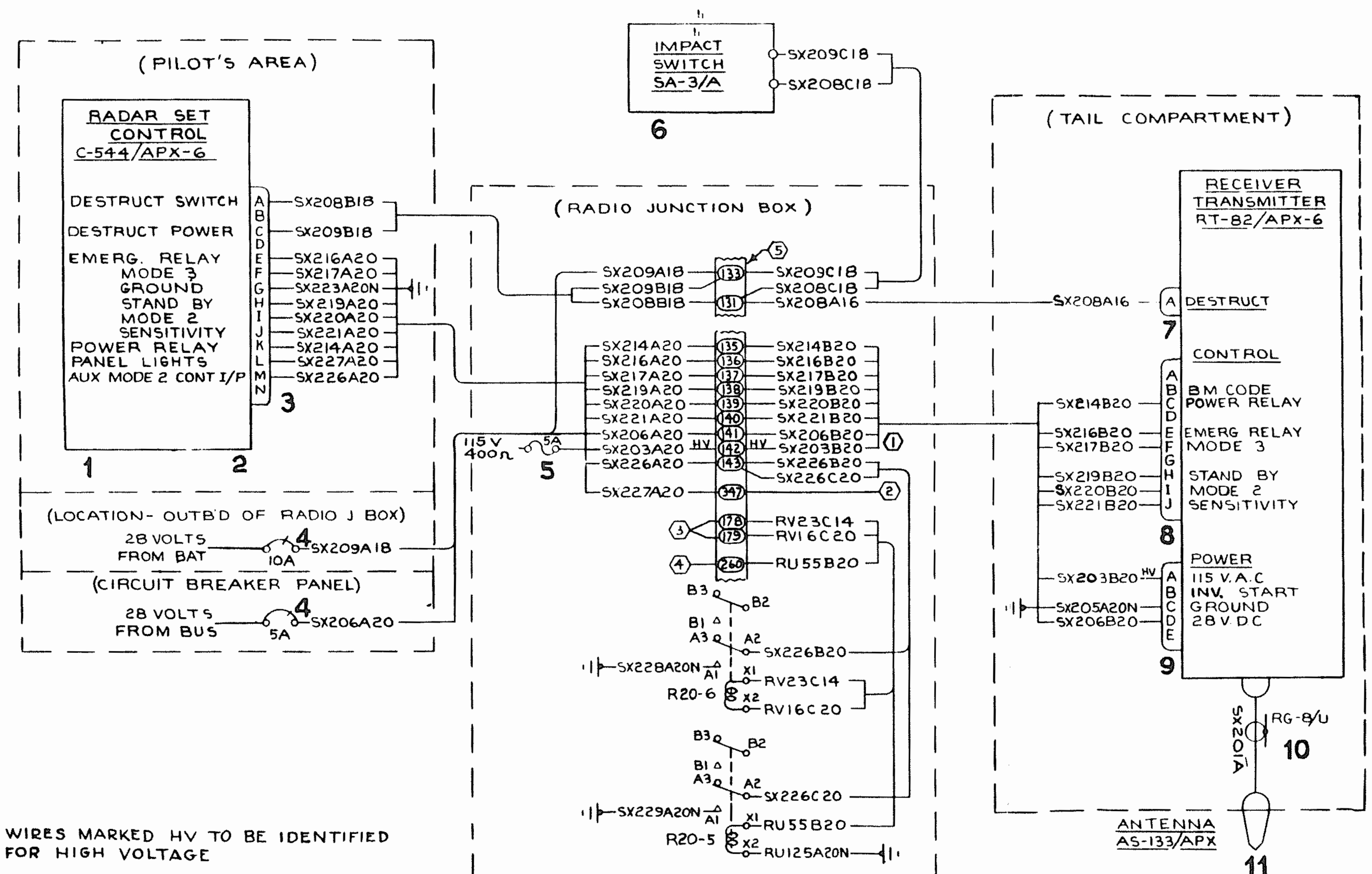
8006

Figure 10-45. Radio Control and Interphone System



ITEM	PARTS AND DESCRIPTION
1	HF ANTENNA
2	HF DYNAMOTOR
3	ANTENNA COUPLER
4	HF RECEIVING UNIT
5	OVERHEAD PANEL UNIT
6	HF TRANSMITTER UNIT
7	GREEN ADF UNIT
8	FREQUENCY METER UNIT
9	ICS JUNCTION BOX UNIT
10	ARR-15 PLUG
11	OVERHEAD PANEL PLUG
12	OVERHEAD PANEL PLUG
13	BC-348 PLUG
14	ARR-15 PLUG
15	ART-13 PLUG
16	LM FREQ METER PLUG
17	BC-348 PLUG
18	ART-13 CONTROL PLUG
19	ART-13 POWER PLUG
20	DY-17ART-13 PLUG
21	DY-17ART-13 PLUG
22	OVERHEAD RECEPTACLE
23	OVERHEAD RECEPTACLE
24	HF DISABLE RELAY
25	BC-348 CIRCUIT BREAKER
26	ART-13 CIRCUIT BREAKER
27	R/O MONITOR SWITCH
28	R/O KEY
29	BC348 ANTENNA CABLE
30	ART-13 CABLE
31	ARR-15 ANTENNA CABLE
32	CU92/APN CABLE

Figure 10-46. Radio HF Communication Equipment



① WIRES MARKED HV TO BE IDENTIFIED FOR HIGH VOLTAGE

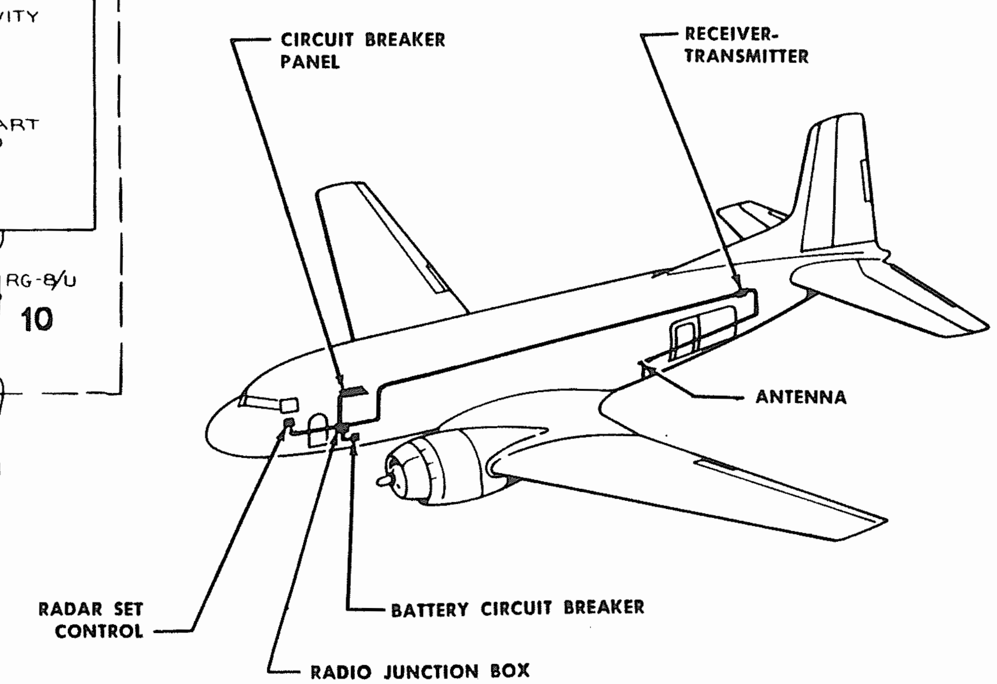
② TO PANEL LIGHT SUPPLY.

③ THIS CIRCUIT ENERGIZED BY VFH TRANSMITTER

④ THIS CIRCUIT ENERGIZED BY UFH TRANSMITTER

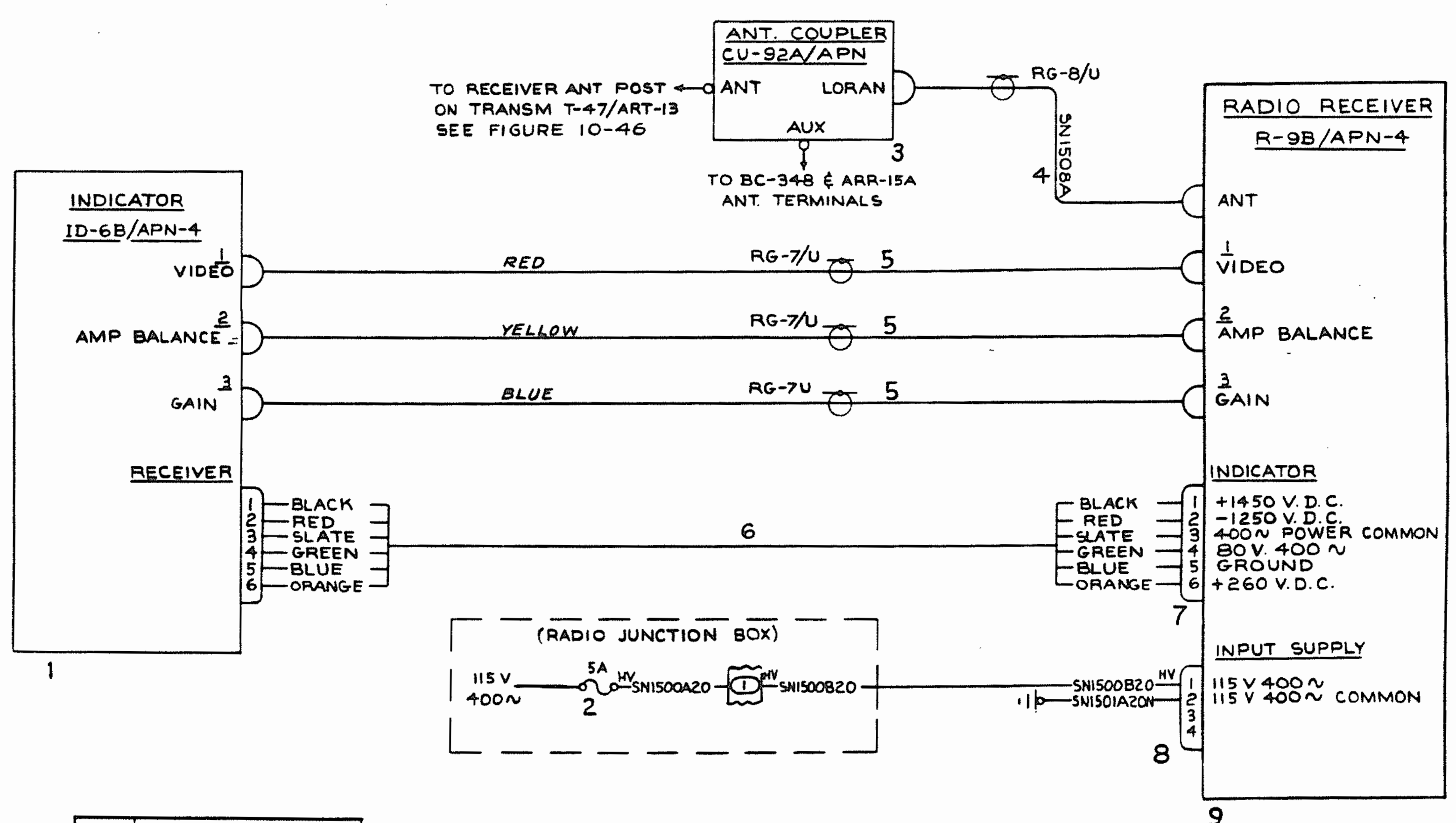
⑤ TERMINALS NUMBERED 131 & 133 TO BE MARKED "IFF" DESTRUCTOR ONLY".

ITEM	PART AND DESCRIPTION
1	IFF CONTROL UNIT
2	IFF CONTROL BOX LAMP
3	IFF CONTROL BOX PLUG
4	CIRCUIT BREAKER
5	IFF FUSE
6	IMPACT SWITCH
7	APX-6 (DESTRUCTOR) PLUG
8	APX-6 (CONTROL) PLUG
9	APX-6 (POWER) PLUG
10	IFF ANTENNA CABLE
11	IFF ANTENNA



8000

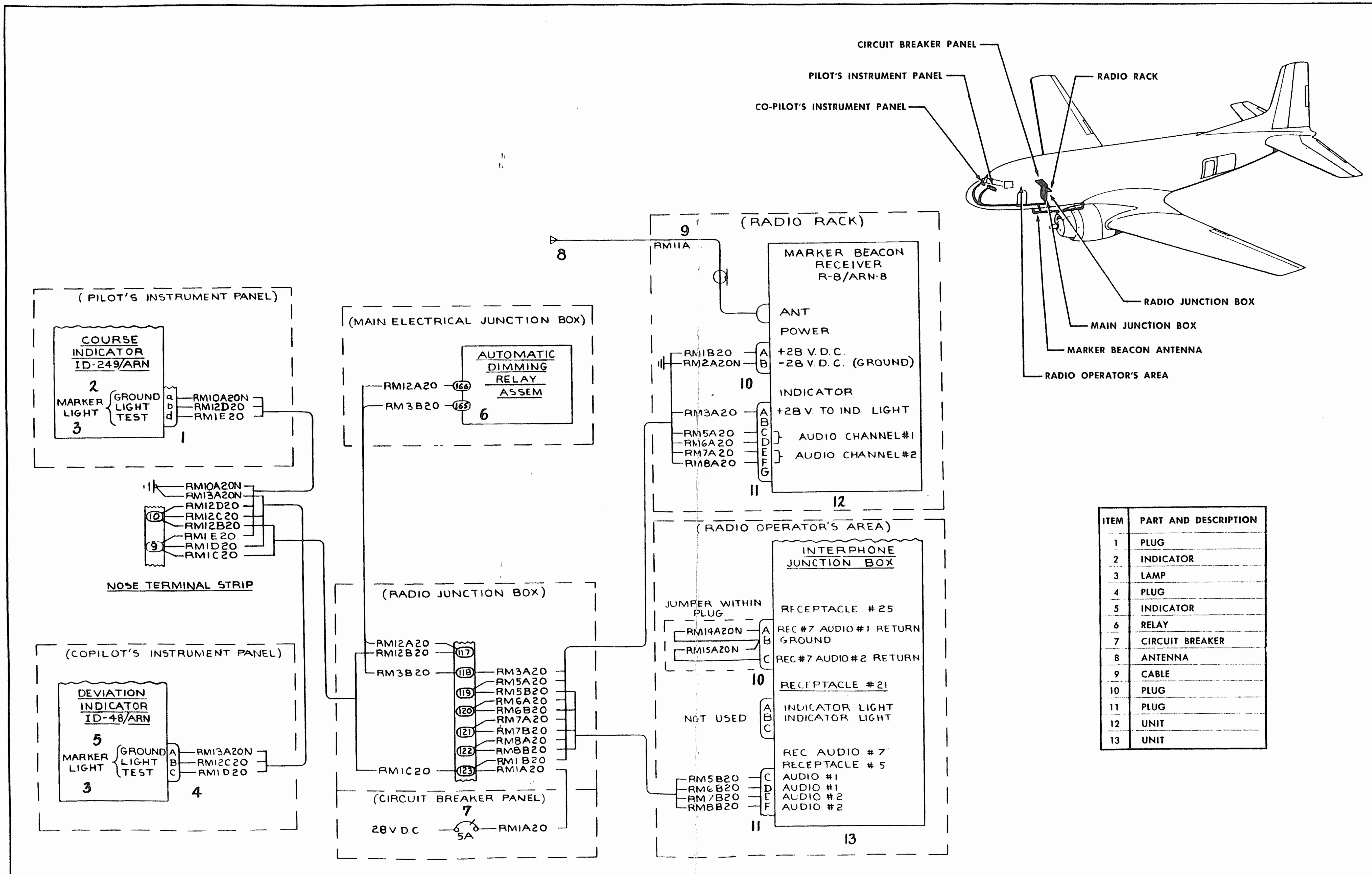
Figure 10-47. Radio IFF Equipment



ITEM	PART AND DESCRIPTION
1	LORAN INDICATOR
2	FUSE (5 AMP)
3	UNIT
4	CABLE
5	CABLE
6	CABLE
7	PLUG
8	PLUG
9	LORAN RECEIVER UNIT

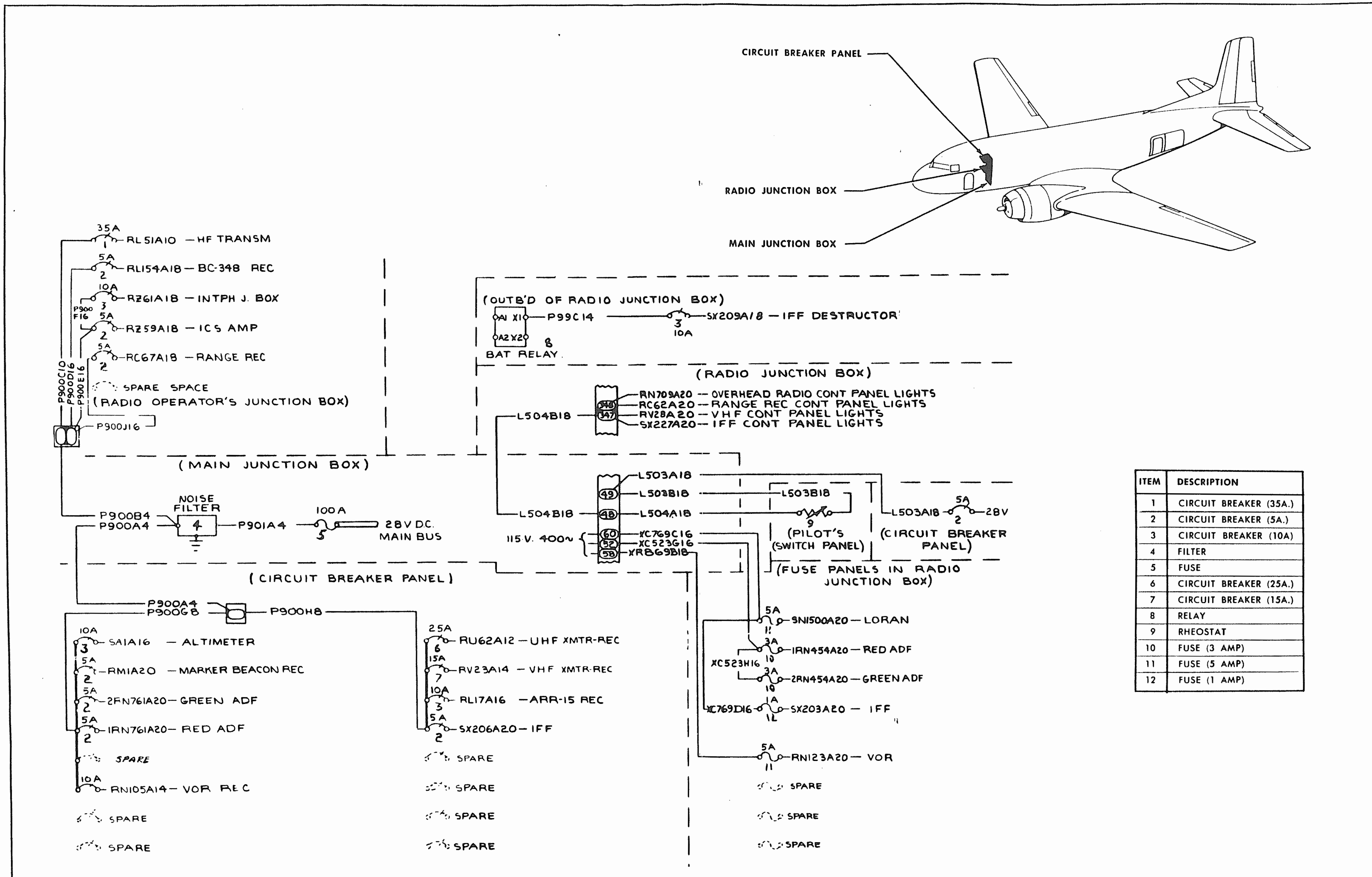
WIRES MARKED "HV" CONDUCT VOLTAGES IN EXCESS OF 70V.

UNLESS OTHERWISE NOTED, UNITS ARE LOCATED AT RADIO OPERATOR'S POSITION.



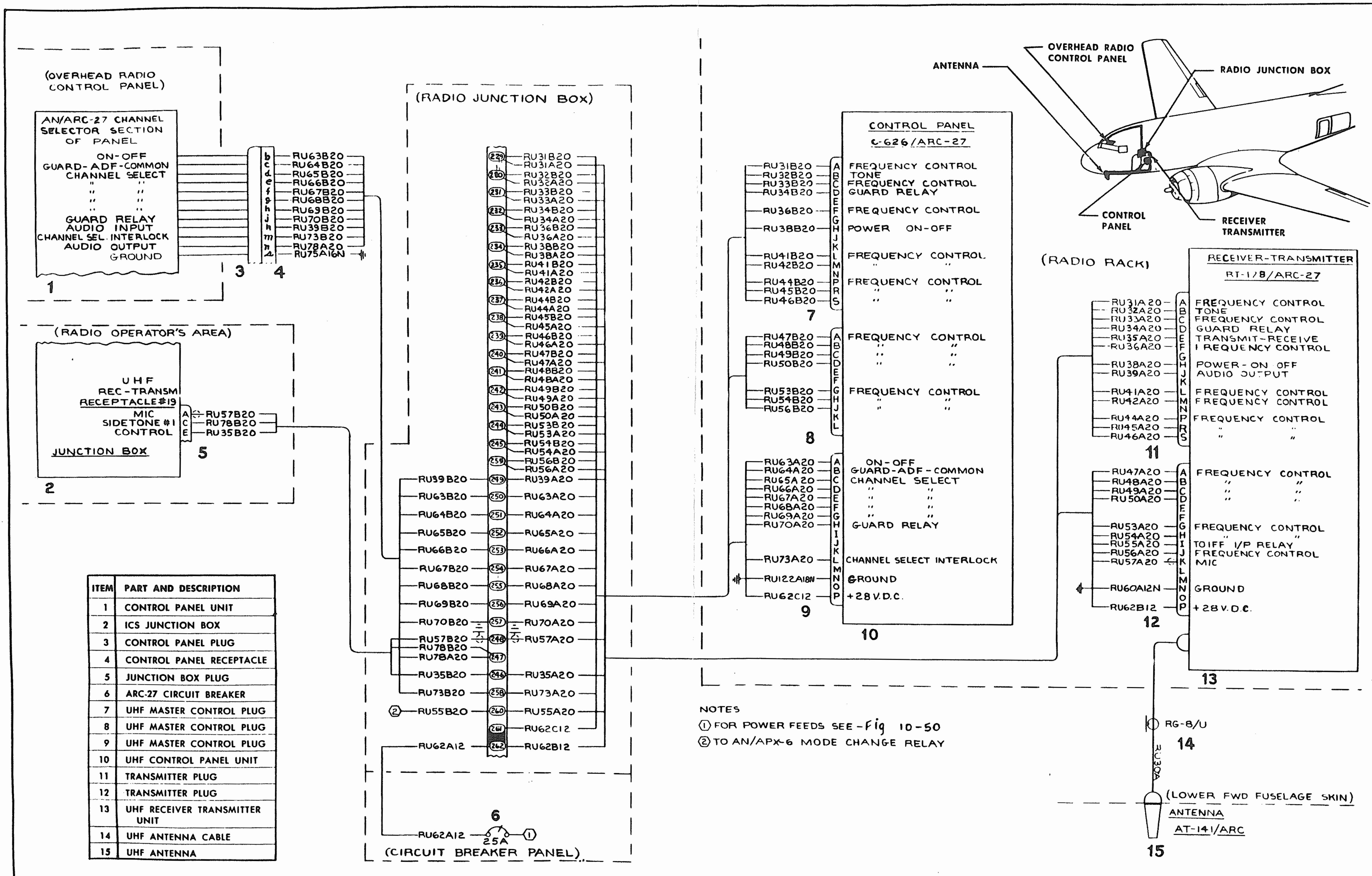
ITEM	PART AND DESCRIPTION
1	PLUG
2	INDICATOR
3	LAMP
4	PLUG
5	INDICATOR
6	RELAY
7	CIRCUIT BREAKER
8	ANTENNA
9	CABLE
10	PLUG
11	PLUG
12	UNIT
13	UNIT

Figure 10-49. Radio Marker Beacon Receiver



ITEM	DESCRIPTION
1	CIRCUIT BREAKER (35A.)
2	CIRCUIT BREAKER (5A.)
3	CIRCUIT BREAKER (10A)
4	FILTER
5	FUSE
6	CIRCUIT BREAKER (25A.)
7	CIRCUIT BREAKER (15A.)
8	RELAY
9	RHEOSTAT
10	FUSE (3 AMP)
11	FUSE (5 AMP)
12	FUSE (1 AMP)

Figure 10-50. Radio Power Feed System

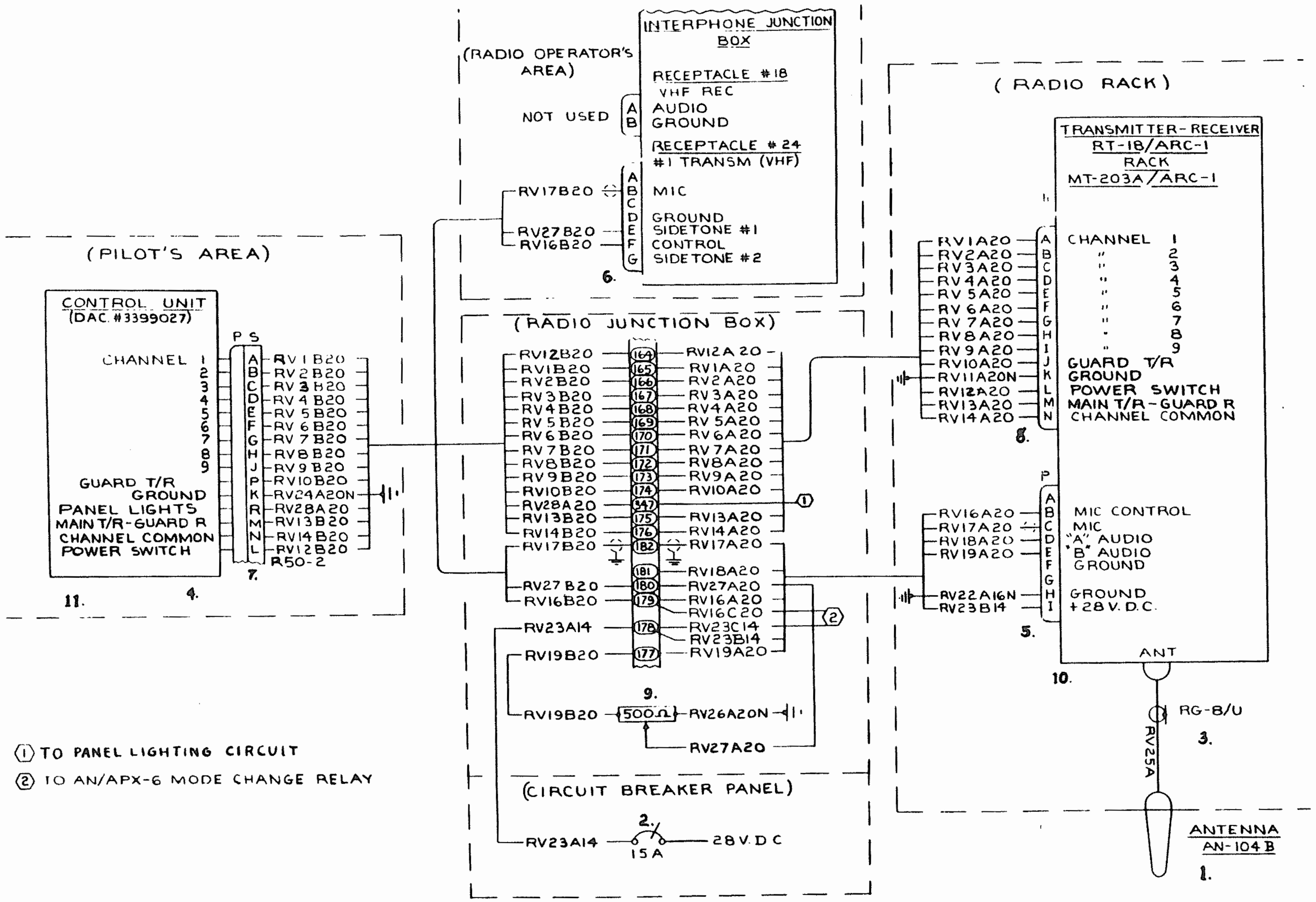


NOTES
① FOR POWER FEEDS SEE - Fig 10-50
② TO AN/APX-6 MODE CHANGE RELAY

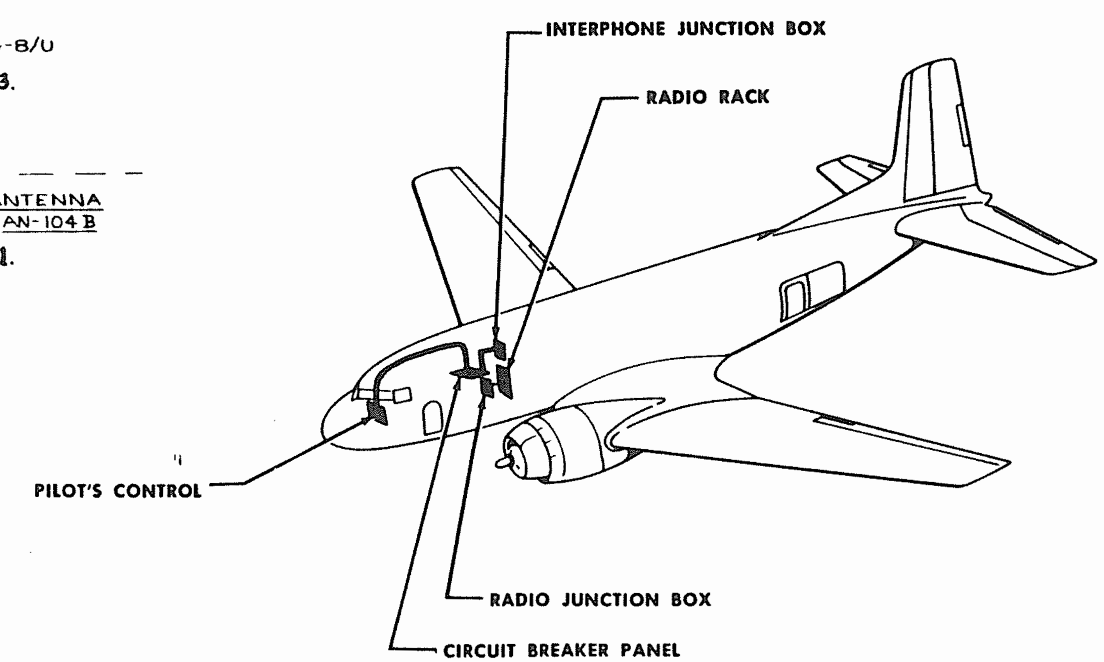
ITEM	PART AND DESCRIPTION
1	CONTROL PANEL UNIT
2	ICS JUNCTION BOX
3	CONTROL PANEL PLUG
4	CONTROL PANEL RECEPTACLE
5	JUNCTION BOX PLUG
6	ARC-27 CIRCUIT BREAKER
7	UHF MASTER CONTROL PLUG
8	UHF MASTER CONTROL PLUG
9	UHF MASTER CONTROL PLUG
10	UHF CONTROL PANEL UNIT
11	TRANSMITTER PLUG
12	TRANSMITTER PLUG
13	UHF RECEIVER TRANSMITTER UNIT
14	UHF ANTENNA CABLE
15	UHF ANTENNA

8013

Figure 10-53. Radio UHF Transmitter-Receiver

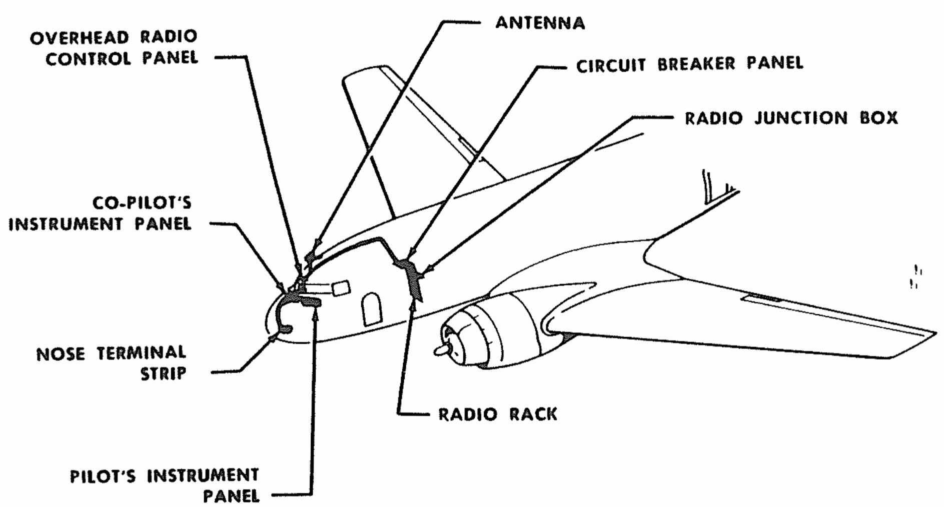


ITEM	PART AND DESCRIPTION
1	VHF ANTENNA
2	CIRCUIT BREAKER (15A)
3	ANTENNA CABLE
4	CONTROL PANEL LAMP
5	POWER PLUG
6	TRANSMITTER PLUG
7	CONTROL UNIT PLUG
8	FREQUENCY CONTROL PLUG
9	VOLUME RESISTOR
10	TRANSMITTER RECEIVER
11	CONTROL UNIT



8014

Figure 10-54. Radio VHF Transmitter-Receiver



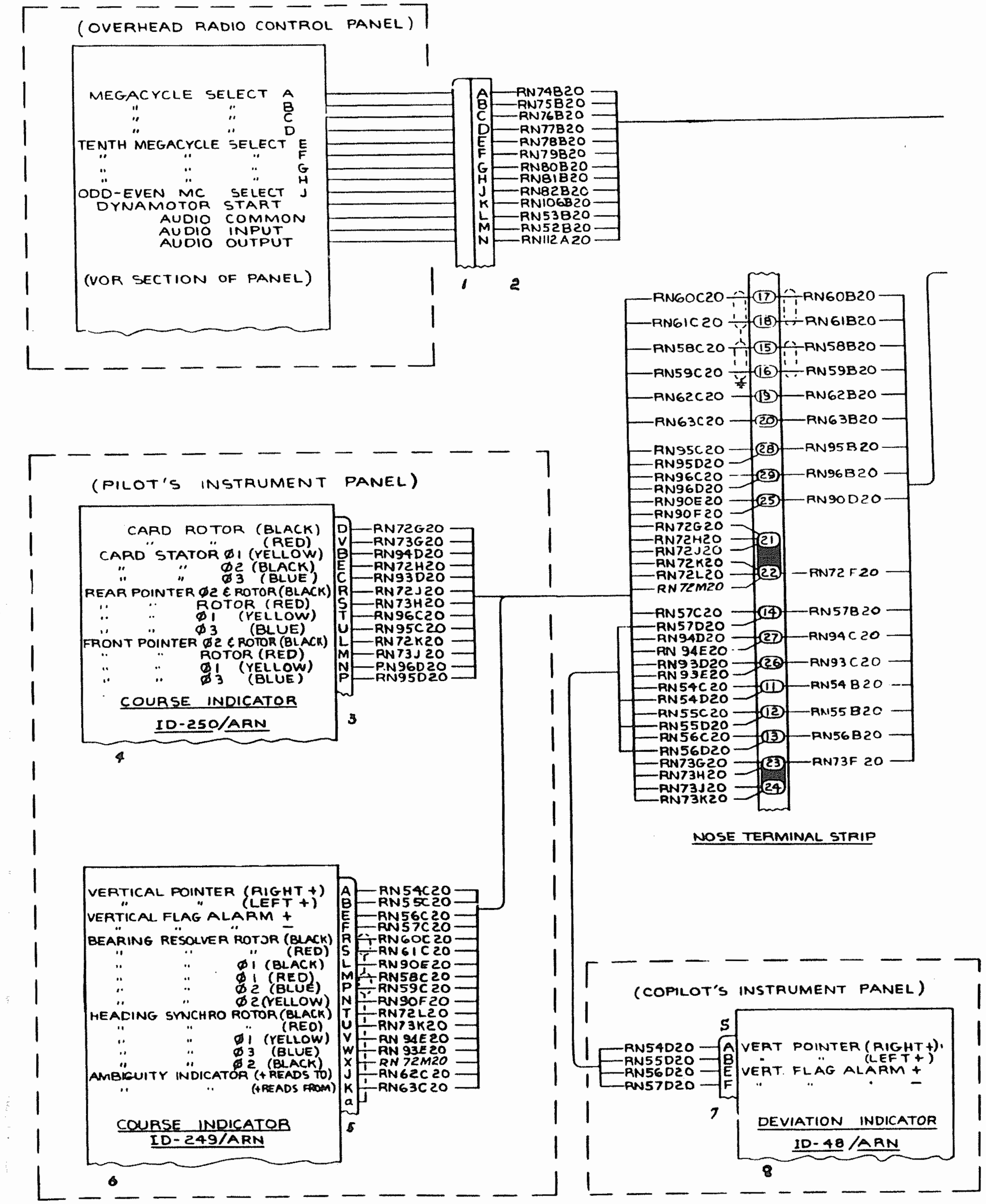
ITEM	PART AND DESCRIPTION
1	PLUG (OVERHEAD CONT PANEL)
2	RECEPTACLE (CONT PANEL)
3	PLUG
4	COURSE INDICATOR
5	PLUG
6	COURSE INDICATOR
7	PLUG
8	DEVIATION INDICATOR
9	CIRCUIT BREAKER
10	FUSE (1 AMP)
11	TRANSFORMER
12	RESISTOR
13	ANTENNA
14	CABLE
15	PLUG
16	ICS JUNCTION BOX
17	PLUG
18	ADAPTOR
19	PLUG
20	INDICATOR
21	PLUG
22	VOR RECEIVER
23	PLUG
24	VOR DYNAMOTOR
25	PILOT'S POINTER SELECTOR
26	CO-PILOT'S POINTER SELECTOR

NOTES:

(1) FOR POWER FEEDS SEE FIGURE 10-48

(2) SEE ELECTRICAL WIRING DIAGRAM FOR AUTO PILOT CONNECTIONS

WIRES MARKED HV TO BE IDENTIFIED FOR HIGH VOLTAGE PER DOUGLAS PROCESS STANDARD 1.34



8015

Figure 10-55. Radio VOR Navigation Equipment (Aircraft A, B, and 1 through 68) — Controls and Indicators

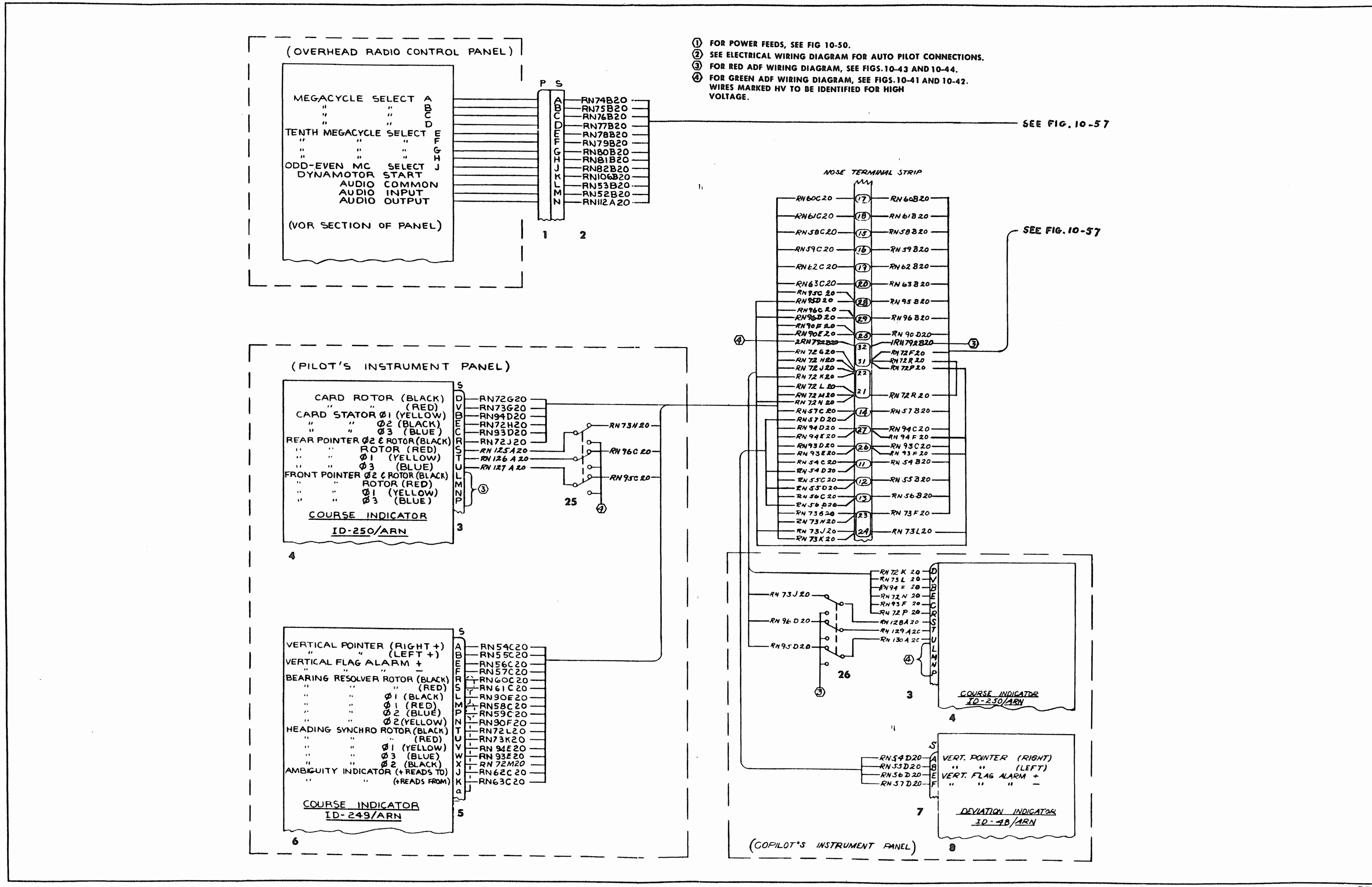


Figure 10-56. Radio VOR Navigation Equipment (Aircraft C, D, and 69 through 96) — Controls and Indicators

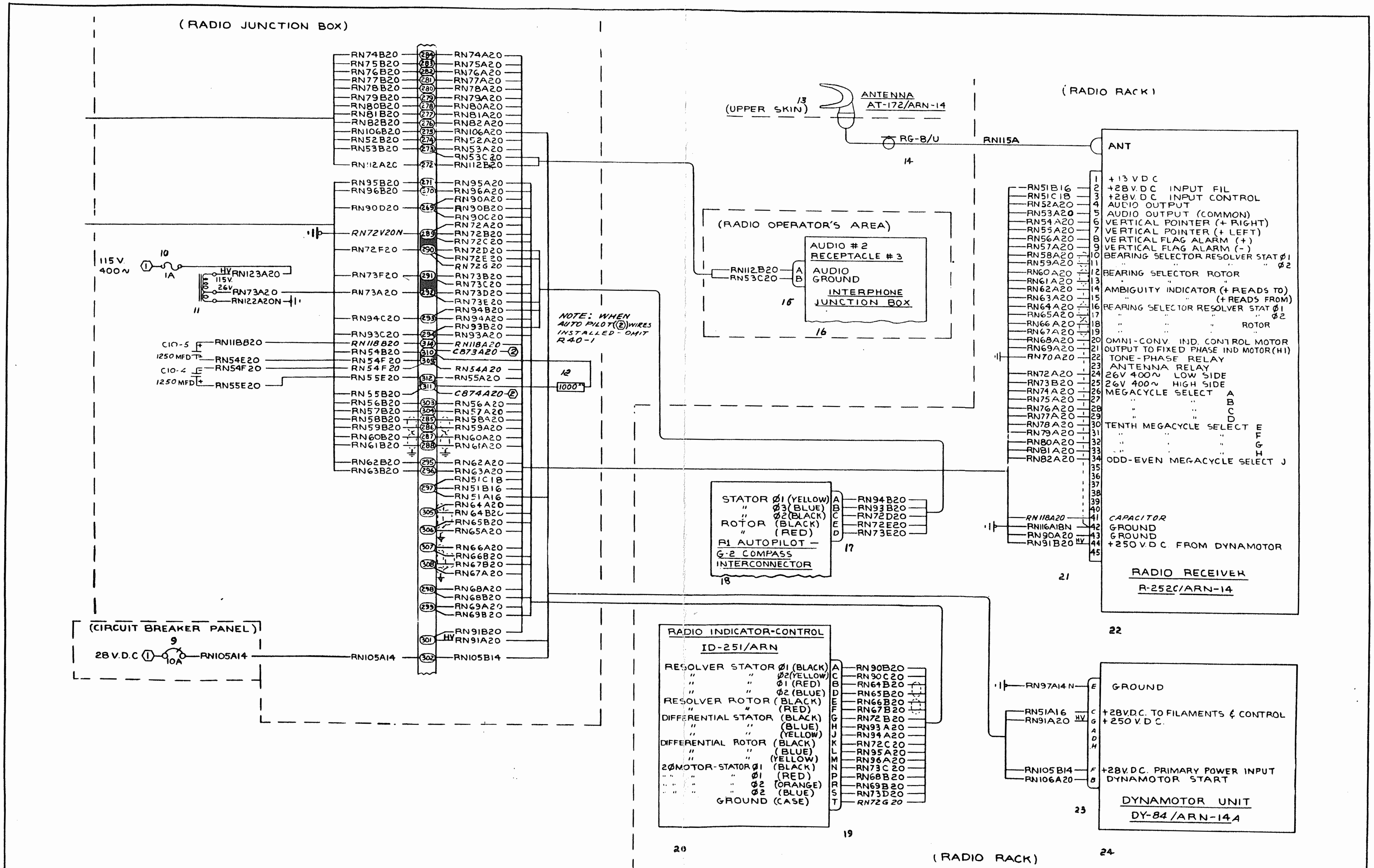
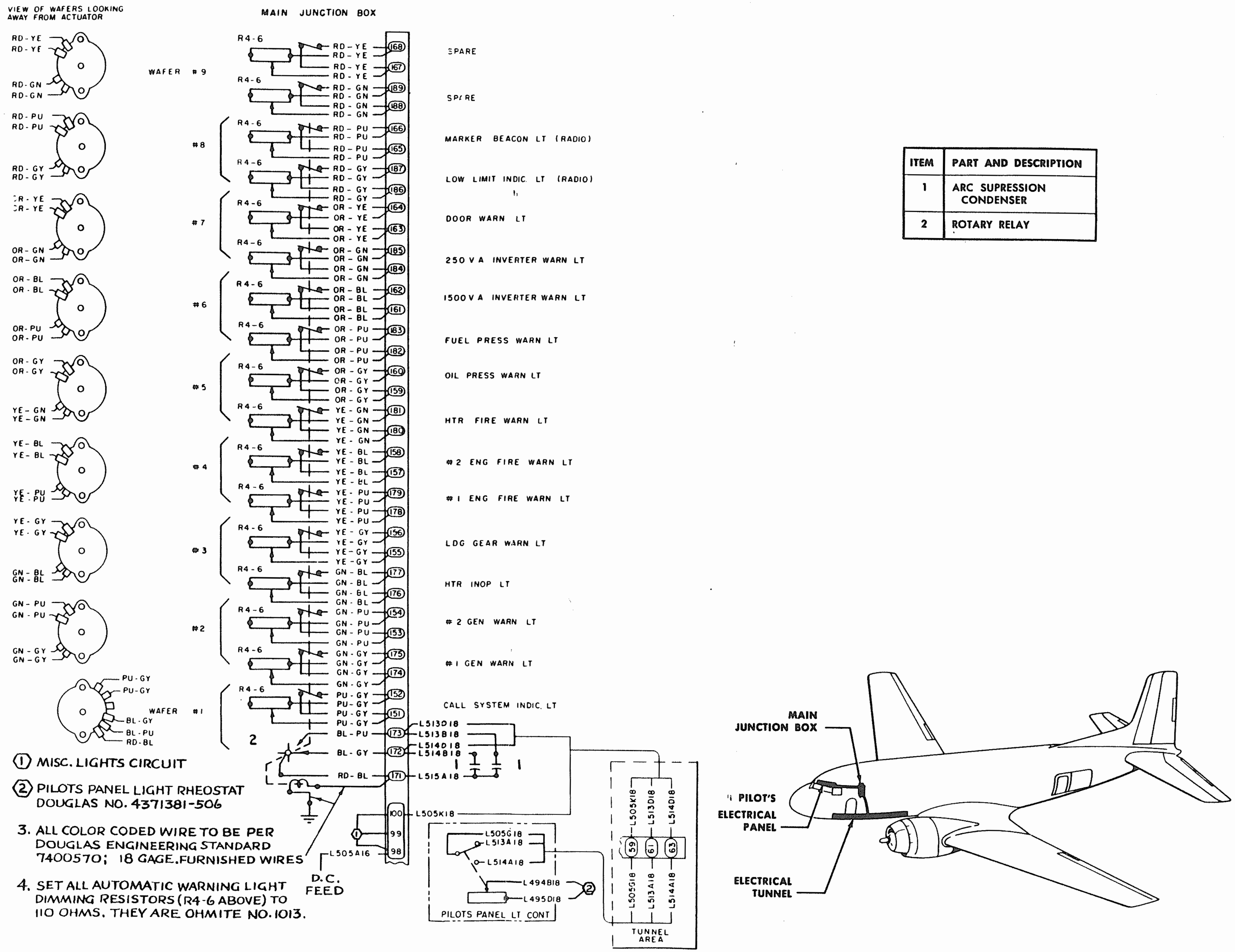


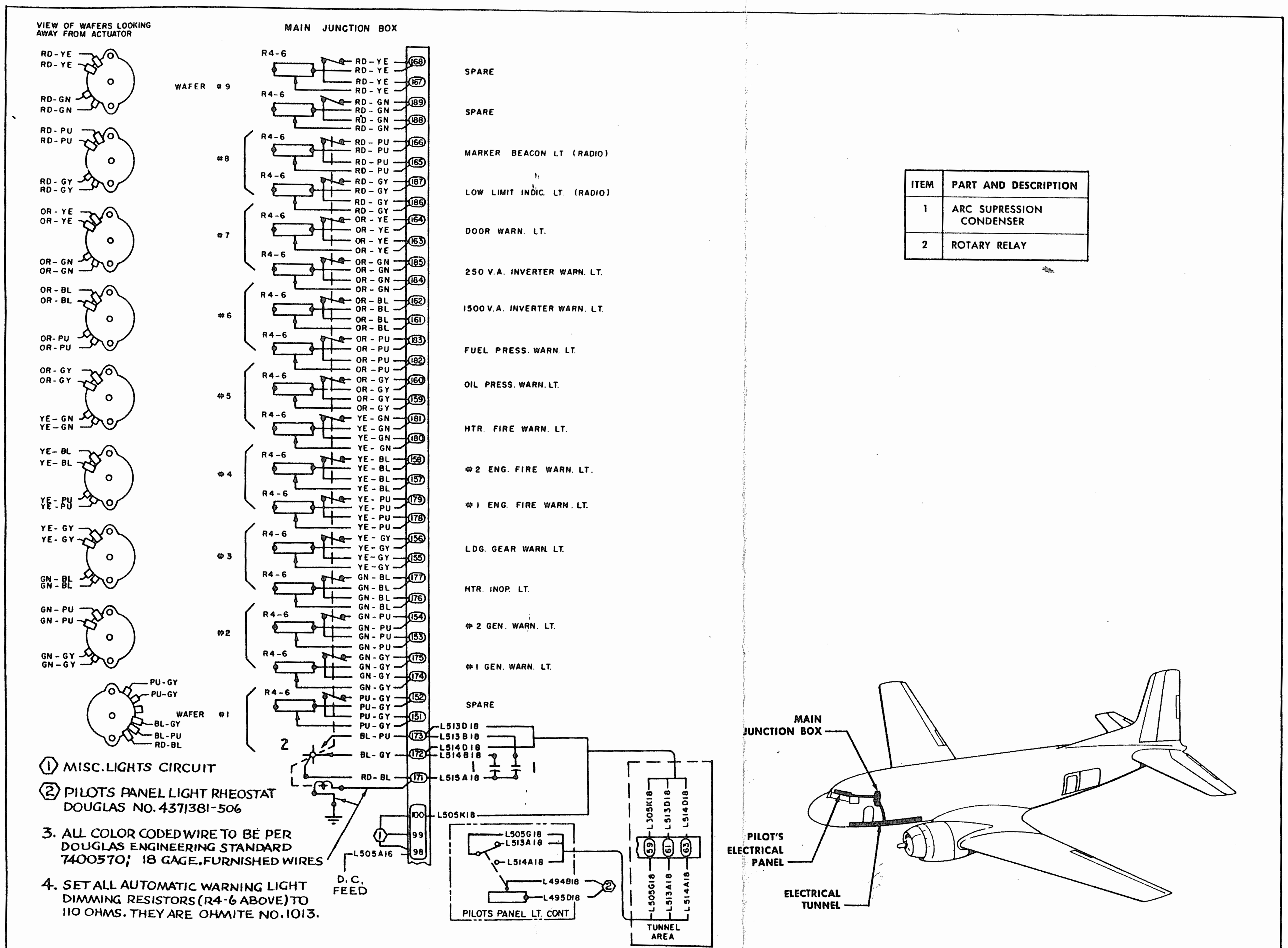
Figure 10-57. Radio VOR Navigation Equipment — Receiver, Power Feed and Junction Box

8016



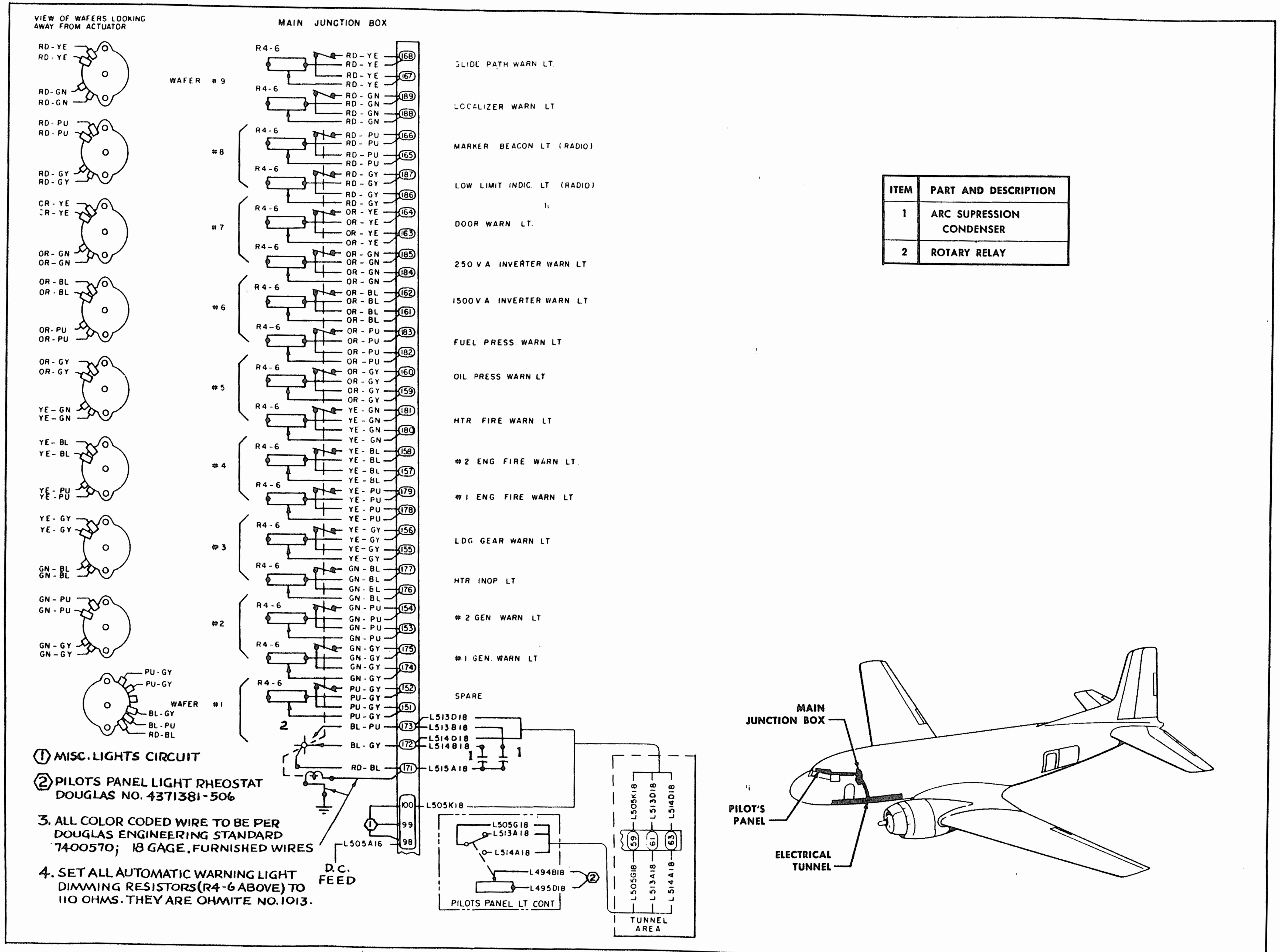
ITEM	PART AND DESCRIPTION
1	ARC SUPRESSION CONDENSER
2	ROTARY RELAY

Figure 10-58. Warning — Automatic Dimming System (Aircraft A and B)



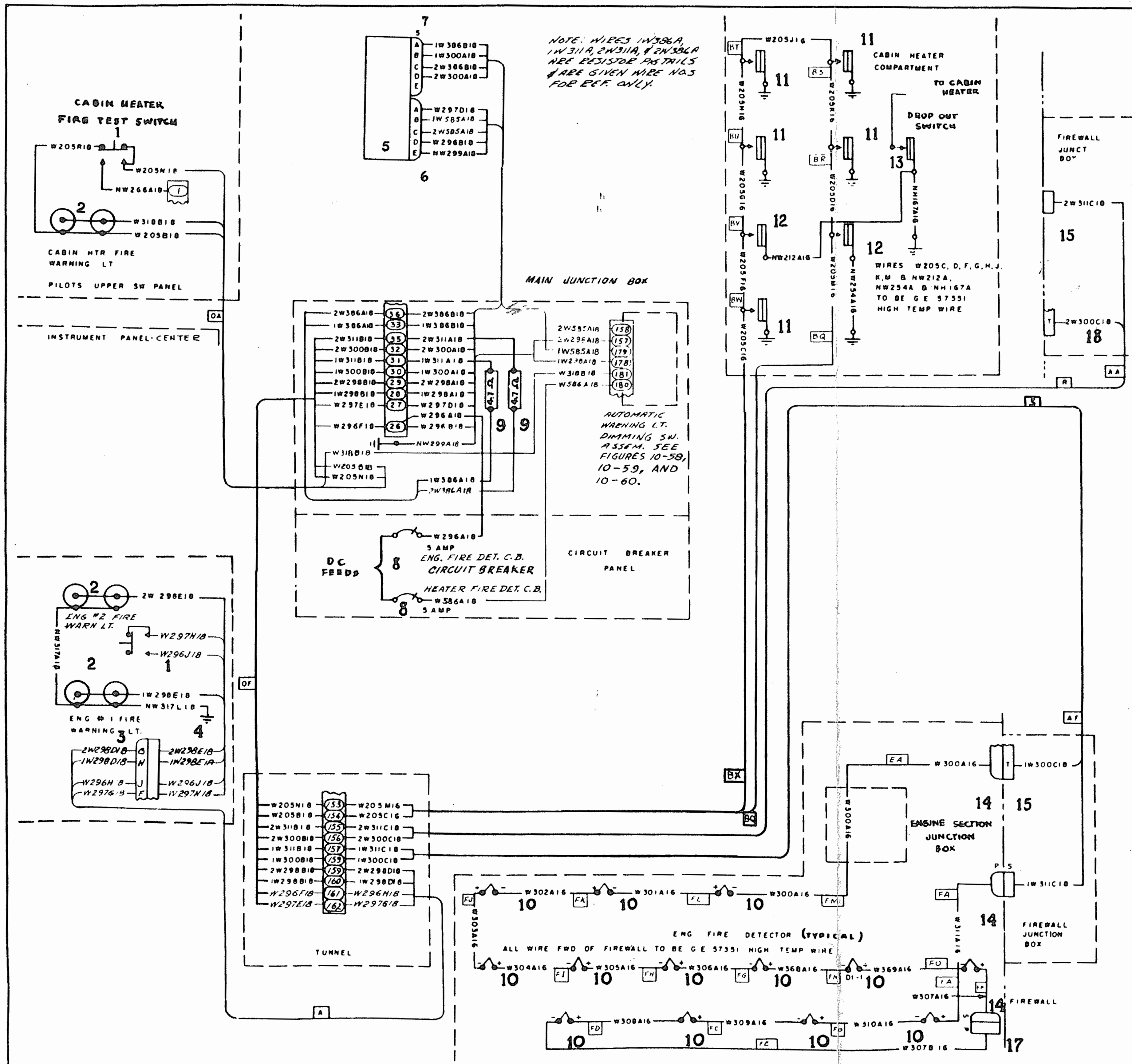
ITEM	PART AND DESCRIPTION
1	ARC SUPRESSION CONDENSER
2	ROTARY RELAY

Figure 10-59. Warning — Automatic Dimming System (Aircraft 1 through 8)

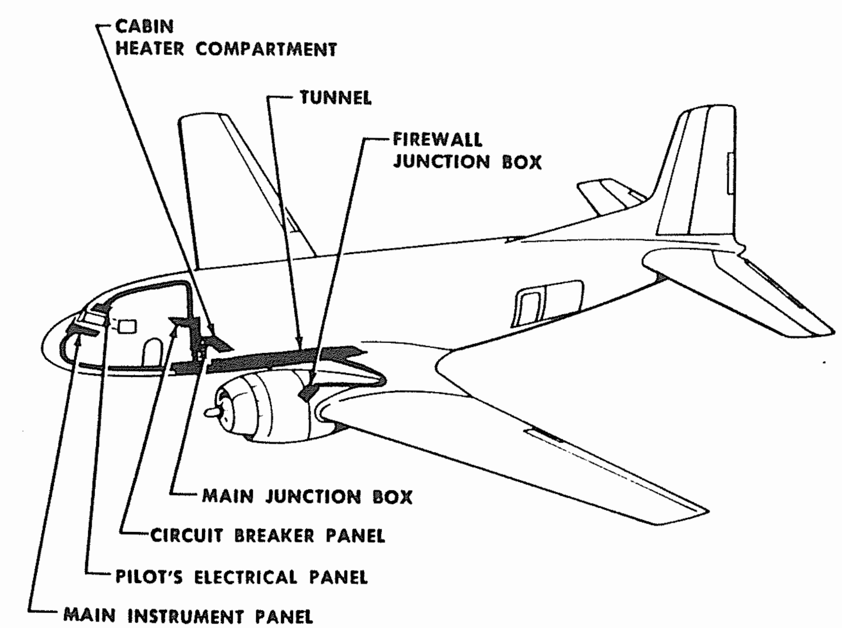


ITEM	PART AND DESCRIPTION
1	ARC SUPRESSION CONDENSER
2	ROTARY RELAY

Figure 10-60. Warning — Automatic Dimming System (Aircraft C, D, and 9 through 96)

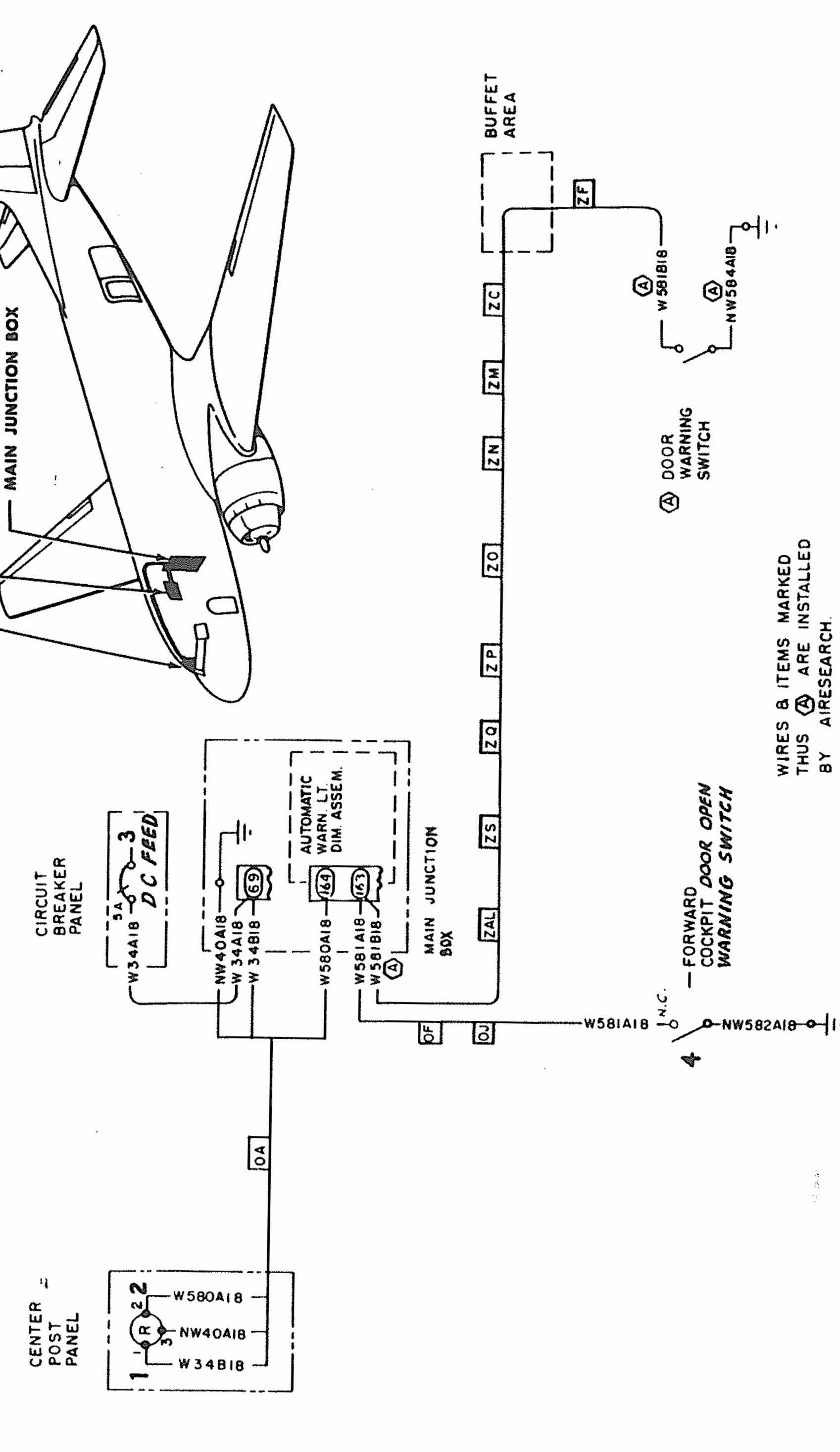


ITEM	PART AND DESCRIPTION
1	SWITCH
2	LAMP ASSEMBLY LAMP
3	PLUG
4	RECEPTACLE
5	FIRE DETECTOR RELAY
6	PLUG
7	PLUG
8	CIRCUIT BREAKER (5A.)
9	RESISTOR
10	FIRE DETECTOR
11	FIRE DETECTOR
12	FIRE DETECTOR
13	SWITCH
14	PLUG
15	RECEPTACLE
16	PLUG
17	RECEPTACLE
18	RECEPTACLE

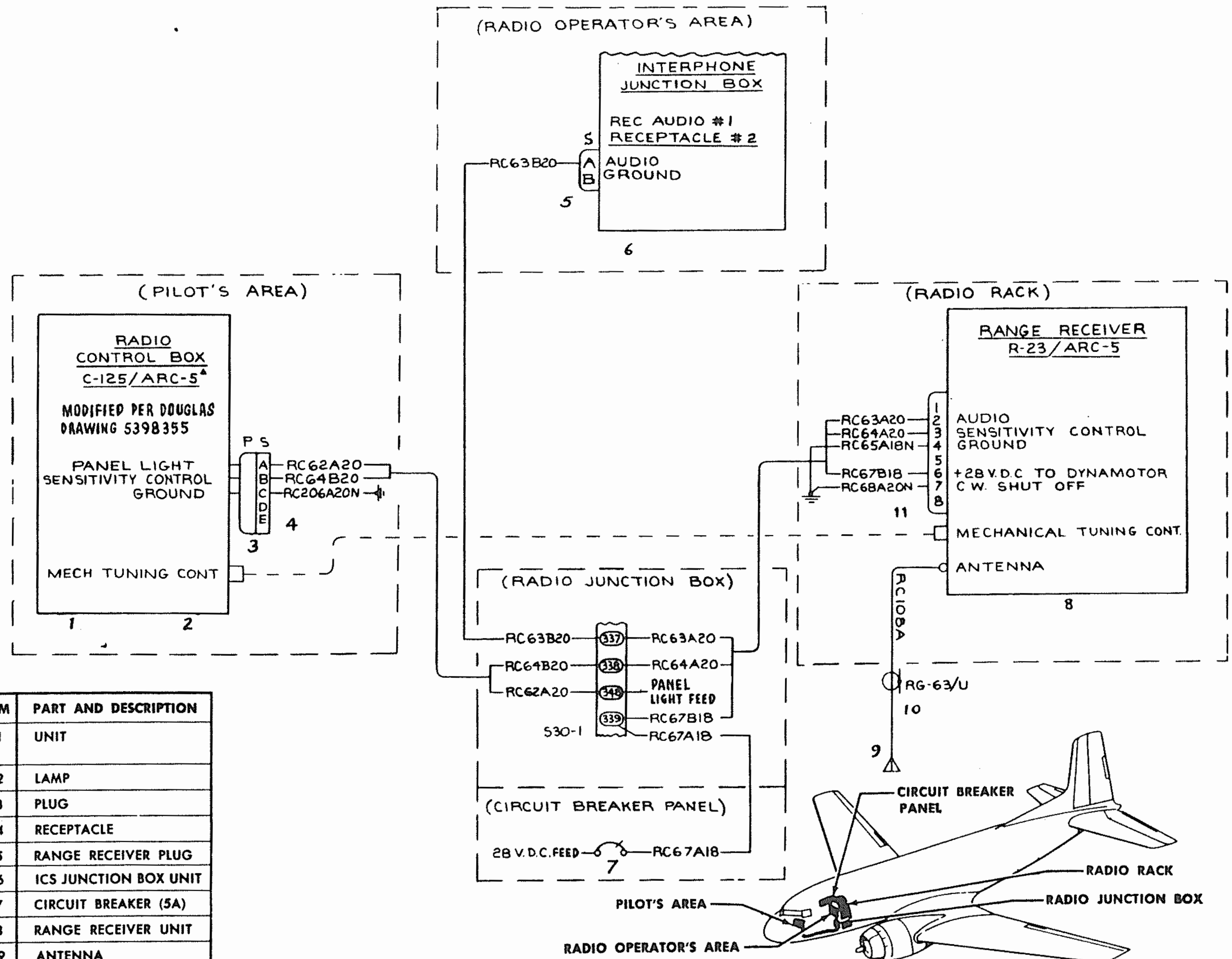


8020

Figure 10-61. Warning — Cabin Heater and Engine Fire



ITEM	PART AND DESCRIPTION
1	LAMP
2	LAMP ASSEMBLY
3	CIRCUIT BREAKER
4	SWITCH



ITEM	PART AND DESCRIPTION
1	UNIT
2	LAMP
3	PLUG
4	RECEPTACLE
5	RANGE RECEIVER PLUG
6	ICS JUNCTION BOX UNIT
7	CIRCUIT BREAKER (5A)
8	RANGE RECEIVER UNIT
9	ANTENNA
10	ANTENNA CABLE
11	PLUG

8011

Figure 10-51. Radio Range Receiver (Aircraft A, B, and 1 through 8)

ITEM	PART AND DESCRIPTION
1	PLUG
2	RECEPTACLE
3	RANGE RECEIVER PLUG
4	ICS JUNCTION BOX
5	CIRCUIT BREAKER
6	PLUG
7	RANGE RECEIVER
8	CABLE
9	ANTENNA

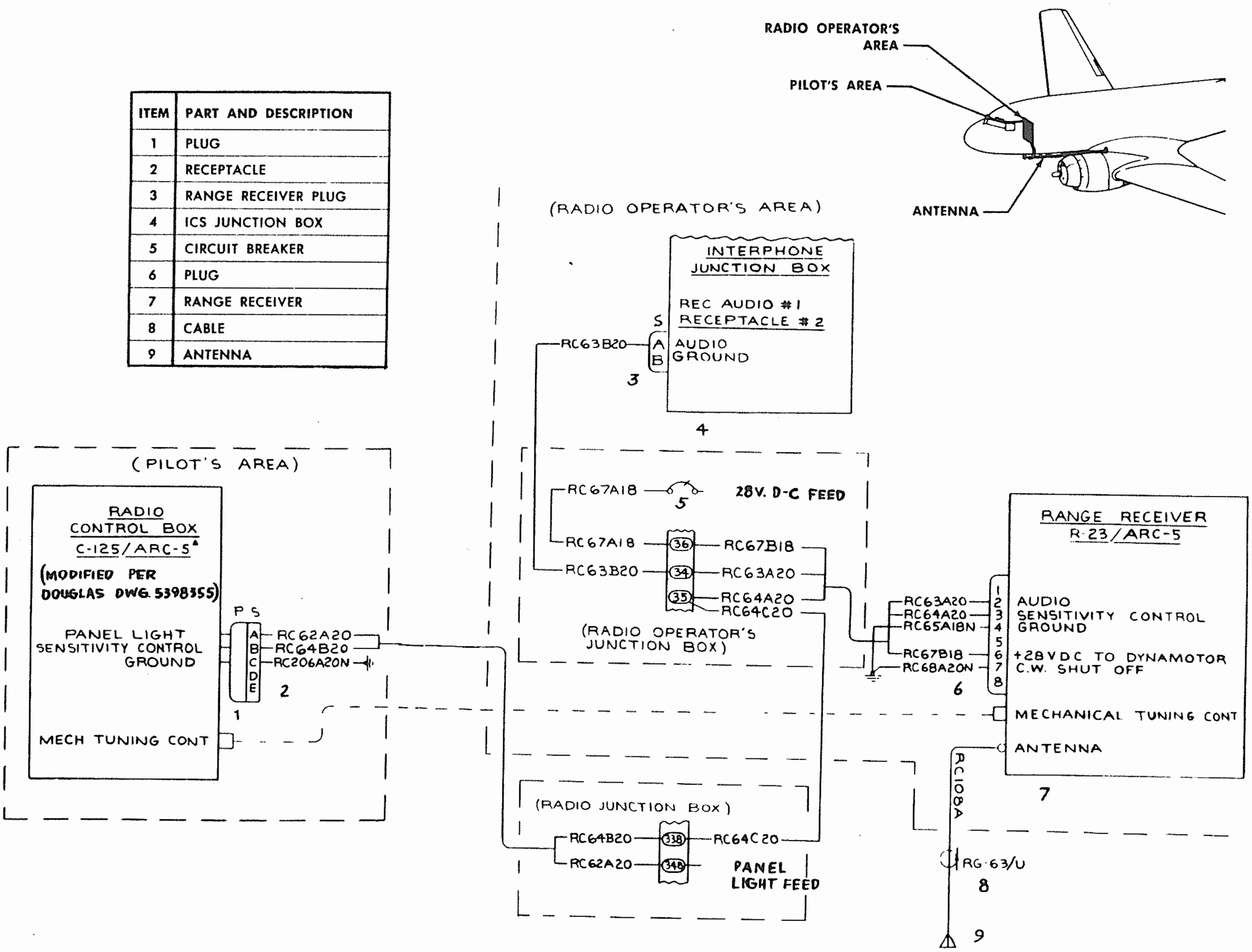


Figure 10-52. Radio Range Receiver (Aircraft C, D, and 9 through 96)

ITEM	PART AND DESCRIPTION
1	LAMP
2	LAMP ASSEMBLY
3	CIRCUIT BREAKER
4	SWITCH
5	SWITCHES

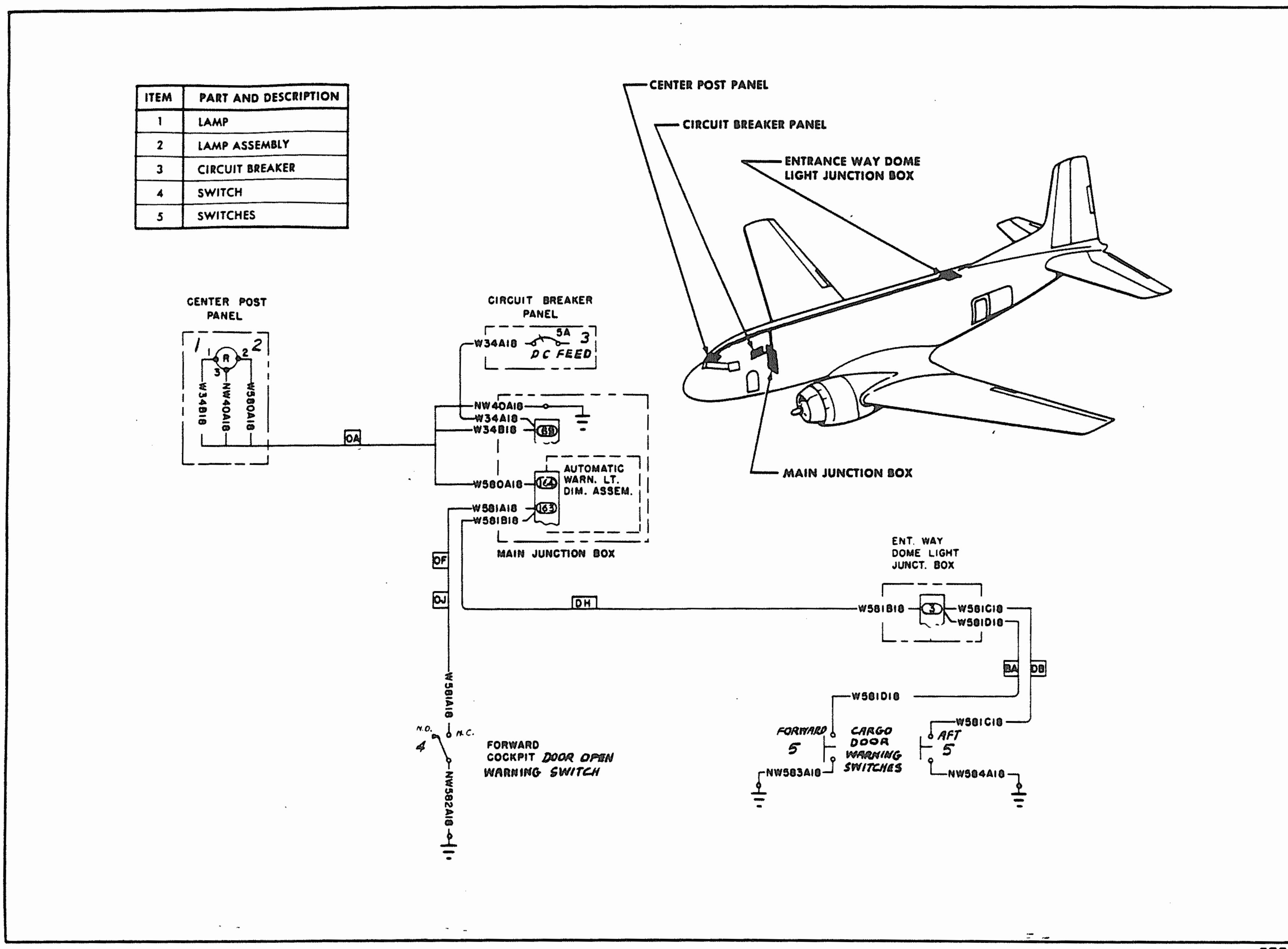
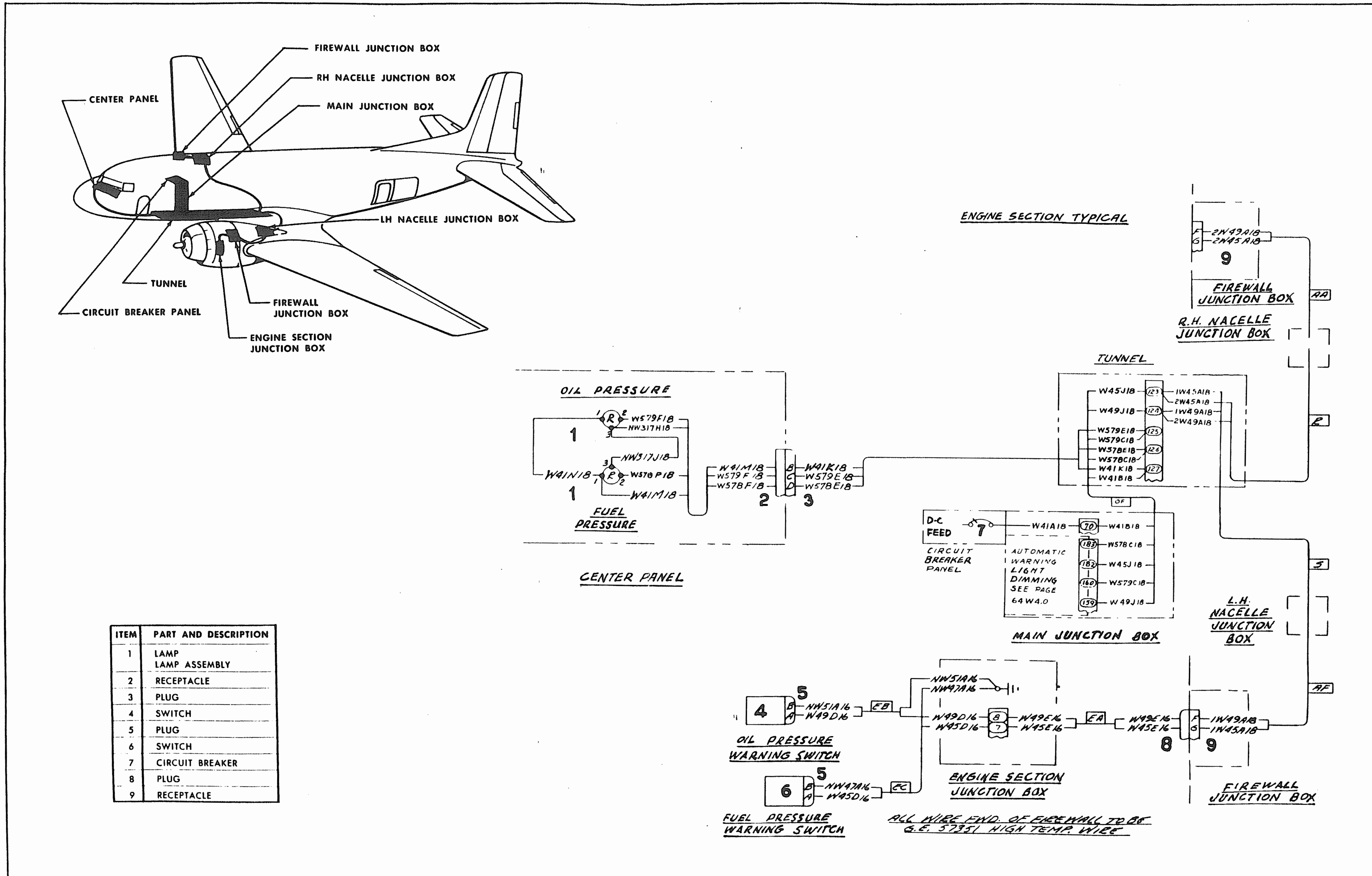


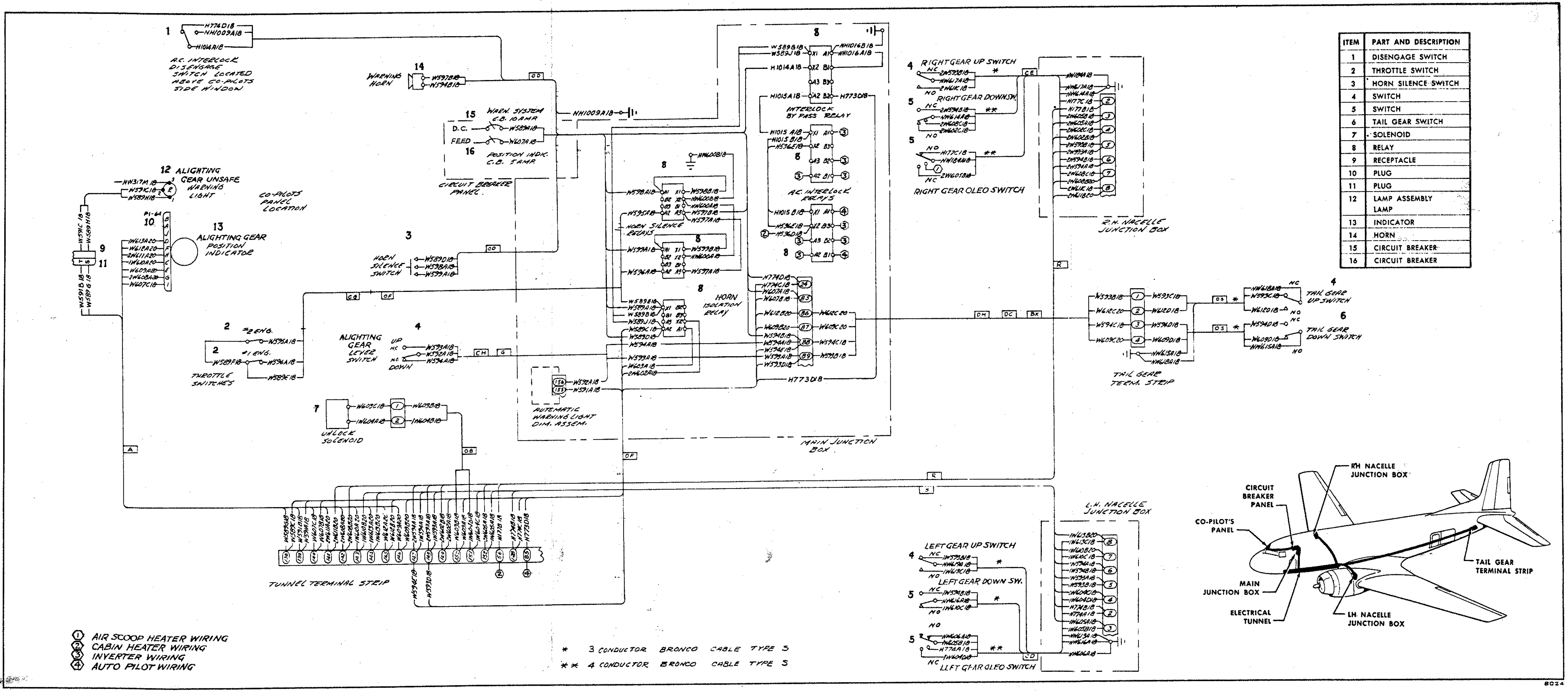
Figure 10-63. Warning — Door Light (Aircraft 1 through 96)



ITEM	PART AND DESCRIPTION
1	LAMP LAMP ASSEMBLY
2	RECEPTACLE
3	PLUG
4	SWITCH
5	PLUG
6	SWITCH
7	CIRCUIT BREAKER
8	PLUG
9	RECEPTACLE

8023

Figure 10-64. Warning — Fuel and Oil Pressure

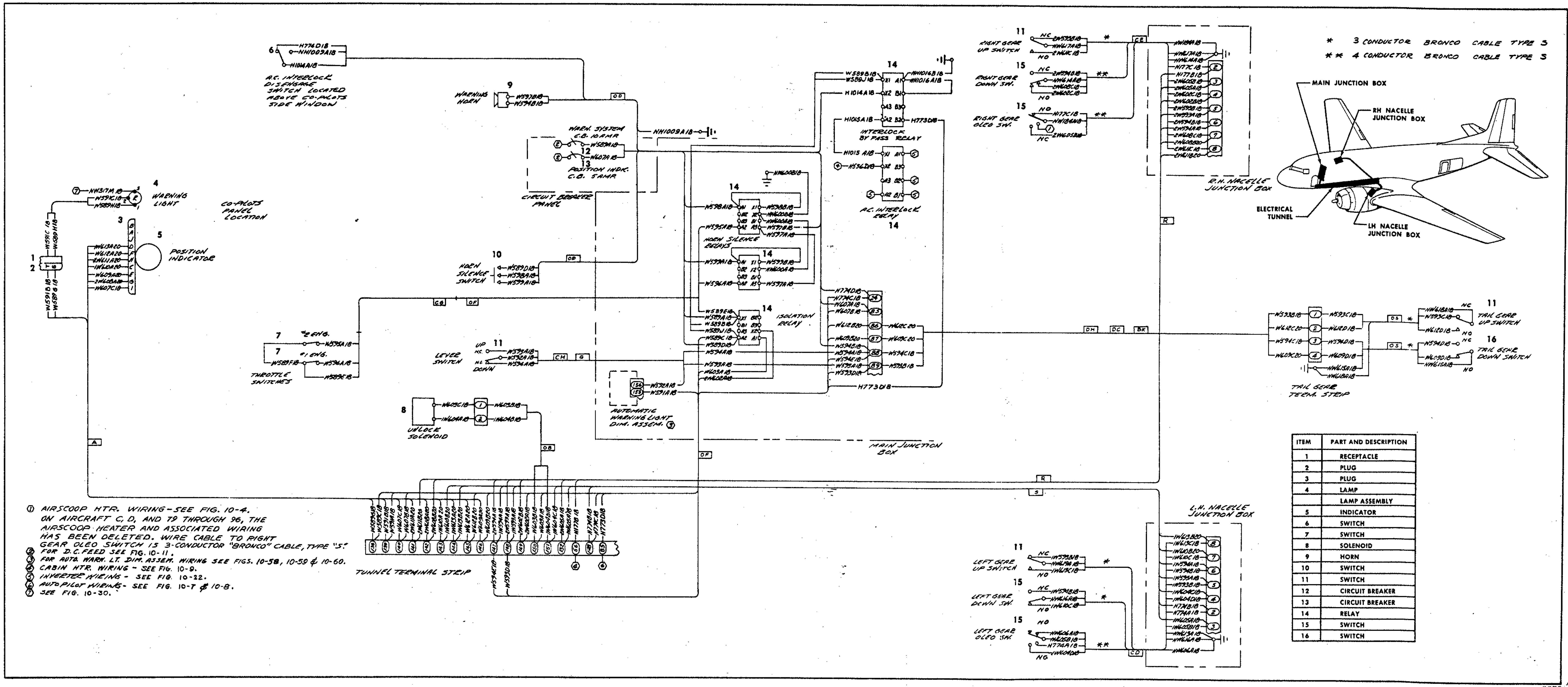


ITEM	PART AND DESCRIPTION
1	DISENGAGE SWITCH
2	THROTTLE SWITCH
3	HORN SILENCE SWITCH
4	SWITCH
5	SWITCH
6	TAIL GEAR SWITCH
7	SOLENOID
8	RELAY
9	RECEPTACLE
10	PLUG
11	PLUG
12	LAMP ASSEMBLY LAMP
13	INDICATOR
14	HORN
15	CIRCUIT BREAKER
16	CIRCUIT BREAKER

1. AIR SCOOP HEATER WIRING
2. CABIN HEATER WIRING
3. INVERTER WIRING
4. AUTO PILOT WIRING

* 3 CONDUCTOR BRONCO CABLE TYPE S
** 4 CONDUCTOR BRONCO CABLE TYPE S

Figure 10-65. Warning — Alighting Gear (Aircraft A, B, and 2 through 26)



① AIRSCOOP HTR. WIRING—SEE FIG. 10-4.
ON AIRCRAFT C, D, AND 79 THROUGH 96, THE AIRSCOOP HEATER AND ASSOCIATED WIRING HAS BEEN DELETED. WIRE CABLE TO RIGHT GEAR OLEO SWITCH IS 3-CONDUCTOR "BRONCO" CABLE, TYPE "S."
② FOR D.C. FEED SEE FIG. 10-11.
③ FOR AUTO. WARN. LT. DIM. ASSEM. WIRING SEE FIGS. 10-58, 10-59 & 10-60.
④ CABIN HTR. WIRING — SEE FIG. 10-9.
⑤ INVERTER WIRING — SEE FIG. 10-22.
⑥ AUTO PILOT WIRING — SEE FIG. 10-7 & 10-8.
⑦ SEE FIG. 10-30.

ITEM	PART AND DESCRIPTION
1	RECEPTACLE
2	PLUG
3	PLUG
4	LAMP
	LAMP ASSEMBLY
5	INDICATOR
6	SWITCH
7	SWITCH
8	SOLENOID
9	HORN
10	SWITCH
11	SWITCH
12	CIRCUIT BREAKER
13	CIRCUIT BREAKER
14	RELAY
15	SWITCH
16	SWITCH

Figure 10-66. Warning — Alighting Gear (Aircraft C, D, 1 and 27 through 96)

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ALPHABETICAL INDEX

A

* **Indicates items discussed under two or more aircraft effectivity groups**
Italics Indicate Illustrations

B

C

*** Indicates items discussed under two or more aircraft effectivity groups**
Italics Indicate Illustrations

C *(Continued)*

D

Indicates items discussed under two or more aircraft effectivity groups
Italics Indicate Illustrations

E

F

F (Continued)

G

H

I

* Indicates items discussed under two or more aircraft effectivity groups
Italics Indicate Illustrations

I (Continued)

J

L

* Indicates items discussed under two or more aircraft effectivity groups
Italics Indicate Illustrations

M

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O

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P (Continued)

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*** Indicates items discussed under two or more aircraft effectivity groups**
Italics Indicate Illustrations

S (Continued)

T

Indicates items discussed under two or more aircraft effectivity groups
Italics Indicate Illustrations

U

V

W

** Indicates items discussed under two or more aircraft effectivity groups*
Italics Indicate Illustrations